REPORT NO. F694
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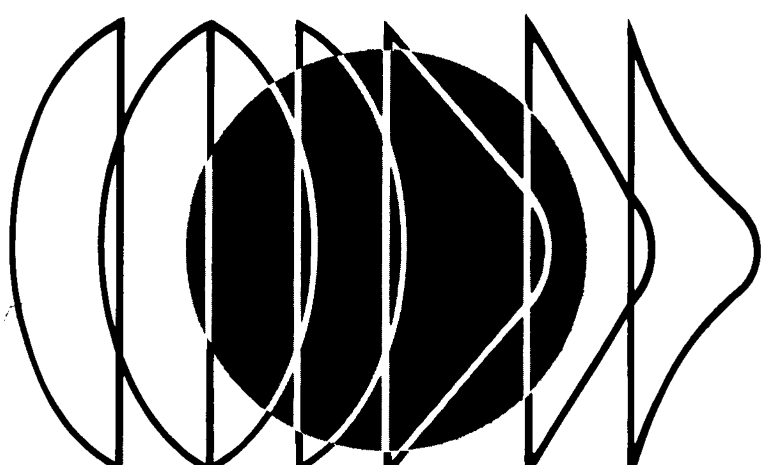
PART A INTRODUCTION AND SUMMARY
PART B MISSION OBJECTIVES AND REQUIREMENTS
PART C CONSTRAINTS
PART D SELECTED DESIGN CONCEPTS

VOYAGER CAPSULE PHASE B FINAL REPORT



VOLUME IV ENTRY SCIENCE PACKAGE

PREPARED FOR:
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MCDONNELL ASTRONAUTICS

REPORT ORGANIZATION

VOYAGER PHASE B FINAL REPORT

The results of the Phase B VOYAGER Flight Capsule study are organized into several volumes. These are:

Volume I	Summary
Volume II	Capsule Bus System
Volume III	Surface Laboratory System
Volume IV	Entry Science Package
Volume V	System Interfaces
Volume VI	Implementation

This volume, Volume IV, describes the McDonnell Douglas selected design for the Entry Science Package. It is arranged in 11 parts, A through K, and bound in 4 separate documents, as noted below.

Part A	Introduction and Summary	
Part B	Objectives and Requirements	
Part C	Design Criteria and Constraints	1 Document
Part D	Selected Design Concept	
Part E	Alternatives and Systems Analysis	
Part F	Future Mission Options	1 Document
Part G	Subsystem Equipment	1 Document
Part H	Reliability	
Part I	Planetary Quarantine	
Part J	Operational Support Equipment	1 Document
Part K	Interface Alternatives	

In order to assist the reader in finding specific material relating to the Entry Science Package, Figure 1 cross indexes broadly selected subject matter, at the system and subsystem level, through all volumes.

VOLUME IV CROSS REFERENCE INDEX

ITEM		PART A INTRODUCTION AND SUMMARY	PART B OBJECTIVES AND REQUIREMENTS	PART C DESIGN CRITERIA AND CONSTRAINTS	PART D DESCRIPTION OF PREFERRED CONFIGURATION	PART E PRINCIPAL ALTERNATIVES AND SELECTION FACTORS
ESP ASPECT						
MISSION	Objectives	Sec. 1.0	1.0	–	–	–
	Operations (& Profiles)	2.0 6.3	2.0	–	2.0–d 4.1 5.0	5.1 5.8
DESIGN	Configuration	3.0 6.2	√	√	1.0	1.1
	Functional	4.0 5.0 6.0	√	√	1.0 2.0 3.0 4.0	1.0 2.0 3.0 4.0 5.0
	Weight	3.0 4.0	–	√	1.4 2.0	3.0 d
Interfaces (See Also Volume V)		Introduction	–	1.0 2.0 3.0	1.3 2.0–d 3.0–d 4.0 5.3.4	1.1 3.0–d 4.1.3 5.0
Implementation (See Also Volume VI		–	–	–	6.0	–
ESP EQUIPMENT OR SUBSYSTEM						
Imaging		Sec. 2.0 4.0 6.0	1.2 2.2	6.1	1.2.1 2.1 4.3.2.4	1.1 2.2 3.2 5.3.2.6
Atmospheric Properties Measurement		2.0 4.0 6.0	1.1	6.2	1.2.2 2.0 4.0–d	1.1 2.1 3.1 5.0–d
Engineering Instrumentation		–	–	–	Figure 3.0–1	Figure 4.2–3 4.2.2.2
Data Storage		6.0	–	–	2.5.3 3.2.1	4.2.5
Telecommunications		5.1	–	6.3	2.5.3 3.2	4.2
Power		5.2	–	6.3	2.5.1 3.1	4.1
Structural/Mechanical		–	–	4.0	2.5.4 3.3	4.3
Cabling and Packaging		–	–	5.0 6.3	2.5.1 3.4	4.4
Thermal Control		5.3	–	6.3	2.5.7 3.5 4.3.2	4.5

√ Denotes that the part or section generally applies to the topic.
d Denotes that the topic is distributed throughout the part or section.

Figure 1

PART F	PART G	PART H	PART I	PART J	PART K
FUTURE MISSION OPTIONS	DETAILED DESCRIPTION OF ESP EQUIPMENT	RELIABILITY	PLANETARY QUARANTINE	OPERATIONAL SUPPORT EQUIPMENT	INTERFACE ALTERNATIVES
√	–	–	–	–	–
√	d	2.3.3 3.1.1	2.0	4.3 4.4 4.5 8.0	–
–	d	–	3.0	OSE Config	1.0 2.0
√	d	1.0 2.0 3.3	–	OSE Functional Design	1.0 2.0
–	d	2.3.2	–	–	–
-	d	–	–	\	1.0 2.0 3.0
–	–	–	d	–	1.0 3.0
√	1.1	3.3	3.2	5.5	1.2
√	1.2 1.3 1.4 1.5	3.3	–	5.5	1.2
–	2.0	3.3	–	d	–
–	3.0	3.3	3.4	5.4	–
√	4.0 5.0 6.0	3.3	–	5.4	2.0
–	7.0	3.3	3.4	5.3 4.4.8	2.0
–	8.0	–	3.1	–	–
–	9.0	–	–	–	–
–	10.0	3.3	3.3	5.6	2.0

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2-1 through 2-3	2-1 through 2-7	10 through 11	2-1 through 2-31
3-1 through 3-5	3-1 through 3-2	12 through 13	3-1 through 3-19
4-1 through 4-2		14 through 15	4-1 through 4-46
5-1 through 5-4			5-1 through 5-16
6-1 through 6-9			6-1 through 6-30

PART A

INTRODUCTION AND SUMMARY

The VOYAGER Entry Science Package (ESP) has been studied and a design concept selected for consideration by NASA and the Jet Propulsion Laboratory. In accordance with the 12 June 1967 VOYAGER Capsule Systems Constraints and Requirements Document (Reference A-1), a major consideration has been to maintain independence, to the extent practicable, between the Entry Science Package, the Surface Laboratory (SL), and the Capsule Bus (CB).

However, McDonnell recognizes that the overall purpose of the VOYAGER Project is the gathering of a sufficient quantity of high-quality scientific data to provide the scientific community with information to analyze the atmospheric, surface, and life-associated characteristics of Mars. With this objective foremost in mind, we considered the various Capsule Bus/Entry Science Package alternatives and selected that interface approach which favored the obtaining of reliable science data even at the price of some additional system weight or interface complexity, but without jeopardizing overall system reliability and probability of mission success.

The ESP 'preferred approach' which we derived in this Phase B study had to satisfy two groups of mission objectives and constraints. The first group is primarily program-oriented:

a. Planetary Quarantine

b. Inviolate launch window

c. Long-life reliability (Pre-launch storage time plus 7 to 9 months transit time)

d. Mission environment, including heat sterilization and ETO decontamination.

The second group is essentially ESP science-oriented:

a. Compatibility with the Capsule Bus's basis aerodynamic and physical characteristics.

b. Design of Capsule Bus to assure the obtaining of high-quality atmospheric and imaging science data.

c. Selection of instruments, measurement times, and data reduction procedures that are compatible with the Capsule Bus atmospheric entry profile and with the range of postulated Mars atmospheric models.

The ESP design and operational mode derived during our Phase B study satisfies all of these constraints and objectives. Three ESP interfaces alternatives were also considered, as follows:

a. Complete independence

b. Integration with the Surface Laboratory System

c. Integration with the Capsule Bus System

As a result, it is our recommendation that the ESP scientific objectives of the program and the implementation of the ESP/CB scientific, operational, and physical interfaces can best be served by having the CB contractor assume both technical and management responsibilities, including interface responsibilities as follows:

a. The definition and technical management of the interfaces

b. The integration of the system test and the supporting OSE

c. The selective integration of some ESP elements within the Capsule Bus System.

This Volume presents the results of our Phase B ESP study. Presented herein are the study's science objectives, the design and environmental constraints employed, the design and operational mission alternatives considered, and, finally, a description of the preferred design, its functional operation, and the rationale for its selection. Specifically, Parts B, C, and D describe objectives and requirements, constraints, and our selected concept, respectively. Parts E and F contain the discussion of alternatives considered, analysis, and a brief indication of future mission options. Part G contains more comprehensive functional descriptions of the major equipments and subsystems for the selected design concept. Parts H, I, and J discuss reliability, planetary quarantine, and operational support equipment, respectively. Following in the same document, Part K discusses interfaces and alternatives for ESP management and technical interface responsibility.

SECTION 1

OBJECTIVES AND CONSTRAINTS

On the basis of the JPL documentation made available to the Phase B Capsule Bus contractors, the science objectives of the Entry Science Package are seen as being the following:

a. Atmospheric properties determination, including:

- the altitude profile of atmospheric density, pressure, and temperature, and
- the atmospheric composition (at Mach 5 and lower).

b. visual imaging

The general purpose of these measurements is:

a. Provide scientific data for use in gaining a better understanding of the static and dynamic properties, and improved inferences on the history, of the Martian atmosphere.

b. Provide a sufficiently detailed model of the Mars atmosphere as to permit more efficient entry vehicle design and provide a higher probability of success with future unmanned and manned Mars Landers.

c. Provide increased information concerning the surface characteristics of Mars, and

d. Improve the probability of success of the 1973 mission by (a) permitting the second capsule entry and landing sequence to be programmed with less uncertainty, (b) increasing the interpretability of both the Surface Laboratory and the Orbiter imaging data, and (c) serving as a back-up to the Surface Laboratory with regard to measuring the atmospheric properties at the surface of Mars.

Specifically, the first objective of the atmospheric properties determination function of the Entry Science Package is to obtain data for the reconstruction of the Martian atmospheric density/altitude profile. Success will permit the next capsule to be operated with greater reliability and precision in addition to permitting subsequent capsules to be designed for more efficient operation. Accurate profile determination requires accurate trajectory and dynamic pressure reconstruction through the zone of significantly high aerodynamic deceleration of the capsule. Accuracy and reliability are enhanced by the use of multiple independent sources of atmospheric data, and by using direct methods to determine altitude. The order of priority of atmospheric regions for initial determination of atmospheric properties for meteorological and atmospheric physics purposes, is considered to be: (1) troposphere and stratosphere, (2) surface, and (3) above stratosphere thru the ionosphere. Thus the post touchdown measurements of pressure, temperature, and composition, while the Entry Science Package/Spacecraft relay communications link is functional, are considered initially to have a higher priority than very high altitude composition measurements.

The desired descent imaging sequence starts with a series of images of the planet limb, taken prior to decelerating entry, for stereoscopic views of the planetary surface area including horizon and landing site and of any horizon clouds. The desired sequence continues with minimum gaps to near touchdown. Such a series of images of changing area coverage and resolution from entry to touchdown not only permits study of surface characteristics and elevation profiles, but also provides for determination of landing site location in the context of Orbiter-derived surface maps. Photometric and color data are desired adjuncts to the immagery. The availability of some higher resolution overlapping descent images for a small fraction of the area imaged at lower resolution from the Orbiter, should enhance the value of the Orbiter images, and visa versa.

Objectives for use of the ESP for future missions include descent imaging of new planetary surface areas, high altitude atmospheric composition determination, additional data for indication of space/time changes in the lower atmosphere, and post-touchdown panoramic imaging with transmission over the UHF relay link.

Major constraints on the selection of a design approach that satisfies the foregoing design objectives are provided by the Jet Propulsion Laboratory Constraints Document (Reference A-1). The reference document specifies (1) the assumed entry science instrumentation, (2) simplicity of interface with the Capsule Bus and Surface Laboratory, and (3) the spread of design atmospheres. The compatibility of our design concept with the Reference A-1 constraints and requirements document is indicated in Figure A-1.

The spread of possible entry velocity which can occur due to the Capsule Bus's allowable range of de-orbit conditions, when coupled with the VM-1 thru VM-10 series of design atmospheres, imposes a serious requirement for ESP operational flexibility. Other Capsule Bus-imposed constraints derive from (1), the Capsule's general configuration and mass properties, (2) the sequence for parachute deployment, Aeroshell-Lander separation and terminal thrusting, and (3) the uncertainties in the Capsule's aerodynamic coefficients, flow field ralationships, dynamic properties, and entry conditions. Any possibility of the CB heat shield's products of ablation affecting the ESP's science measurements are minimized by the use of ceramic rather than

COMPLIANCE OF MCDONNELL DOUGLAS ESP DESIGN WITH JPL "1973 VOYAGER CAPSULE SYSTEMS CONSTRAINTS AND REQUIREMENTS DOCUMENT" (REVISION 2, 12 JUNE 1967)

JPL DOCUMENT		MCDONNELL DOUGLAS ESP DESIGN	
REF. PARA.	APPLICABLE STATEMENT	DESIGN FEATURE	APPL. RPT. SECTION
3.1.1 (a)	Objectives of the CB system are to : (a) obtain data on the Martian environment during entry and landing.	Pressure and temperature sensors, mass spectrometer, tri-axis accelerometer, and imaging will be used to characterize the Martian environment.	Pt. D-2 Pt. E-2, 3 Pt. G-1
3.2.2	To the extent practicable, the CB, SL, and ESP shall be mutually independent, separable and self-supporting.	To assure high quality data, CB-located ESP equipment consists of: (a) 1 press. and 1 temp. transducer located at CB stag. pt., (b) accelerometer near CB c.g., (c) imaging cameras located outboard and parellel to roll axis, and (d) remainder of science instrument plus ESP-supporting equipment, all contained in ESP Principal Unit Package. ESP provides its own independent power, thermal control and telemetry, and is completely self-supporting.	Pt. D-1 Pt. E-4.3 Pt. G-8 Pt. K
3.2.10	It is a design requirement that no potential single failure mode shall cause a catastrophic effect on the mission.	Selective redundance utilized within ESP to prevent single-point failures. System uses SLS redundant battery as backup to ESP power; data interleaving of CB and ESP low rate data, etc.	Pt. D-3 Pt. E-4 Pt. G-4 to 7 Pt. H
3.3.5.2.4	The ESP comprises the science instruments required to perform the entry experiments and all other equipment required to support the entry science experiments and transmit entry science data.	The ESP design presented in this volume complies with this definition.	Vol. IV
3.4.1	CB-SL and CB-ESP physical interfaces shall each consist of a structural field joint and an electrical connector. The CB design shall provide for routing of SL and ESP umbilical functions to SC.	CB-ESP interface described above in Item 2. ESP Principal Unit interface consists of field joint, electrical connectors and CB thermal curtain/ESP UHF antenna. CB provides for routing of ESP umbilical functions to SC.	Pt. D-1 Pt. E-4.3 Pt. G-8 Pt. K
3.4.3.4	FC equipment designed to enter the Martian atmosphere shall be heat-sterilized such that the probability that a live organism will survive the serilization is less than 10^{-3}.	Selected ESP equipment can satisfy this criteria.	Pt. C Pt. I Pt. H-5
3.5.1	Functional Requirement The ESP shall be an automatic device to: (a) Perform all entry science measurements (b) Transmit all entry science data	The ESP design presented in this volume complies with this definition.	Vol. IV
3.5.2	Performance Requirement	Science Inst. 27.0 lb	Pt. A-3

Figure A-1

	ESP shall accommodate at least 45 lb. of science instruments and science support equipment. ESP shall be capable of transmitting at least 5×10^6 bits of science data during entry and descent	Telecomm. S/S. . . 55.0 lb (on CB) Power S/S ..... 22.5 lb Largest data contributor is entry TV (from 800,000' to 90'). For shortest entry time to 90' ($\approx$315 sec), total data output $\approx 16 \times 10^6$ bits of science data during entry and descent.	Pt. A-4 Pt. D-3.2 Pt. E-4.2 Pt. G-4
3.6.1	To the extent practicable the SL and CB shall be mutually independent, the physical interface shall consist of a structural field joint and appropriate electrical connectors.	See Item 5 above for description of physical interface.	
4.1.3.3	<u>CB-ESP Interface</u> CB-ESP interface shall be standardized for a number of missions to the maximum extent practical. Thus interface provisions shall anticipate changes, including possible deletion, of the ESP for subsequent planetary opportunities.	CB-ESP interface specifically designed to provide standardization for a number of missions and also to allow: (a) Removal without requiring any major modification to CB design (b) Capability for growth if decision is made by NASA to extend scope of atmospheric definition during 1975-79 opportunities.	Pt. F Pt. K
4.1.3.3.1	<u>Power</u> 1. Power for battery charging shall be supplied to ESP from SC during cruise. This power shall be routed through the CB.	1. This requirement is complied with by the use of the SC Power Distribution Bus located in the CB.	Pt. D-3.1 Pt. E-4.1 Pt. G-7
	2. SC will provide the turn-on and turn-off capabilities of the power supplied to the ESP.	2. SC CC&S will be utilized to provide the turn-on and turn-off function for the power supplied to the ESP.	
	3. The SC to ESP power distribution circuit shall be such that no SC or ESP switching condition or single-failure mode will allow the ESP to supply power to the SC.	3. A DC to DC converter will be used to provide grounding isolation and SC power regulation. This will not allow the ESP to supply power to SC under any condition.	
	4. Short circuit protection of the CB power supplied to the ESP shall be provided by the CB. Fuses shall not be used.	4. No CB power is supplied to ESP, CB only routes SC-supplied power. Short circuit protection shall be provided by current sensors and switch.	
4.1.3.3.4	<u>Control and Sequencing</u> An initiation command shall be supplied to the ESP from the CB.	Signal from SC turns on CB Sequencer and Timer. After a preset time delay CB S&T initiates ESP warm-up and provides capability to initiate ESP turn-off 10 min. after impact.	Pt. D-3
4.3	<u>ESP Subsystem Constraints</u>		
4.3.1	<u>Science Instruments</u> For purposes of the CB design, the following hypothetical instruments shall be assumed as typical of the capability to be provided.		

Continued on Next Page.

COMPLIANCE OF MCDONNELL DOUGLAS ESP DESIGN WITH JPL "1973 VOYAGER CAPSULE SYSTEMS CONSTRAINTS AND REQUIREMENTS DOCUMENT" (REVISION 2, 12 JUNE 1967)
(Continued)

JPL DOCUMENT				MCDONNELL DOUGLAS ESP DESIGN	
APPLICABLE STATEMENT				DESIGN FEATURE	APPL. RPT. SECTION
Instrument Description	Wt. (lb.)	Power (w)	Volume (cu. in.)		
1. Entry TV Dual one inch vidicons with 200 x 200 picture elements, 6 bits per element. Format size is 0.44 in. Obtain resolutions identifiable with orbiter TV down to 1 meter per line pair. Camera optical axis must be parallel to Capsule roll axis.	14	20	700	1. Entry TV Two cameras, one high-resolution (8° field of view) and one wide angle (50° field of view) used for imaging. Functional capabilities and physical characteristicts same as descirption in JPL Document.	Pt. A-4 Pt. D-2.1 Pt. E-2.2 Pt. G-1.1
2. Accelerometers Accel. experiment is utilized to measure atmos. density profile.. 3 accel. mounted normal to each other with their respective axes parallel to roll, pitch and yaw axes ahead of and near the C.G. of	2	2	15	2. Accelerometers Tri-axis accelerometer located ahead of and near the C.G. is utilized to measure required profile. System provides following capability: Roll Axis: Dual Range – 0 to 5 g 0 to 30 g Pitch & Yaw Axes: ±2 g Physical characteristics: 2 lb, 4 watts, 9.6 cu. in.	Pt. A-4 Pt. D-2.3 Pt. E-3.1 Pt G-1.2

1-4-1

entry capsule. Full scale range shall be from 0 to 30 Earth g.					
3. Pressure Transducers An experiment to determine the atmos. pressure profile shall consist of two pressure transducers	1 (ea.)	0.5 (each)	2 (each)	Two pressure transducers utilized to measure stag. pt. and base region pressures. Measurement range: Stag. Pt. – 0 to 3 psia Base Region – 0 to 0.5 psia Physcial characteristics (each) – 1 lb, 1.4 watts, 6.3 cu. in.	Pt. A-4 Pt. D-2.3 Pt. E-3.1 Pt. G-1.3
4. Temperature Probes The atmos. temp. profile may be measured at Mach 5 or less by use of two temp. probes such as thermistors or platinum resistance thermometers	0.5 (ea.)	0.1 (each)	3.0 (each)	Two tempe.uture probes (platinum resistance thermometers) utilized to measure stag. pt. and base region temperatures starting at Mach 5. Measurement range: Stag. pt. – 150 to 1200° K Base Region – 150 to 330° K Physical Characteristics (each): 0.5 lb, 0.01 watts, 1.7 cu. in.	Pt. A-4 Pt. D-2.3 Pt. E-3.1 Pt. G-1.4
5. Mass Spectrometer The atmos. composition will be determined at Mach 5 or less with a quadripole or magnetic sector mass spectrometer capable of covering a mass range from 10 to 60 Data rate shall be 400 bits per sample with a 2 sec scan time	8	7	200	Quadripole mass spectrometer utilized for measuring atmos. composition starting at Mach 5. Samples obtained from both stag. pt. and base region ports. Functional capabilities and physical characteristics same as description in JPL Document.	Pt. A-4 Pt. D-2.4 Pt. E-3.1 Pt. G-1.5

gure A–1 (Continued)

ablative heat protection material on the spherical nose portion of the aeroshell and surrounding the entry imaging system's optical window. Similarly, constraints imposed by the dynamic behavior of the Capsule Bus are alleviated by the use of Capsule Bus attitude rate damping, and by designing the Aeroshell so that it transmits to the ESP accelerometer only a low level of vibration due to aerodynamic buffet.

For the specified 'typical' science instruments and mode of operation, weight in particular constrains the maximum amount of imaging data obtained during descent, the extent of the post-touchdown operations, the simplicity of the mechanical interface between the ESP and the Capsule Bus, and the type and extent of redundancy within the EPS.

Various candidate design features were studied in some depth during the Phase B Study which have special significance with regard to the JPL reference Document constraints concerning (a) independence of ESP/CB/SLS, (b) single structural attachment for ESP, and (c) assumed 'typical' instruments. These subject design features include the following:

a. Interleaving of low rate ESP data and Capsule Bus data for transmission over separate and redundant transmitters. (Recommended by incorporation in 'preferred approach'.)
b. Utilization of redundant source of electrical power from Surface Laboratory as backup for ESP power. (Recommended by incorporation in 'preferred approach'.)
c. Single data-routing interface unit on board the spacecraft between the Radio subsystems of ESP and Capsule Bus and Spacecraft telemetry. (Recommended by incorporation in 'preferred approach'.)
d. Multiple ESP equipment units, requiring separate mounting, consisting of a single principal unit and three separately mounted sensor units. Physically separated ESP components operate, however, as a functional and electrical unit. (Recommended for incorporation in 'preferred approach'.)
e. Mass spectrometer sampling permitting operation throughout the trajectory starting at the beginning of the continuum flow region (Not included in 'preferred approval' design concept, but recommended as a priority addition experiment).
f. The use of additional science instrumentation (Not included in 'preferred approach' design concept, but a group of priority addition experiments is recommended).

SECTION 2

MISSION PROFILE

The sequence of events in the Entry Science Package's mission profile and the associated zones of operation for the science instruments are shown in Figure A-2. Post warm-up data transmission from all science instruments, except the stagnation point temperature and the mass spectrometer, is initiated at 800,000 feet. The latter two instruments are programmed to start operation 15-40 seconds after peak dynamic pressure. The velocity at this point, for the range of possible entry trajectories will be approximately Mach 5.

REPRESENTATIVE ALTITUDES FOR MACH 5

VM-Atmosphere	Surface Pressure, mb	γ_e, deg.	V_e, fps.	Alt., ft.
8	5	-10.9	13,000	53,500
9	20	-10.9	13,000	175,000
8	5	-20	15,000	33,000
9	20	-20	15,000	140,000

The TV operation is conducted with slow scan vidicons and it is continuous until shortly before touchdown (cameras are jettisoned about 90 feet above the surface) with 5 second intervals between images from alternate cameras. All science data, except TV, is stored during the communication blackout period. The anticipated altitude range for blackout is indicated below.

BLACKOUT ALTITUDES

VM-Atmospheres	γ_e, deg.	V_e, fps.	Blackout Alt., Ft. Start	Finish	Comments
9	-10.9	13,000	462,500	250,000	Earliest emergence
2	-10.9	13,000	170,500	93,000	Shortest Altitude Interval
8	-20	15,000	167,500	59,000	Latest Emergence
3	-20	15,000	435,000	130,000	Longest Altitude Interval

MISSION PROFILE FOR ENTRY SCIENCE PACKAGE

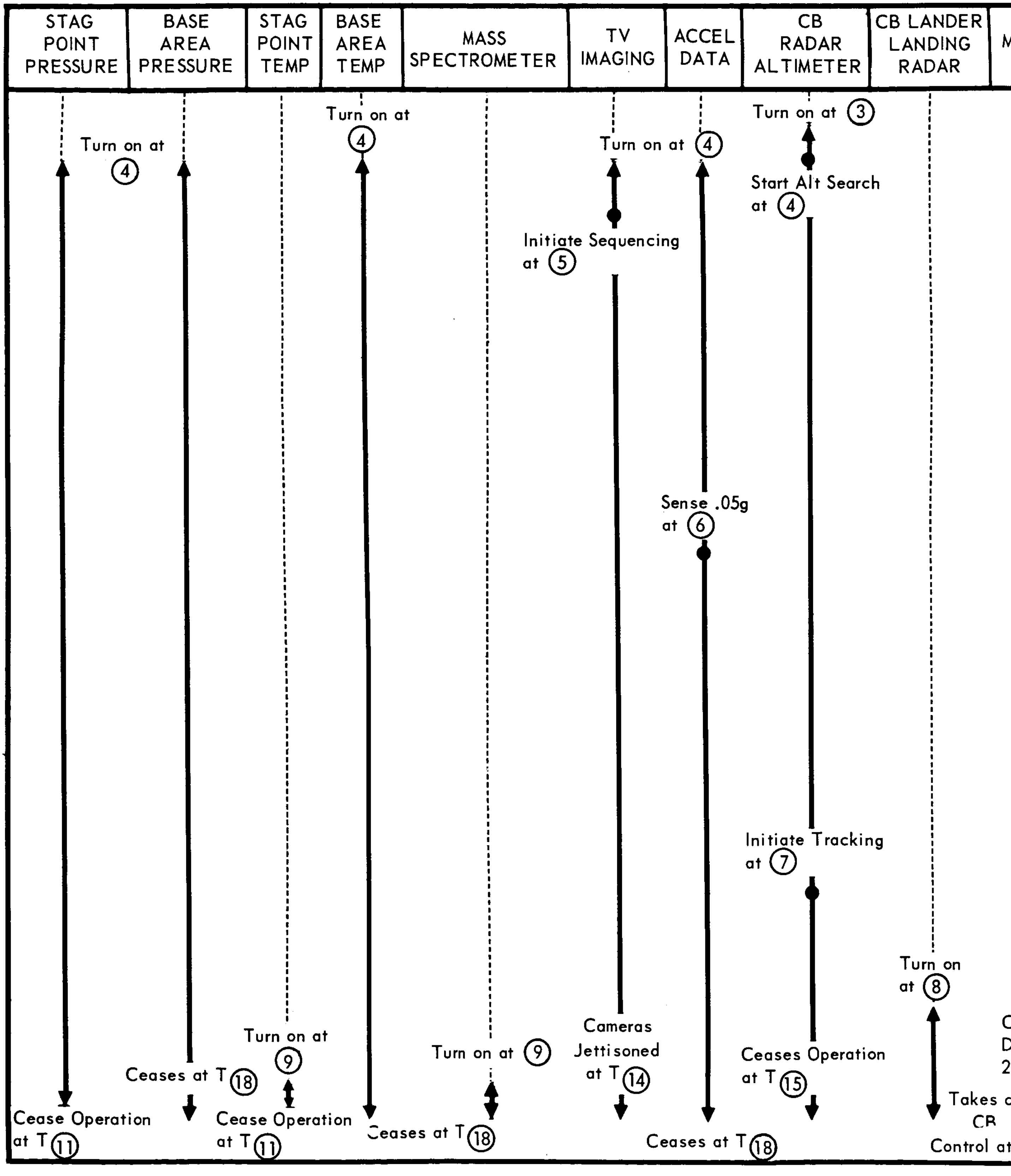
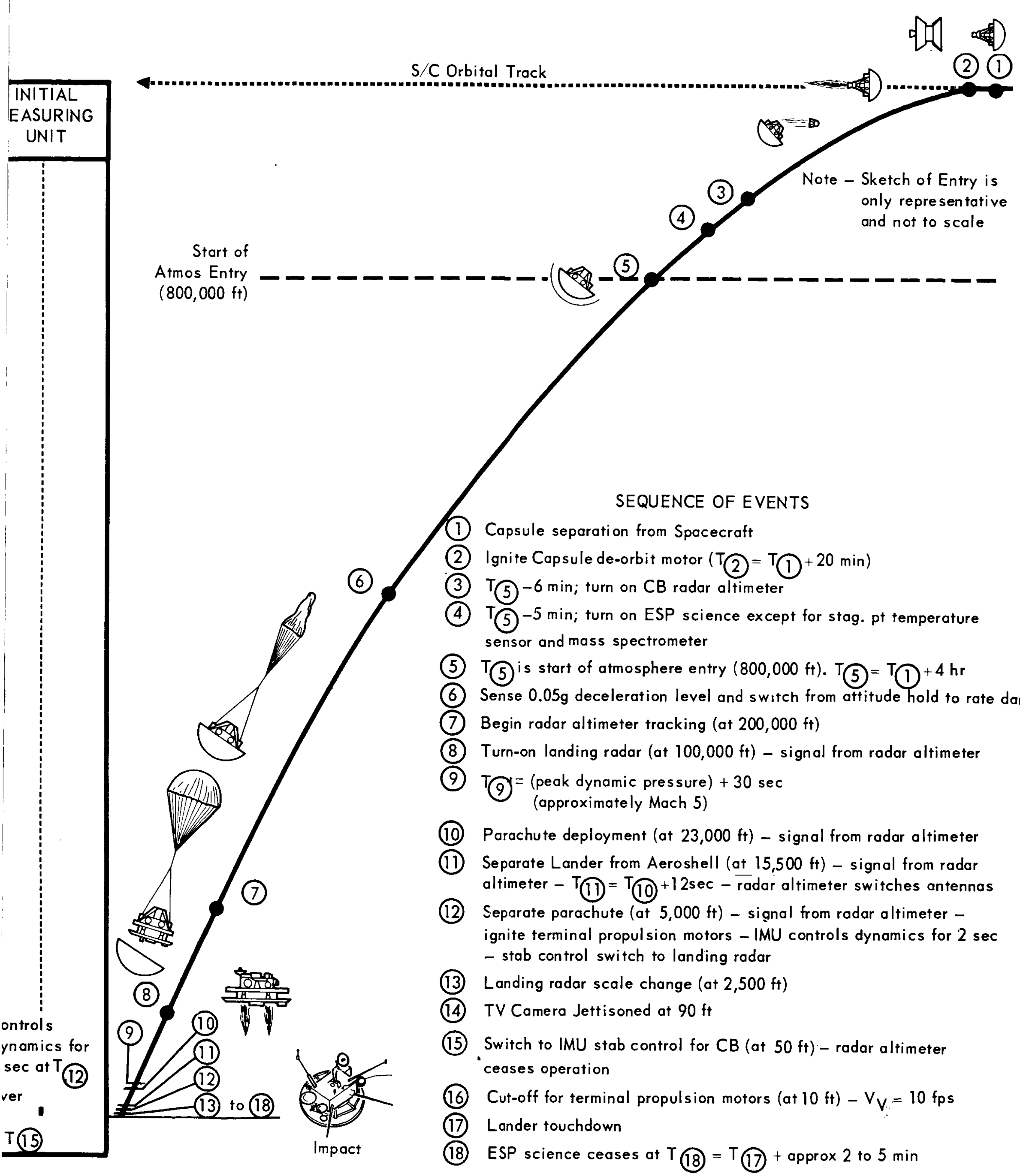


Figure A-2

2-2 -1



SEQUENCE OF EVENTS

(1) Capsule separation from Spacecraft

(2) Ignite Capsule de-orbit motor ($T_{(2)} = T_{(1)} + 20$ min)

(3) $T_{(5)} - 6$ min; turn on CB radar altimeter

(4) $T_{(5)} - 5$ min; turn on ESP science except for stag. pt temperature sensor and mass spectrometer

(5) $T_{(5)}$ is start of atmosphere entry (800,000 ft). $T_{(5)} = T_{(1)} + 4$ hr

(6) Sense 0.05g deceleration level and switch from attitude hold to rate damping

(7) Begin radar altimeter tracking (at 200,000 ft)

(8) Turn-on landing radar (at 100,000 ft) – signal from radar altimeter

(9) $T_{(9)}$ = (peak dynamic pressure) + 30 sec (approximately Mach 5)

(10) Parachute deployment (at 23,000 ft) – signal from radar altimeter

(11) Separate Lander from Aeroshell (at 15,500 ft) – signal from radar altimeter – $T_{(11)} = T_{(10)} + 12$sec – radar altimeter switches antennas

(12) Separate parachute (at 5,000 ft) – signal from radar altimeter – ignite terminal propulsion motors – IMU controls dynamics for 2 sec – stab control switch to landing radar

(13) Landing radar scale change (at 2,500 ft)

(14) TV Camera Jettisoned at 90 ft

(15) Switch to IMU stab control for CB (at 50 ft) – radar altimeter ceases operation

(16) Cut-off for terminal propulsion motors (at 10 ft) – $V_V = 10$ fps

(17) Lander touchdown

(18) ESP science ceases at $T_{(18)} = T_{(17)}$ + approx 2 to 5 min

Stagnation point pressure and temperature transducers are separated with the aeroshell at approximately 15,500 to 18,000 feet. The base region temperature and pressure, in conjunction with composition data, will continue to be transmitted until about 2 to 5 minutes after Lander touchdown.

SECTION 3

CONFIGURATION AND CAPSULE BUS/ESP ACCOMMODATIONS

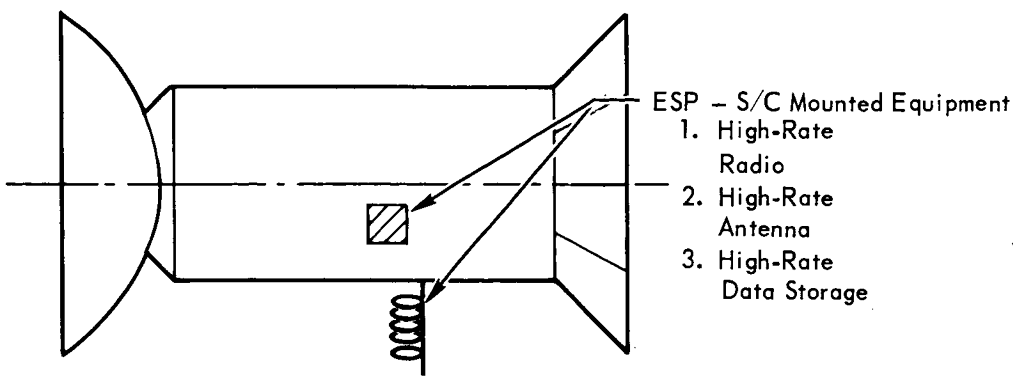
The recommended arrangement of the Entry Science Package Equipment within the Capsule is shown schemmatically in Figure A-3 and by layout in Figure A-4. The weight breakdown is as indicated in Figure A-5. The principal equipment unit, incorporating the base region pressure and temperature sensors, the mass spectrometer, and all supporting subsystem equipment including the UHF spacecraft relay link antenna is located on the Capsule Bus Lander to the side of the Surface Laboratory.

The pair of TV cameras are attached to the landing pad for simple mounting. Viewing is through the single fused silica window located in the conical section of the aeroshell just aft of the spherical nose cap. This cap has ceramic heat protection material. The TV cameras are ejected prior to touchdown with a pyrotechnic cartridge aimed through their CG and at an approximately 30 degree angle with the capsule roll axis.

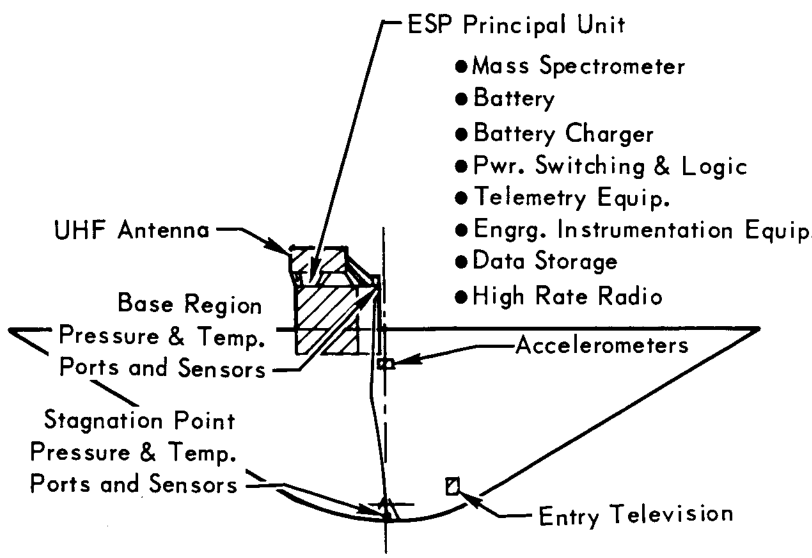
The stagnation point pressure and total temperature transducers are mounted inside the cone of the radar disc-cone antenna, behind a beryllium heat sink plug which also provides for assembly and access during systems test. The outlet bleed line from the stagnation point temperature sensor also provides for the flow from which a sample is introduced, through a molecular leak, into the mass spectrometer. The stagnation point pressure transducer is thermally protected by insulation from the approximately 400°F temperature at the inside face of the beryllium nose plug and by heat sink in the pressure line. The platinum wire in the stagnation temperature sensor is maintained below stagnation temperature during peak heating by keeping its outlet vent line closed above approximately Mach 5.

The mutual accommodation of the Entry Science Package and the Capsule Bus is summarized in Figure A-6.

LOCATION OF ESP IN CAPSULE BUS & SPACECRAFT BUS



SPACECRAFT BUS



CAPSULE BUS

Figure A–3

ENTRY SCIENCE PACKAGE INSTALLATION

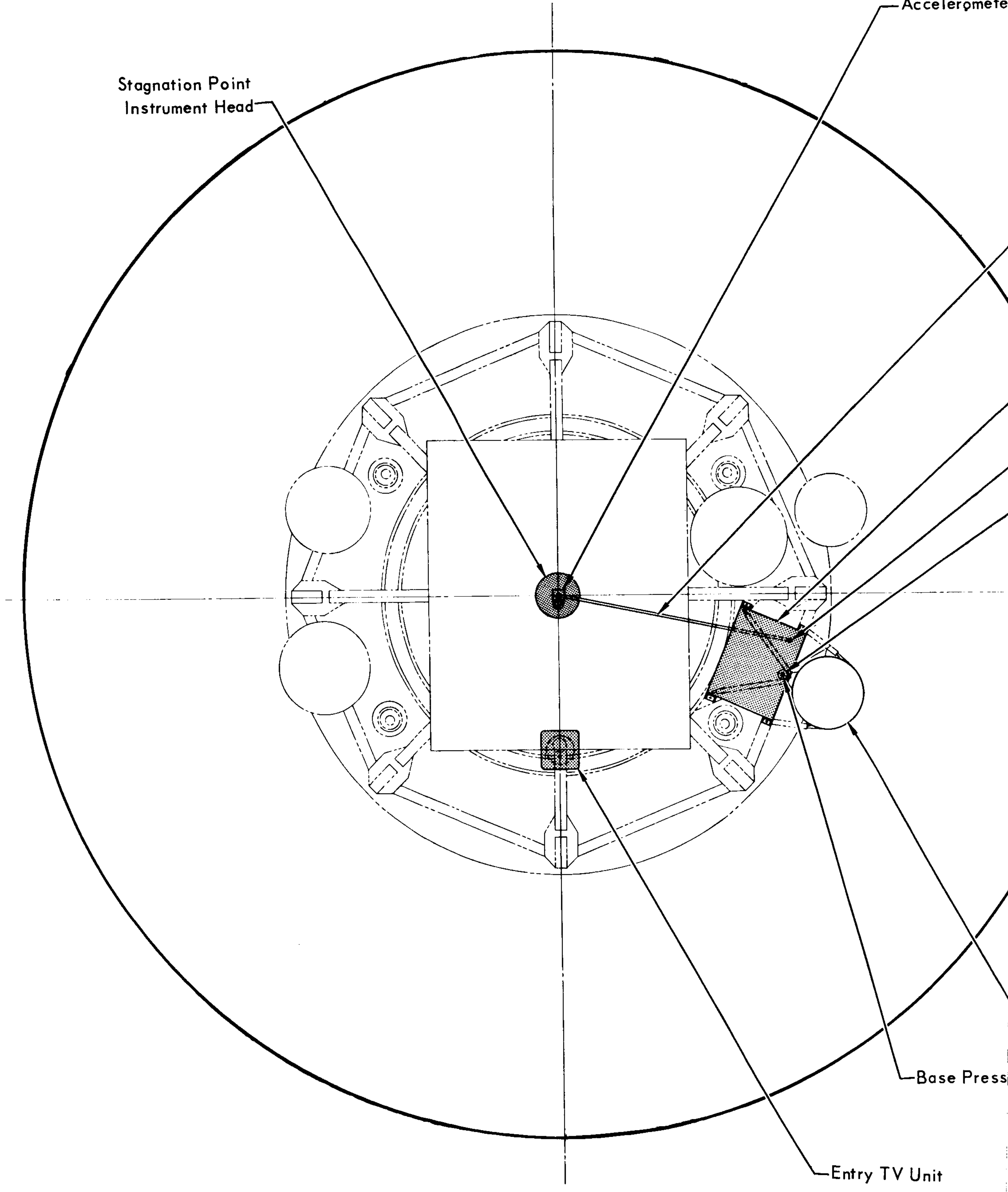
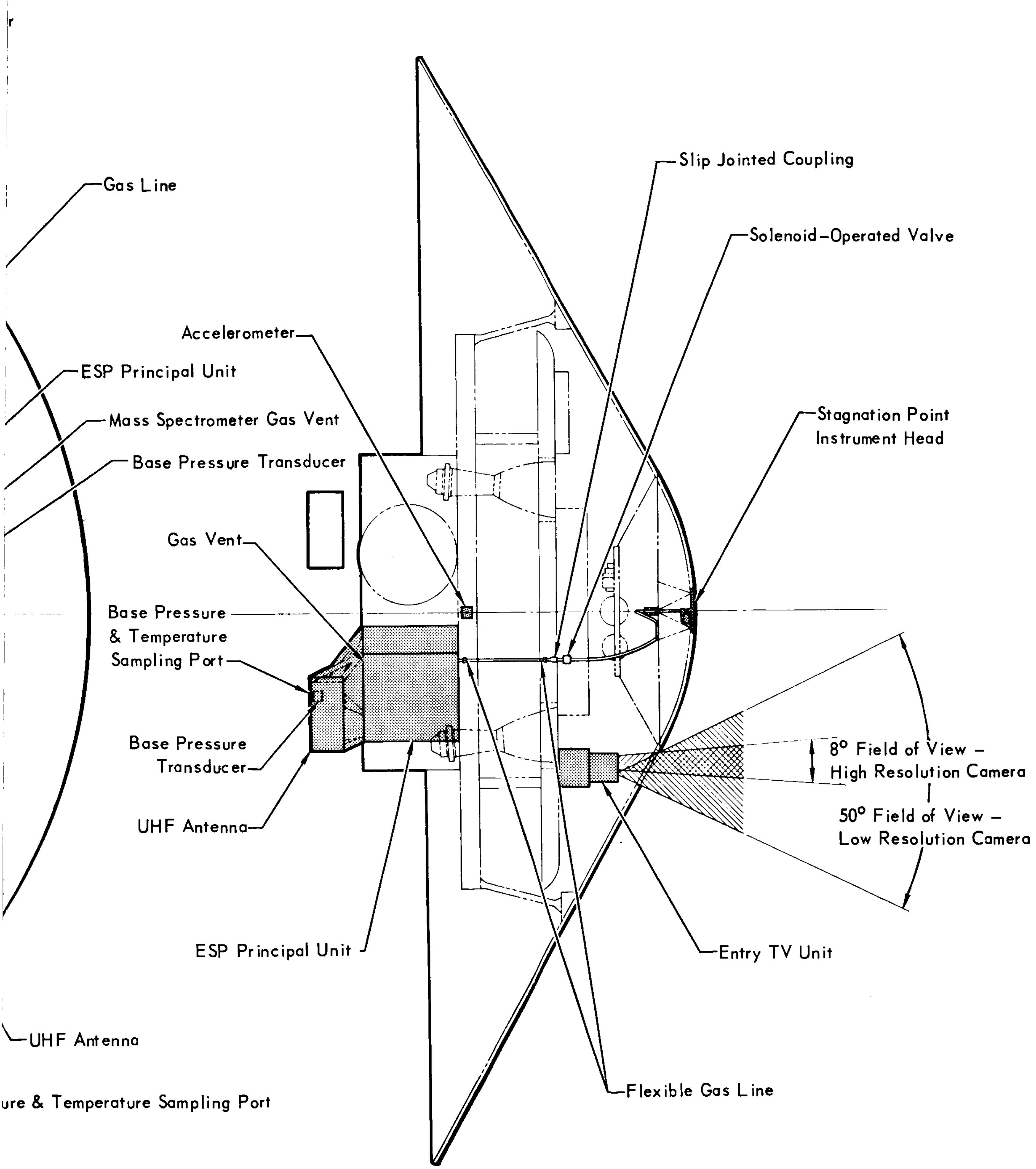


Figure A-4

3-3-1


r
Gas Line
Slip Jointed Coupling
Solenoid-Operated Valve
Accelerometer
ESP Principal Unit
Mass Spectrometer Gas Vent
Base Pressure Transducer
Stagnation Point
Instrument Head
Gas Vent
Base Pressure
& Temperature
Sampling Port
Base Pressure
Transducer
UHF Antenna
8° Field of View –
High Resolution Camera
50° Field of View –
Low Resolution Camera
ESP Principal Unit
Entry TV Unit
UHF Antenna
Flexible Gas Line
ure & Temperature Sampling Port

ESP WEIGHT STATEMENT

Capsule Bus Equipment*			180.6 lb
Experiment S/S		27.0 lb	
Imaging (2 Cameras)	14.0		
Mass Spectrometer	8.0		
Temperature Sensors (2)	1.0		
Pressure Sensors (2)	2.0		
Tri-axis Accelerometer	2.0		
Telecommunication S/S		55.0 lb	
Telemetry Equipment	9.0		
Instrumentation Equipment	6.5		
Radio S/S	26.0		
Data Storage S/S	7.5		
UHF Antenna	6.0		
Power S/S		22.5 lb	
Battery	15.0		
Battery Charger	1.5		
Power Switching & Logic	6.0		
Structure (ESP Principal Unit)		14.3 lb	
Thermal Control		5.0 lb	
Mounting Provisions (Shelf & Bracketry)		19.9 lb	
Wiring & Connectors		36.9 lb	
Spacecraft Bus Equipment			19.3 lb
Telecommunication S/S (High Data Rate)		19.3 lb	
UHF Antenna	1.0		
Radio S/S	9.0		
Data Storage S/S	8.0		
Directional Coupler	0.8		
Attenuator	0.5		
Total ESP System (CB and SC)			199.9 lb

* ESP – associated equipment included with CB Wt. Statement:
Stag. Probe Transducer Insulation 0.4 lb
ESP Test Antenna (on CB Adapter) 7.0 lb

Figure A-5

CAPSULE BUS AND ENTRY SCIENCE PACKAGE MUTUAL ACCOMMODATION

	CAPSULE BUS DESIGN OR OPERATIONAL FEATURE	SELECTED ESP ACCOMMODATION OR EFFECT	ADDITIONAL CHARACTERISTIC OF FEATURE/ACCOMMODATION
Imaging	• No roll attitude control • Non-ablative nose cap* • Four Terminal Descent Engines • Attitude Rate Damping*	• Image camera optically aligned with roll axis • Single fuzed silica window adjacent to non-ablative spherical nose cap • Camera location to be maximum possible distance from engines compatible with above window location. • Improvement in image resolution.	• No roll attitude reference required. • Good viewing with minimum programming of images • Minimum optical disturbance by exhaust plume during terminal descent.
Pressure, Temperature and Composition	• Beryllium Nose Plug at stagnation point in non-ablative nose cap* • Attitude Rate Damping* • Base region thermal curtain with provision for protruding sensors*	• Stagnation point instrument head, parts flush with Aeroshell mold line. • Increased accuracy of stagnation point measurements with low α • Uncomplicated mathematical relationship of stagnation measured data to free stream data • Base region pressure and temperature sensors mounted to principal unit structural unit.	• Composition Sample contamination avoided. • Beryllium nose plug acts as a heat sink for the temperature measurement plus provide structural support for the instrument head. • System Test access to instruments.
Acceleration	• Parachute mortar along roll axis • Attitude rate damping*	• Mount accelerometers on mortar base structural fitting as close to center of gravity as physically possible. • Reduces effect of uncertainty in acceleration corrections and $C_L(\alpha)$	• Minimize acceleration errors due to center of gravity location.
ESP Subsystems	• View looking aft unobstructed* • Thermal curtain design for capsule bus must be RF transparent* • Jettisoning of de-orbit motor substructure to provide antenna view angle for ESP.	• Antenna located with principal unit. • Principal unit contains all ESP subsystems	• Minimize Capsule Bus/ESP structural interface.
Other Trajectory/ Atmosphere Reconstruction Aids	• High altitude altimeter* • Rate gyros* • De-Orbit monitoring*	• Extends capability to 200,000 ft • Monitor during entry for $(\theta-\gamma)$ time history	

*Indentifies items influenced by ESP requirements.

Figure A-6

SECTION 4

CHARACTERISTICS OF ENTRY SCIENCE INSTRUMENTS

The general characteristics of the Entry Science instruments indicated in the Reference A-1 constraints document have been retained for the selected design concept. Their characteristics as further developed for purposes of operational and support requirements are summarized in Figure A-7.

The measurement of density and temperature as a function of altitude using these instruments is indirect for two major reasons. First, the quantities directly measured are functions of the capsule characteristics and behavior as well as of the atmospheric and trajectory quantities of interest. Second, the accuracy with which the trajectory and the atmospheric property profiles may be reconstructed is greatly enhanced by appropriately reflecting the data taken throughout the trajectory to each computation point along the trajectory. A pilot computer program, using maximum likelihood techniques, has been developed and operated for this latter purpose. Illustrative simulation analysis has been performed.

ENTRY SCIENCE INSTRUMENTS FOR SELECTED DESIGN

INSTRUMENT	FUNCTION	DESIGN REQUIREMENTS				TEMPERATURE TOLERANCE		OPERATIONAL SEQUENCE	ELECTRICAL POWER		
		Weight (Lb)	Size (In.)	Volume ($In.^3$)	Viewing Deployment or Access Requirements	Operating	Non Operating		Ave Power (Watts)	Energy (Watt Hr)	Para
Pressure Transducer	Obtain stagnation pressure measurements	1.0	2D x 2	6.3	Access to atmosphere at nominal stagnation point	$-200°$ to $125°C$	$-200°$ to $135°C$	Turned on 300 sec prior to h_E = 800K ft. Separates with Aeroshell	1.4	.35	Pres 0 to 3
Temperature Transducer	Obtain stagnation temperature measurements	0.5	1D x 1.9, or Less	1.7	Access to atmosphere at nominal stagnation point	$150°$ to $1200°K$	$100°$ to $1500°K$	Turned on 300 sec prior to h_E = 800K ft. Samples from Mach 5 to Aeroshell separation.	0.01	.0025	Temp ture to 12
Pressure Transducer	Obtain base region pressure measurements	1.0	2D x 2	6.3	Access to atmosphere at base of capsule	$-200°$ to $125°C$	$-200°$ to $135°C$	Turned on 300 sec prior to h_E = 800K ft. Operates until ESP power down 120 sec after touchdown.	1.4	.42	Pres 0 to (psia
Temperature Transducer	Obtain base region temperature measurements	0.5	1D x 1.9	1.7	Access to atmosphere at base of capsule	$150°$ to $330°K$	$100°$ to $1500°K$	Turned on 300 sec prior to h_E = 800K ft. Operates until ESP power down 120 sec after touchdown.	0.01	.003	Temp ture to 33
Accelerometer	Obtain tri-axial acceleration measurements during the entry	2.0	2.75 x 1.75 x 2.0	9.6	Alignment parallel to CB axis; required location ahead of and as near the c.g. as possible.	$40°$ to $93°C$	$10°$ to $135°C$	Turned on 300 sec prior to h_E = 800K ft. Operates until ESP power down 120 sec after touchdown.	4.0	1.2	(3) A eratic 0 to 3 0 to 5 $\pm 2g$ Roll
Mass Spectrometer	Determine atmospheric composition	8.0	2 x 7 x 14	200	Atmosphere access via both stagnation and base region	$-10°$ to $65°C$		Turned on 300 sec prior to h_E = 800K ft. One spectrum every 10 seconds terminates with surface operations 120 sec after touchdown	7.0	2.1	Spect Monit ing
Vidicon Cameras	Obtain landing area images during the entry	14.0	8D x 12	700	Optical axes of both imagers aligned with CB entry configuration roll axis $\pm 2°$ viewing access through Aeroshell to accommodate both $8°$ and $50°$ fields-of-view	$-5°$ to $+40°C$	$-20°$ to $+60°C$	Turned on 300 sec prior to h_E = 800K ft. Operates until ejection at 90 ft. Each camera takes one image every 10 sec. One image is taken every 5 sec by alternating the cameras.	20	≈ 5.4	Vide Statu Monit ing

Figure A-7

	REQUIREMENTS ON DATA SUBSYSTEM				REMARKS
ieter	Type (1)	Sample Rate	Word Length (Bits)	Total Data Calculation	
ure psia	HL	1 sps	8	8 bits/sample x 1090 samples = 8720 bits	Most useful data obtained from P_s = .015 psia until Aeroshell separates
era- 50° 00°K	LL	1 sps	8	8 bits/sample x 1090 samples = 8720 bits	Most useful data obtained from peak stagnation pressure + 30 sec until Aeroshell separation
sure .5	HL	1 sps	8	8 bits/sample x 1090 samples = 8720 bits	
era- 50°)°K	LL	1 sps	8	8 bits/sample x 1090 samples = 8720 bits	
cel- g or and	D	20bps each	10	10 bits/sample x 2180 samples 21,800 bits each axis, 65,400 total	Most useful data from threshold (.005g) until touchdown
ange	BL	2 sps	1	1 bit/sample x 2180 samples = 2180 bits	
um	D	50 bps	8	50 bits/second x 8/10 readout fraction x1090 sec = 43,600 bits(400 bits/sample	Most useful data from peak stagnation pressure plus 30 sec until ESP surface operations termination.
r- (5)	HL	.1 sps	7	5 channels x 7 bits/sample x 109 samples = 3815 bits	
(1)	BL	.1 sps	1	1 channel x 1 bit/sample x 109 samples = 109 bits	
	D	50Kbps	6	Sample rate x word length x time x number of channels 8000 x 6 x 670 x 1 = 32.16 (10^6) bits 8000 x 6 x 300 x 1 = 14.4 (10^6) bits* (* During 300 sec warmup)	Most useful data from 500Kft to 90 ft for 8° field-of-view imager and from 150Kft to 90 ft for 50° field-of-view imager
(2)	D	.2 sps	6	.2 x 6 x 970 x 2 = 2328 bits	
(1)	D	.2 sps	2	.2 x 2 x 970 x 1 = 388 bits	
(1)	BL	.2 sps	1	.2 x 1 x 970 x 1 = 194 bits	Slow scan 200 x 200 element vidicon, .44 x .44 in. format
r-(5)	LL	.1 sps	7	.1 x 7 x 970 x 5 = 3395 bits	
(2)	HL	.1 sps	7	.1 x 7 x 970 x 2 = 1358 bits	
(2)	BL	1 sps	1	1 x 1 x 970 x 2 = 1940 bits	
(1)	HL	.2 sps	7	.2 x 7 x 970 x 1 = 1358 bits	

SECTION 5

PRIMARY ESP SUPPORT EQUIPMENT

The three most important ESP support subsystems are:

a. Telecommunications

b. Electrical Power

c. Thermal Control

5.1 TELECOMMUNICATIONS - The ESP telecommunications is composed of the telemetry, radio, antenna, and data storage subsystems.

Since the ESP telecommunications utilizes a relay link via the spacecraft, these subsystems are divided between the Capsule Bus and the spacecraft

The telemetry subsystem receives all the scientific and engineering data generated by the ESP scientific instruments and provides additional sensor and signal processing of the engineering data and all of the required multiplexing, formatting, and timing functions.

The radio subsystem located in the ESP/Capsule Bus modulates and transmits while that portion in the ESP/spacecraft receives, demodulates, and reconstructs the PCM data stream.

The antenna subsystem consists of a radiating antenna on the ESP/Capsule Bus and a receiving antenna on the ESP/spacecraft.

The data storage subsystem on the ESP/Capsule Bus stores for later transmission the data (except television) collected during blackout of radio transmission. In the spacecraft, the received ESP data is stored for later transmission to Earth at a rate compatible with the spacecraft telecommunications capacity.

A functional block diagram and the performance characteristics of the ESP telecommunications system are shown in Figure A-8. The stored program control of the ESP telemetry equipment provides flexibility to change in scientific payload data requirements, while a core memory logic is used instead of a hard-wired transistor-diode logic due to the core memory's greater reliability. In order to provide greater reliability with regard to possible equipment failure and multipath fading, transmission of all ESP data, except television, is provided with an alternate path via the Capsule Bus. All data (except television) accumulated during blackout is stored in a core memory and read out 50 and 150 seconds later. Thus, each bit is transmitted up to three times, providing a greater performance margin following blackout and under multipath conditions.

BLOCK DIAGRAM AND PERFORMANCE CHARACTERISTICS OF ESP TELECOMMUNICATIONS

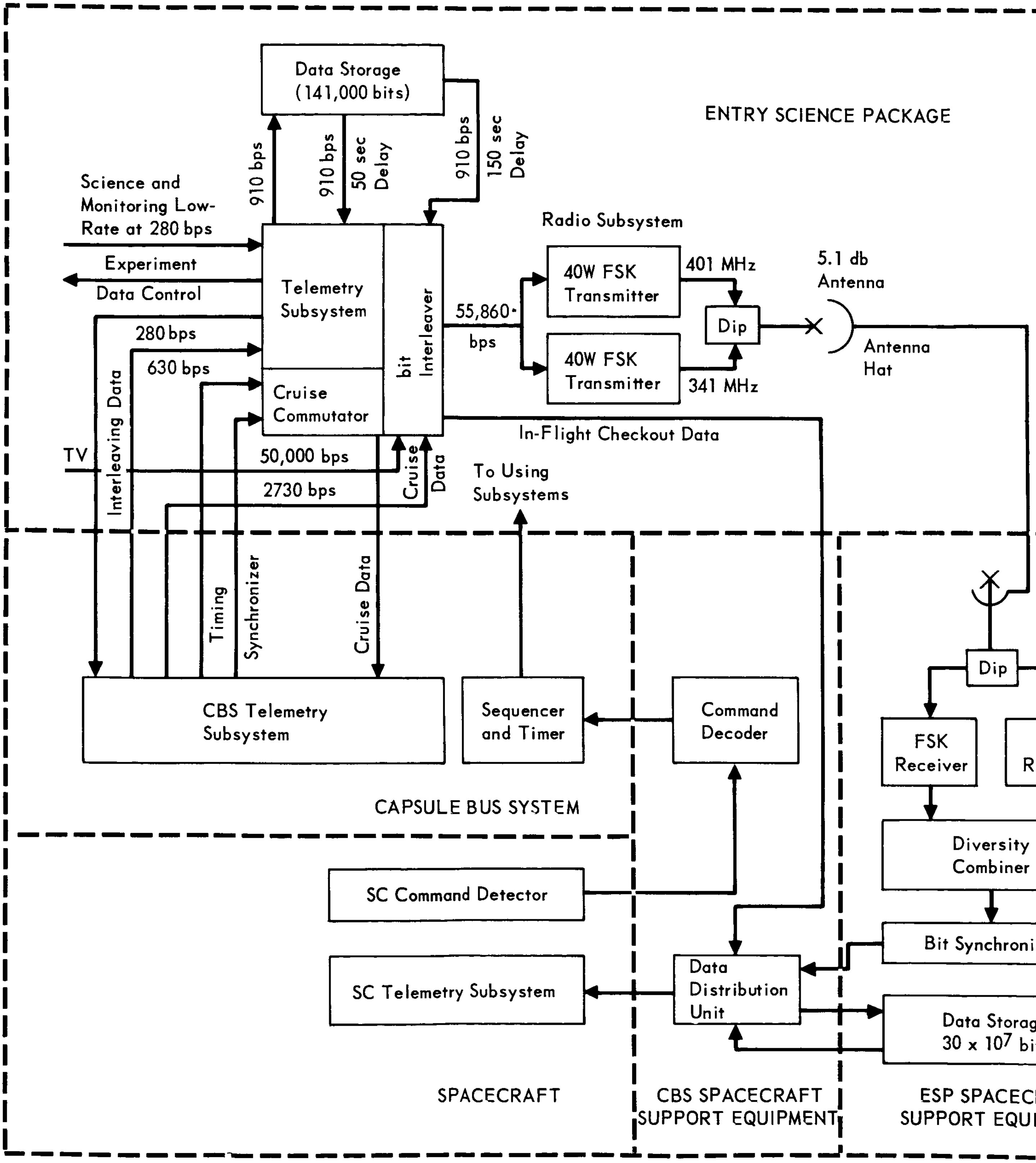
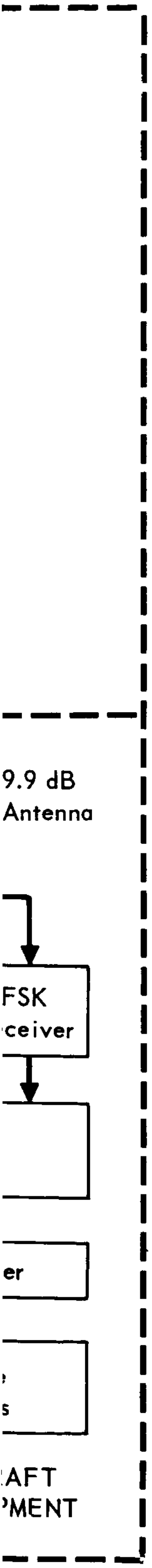


Figure A-8

5-2 -1



INPUT DATA RATE:

Low Rate Science and Engineering:	208 bps
High Rate Science (TV)	50,000 bps

OUTPUT DATA RATE: 55,860 bps

DATA STORAGE: 50 sec and 150 sec delay storage to provide low rate science and engineering data accumulated during blackout.

RADIO LINK:

Modulation:	FSK with split-phase coding
Frequency Diversity:	
Carrier Frequency:	341 MHz and 401 MHz
Transmitter Power:	40W Each

ANTENNAS:

Transmitting: 5.1 db cavity-backed spiral; 95° beam
Receiving: 9.9 db single helix; 55° beam

TELEMETERY PROGRAMMER:

Reprogrammable by command prior to separation from spacecraft

REDUNDANT PATH ALTERNATIVES:

Low Rate Science and Engineering: Through CBS link
High Rate Science (TV): Available via single link of dual radio link with less multipath margin

The resulting data stream of 55,860 bits per second is transmitted by two solid-state 40 watt FSK transmitters simultaneously operating at frequencies of 341 MHz and 401 MHz. The transmitter outputs are combined by a diplexer and radiated by a single-element cavity locked spiral antenna with a maximum gain of 5.1 dB and a total half power beamwidth of 95°. The spacecraft antenna is a single axial-mode helix mounted on a mast with a maximum gain of 9.9 dB and total half-power beamwidth of 55 degrees. A dual diversity FSK receiver with square-law detection is utilized to give improved performance under multipath conditions. Either transmitter-receiver pair operating alone will provide adequate performance except for severe multipath interference.

Split-phase (Manchester) coding is used to provide a strong signal component at the bit frequency, to minimize effects of differential gain variation between the FSK receiver mark - space channels, and to provide an additional performance improvement with multipath interference. A unique bit synchronizer employing a time gate locked about zero crossings occurring at the data rate is used to eliminate synchronization ambiguity and to help discriminate against multipath.

A tape recorder with a 30-40 million bit capacity is used to store the data in the spacecraft until transmission to Earth can be scheduled. The playback rate will be compatible with the spacecraft-to Earth link capacity, probably 0.25 to 0.10 times the recording rate of 55,860 bps.

5.2 ELECTRICAL POWER - The electrical power subsystem provides power for inflight monitoring during cruise periods when Flight Spacecraft power is not available to the Flight Capsule, and power for equipment operation from preseparation to shutdown of the Entry Science Package after landing. The subsystem also distributes power, either from the spacecraft or internal battery power, to the ESP equipment. The subsystem contains a battery, battery charger, and a Power Switching and Logic Unit. The battery is an 8.5 amp hour, sealed silver-zinc battery, designed for a high discharge rate. The battery charger is a two step float charger.

The Power Switching and Logic Unit contains the power distribution and control switching for selecting either external power from the CB or internal power from the ESP battery. It also contains on-off switches (latching relays) for turning on and off the ESP equipment, and power source fault protection devices.

System redundancy is obtained by utilizing, as a back-up to the ESP battery, the power available in the Surface Laboratory. For VM-models, 1, 3, 5, 7 and 9 (surface temperature = 495°R) the SL battery has enough capacity to supply the ESP power for its entire entry mission and still maintain a sufficient margin to permit the SL to perform its entire diurnal cycle surface life profile.

5.3 THERMAL CONTROL - The ESP thermal control subsystem maintains equipment temperature levels within their allowable ranges throughout all mission phases. Temperature control is provided for both the ESP science instruments and any Capsule Bus-mounted ESP support equipment. Until Lander separation from the Aeroshell, the subsystem operates within the overall temperature environment provided by the Capsule Bus thermal control subsystem and will maintain the ESP equipment at between 50° to 125° throughout the mission. The major elements of the system include electrical heaters, insulation, and thermal control surfaces.

The heaters provide 2 and 3 watts of power to the ESP Principal Unit and to the descent TV, respectively. These heaters operate continuously when power is available from the spacecraft. The fiberglass insulation minimizes the required heater power, prevents excessive cool-down of equipment if power interruptions occur during spacecraft midcourse corrections, and avoids overheating during the brief Martian entry period. In addition to the insulation, low emittance surface finishes are used on the ESP packages to assist in cruise heat retention and reduce the heat input during entry.

SECTION 6

RATIONALE FOR SELECTED DESIGN CONCEPT

In the process of deriving the selected ESP concept, various alternatives were considered in such critical decision areas as:

a. Selection of science instruments

b. Single ESP unit versus physically separated components

c. Sequencing mode for entry TV

Since the decision reached in these three areas significantly affected the synthesis of the ESP system, its Capsule Bus and spacecraft interfaces, and its mode of operation, a discussion of the alternatives considered and the rationale that led to our selected approach is believed warranted.

6.1 SELECTION OF SCIENCE INSTRUMENTS - The science instruments specified in Reference A-1 as typical of the capability to be provided are of high priority and have been utilized for the purposes indicated without major exception. Several additional instruments appear desirable for breadth and accuracy of data but none are necessarily of higher priority than those specified. It was considered highly desirable to make pressure and temperature measurements at the capsule stagnation point and the base region. The stagnation point offers a location which has simpler flow field relationships for interpretation of data for use in post-flight trajectory/atmospheric reconstruction. At Aeroshell separation, however, the stagnation point measuring source is lost, and therefore the use of base region sensors is also desirable for some continuity of data throughout the trajectory to the post-touchdown sampling period on the surface of Mars.

The decision has been made, then, to employ in our selected ESP design the instrument group specified in the JPL Constraints Document. The detailed characteristics of these selected instruments are presented in Figure A-7. However, consideration has been given to other instruments felt to be of high priority for: (a) increased science payload weight or scientific objectives of the 1973 mission ESP system and/or (b) for supplemental data from future missions. Priority addition science measurements/instrumentation include the following:

a. Measurement of γ backscatter from outside the shock wave for direct determination of the density of the ambient atmosphere.

b. Measurement of solar UV and X-ray radiation absorption by the atmosphere above the descending capsule for supplemental density, composition, and aerosol data.

c. Mass spectrometer measurement of composition above the region of aerodynamic heating (but in the continuum flow regime) and also in the region below peak heating. During the remaining high temperature portion of the trajectory, theoretically interpretable data may also be obtained at essentially no cost.

d. Measurement of differential pressure between the nominal stagnation point and a point on the side of the spherical nose section, for supplemental determination of dynamic pressure independent of the lift and drag coefficients for the capsule and the capsule/deployed parachute combination.

e. Post-touchdown imaging by means of a facsimile camera, erected from the principal ESP unit after touchdown, utilizing the UHF spacecraft relay link for the few minutes available immediately after touchdown for relay data.

Figure A-9 illustrates an ESP design concept which incorporates these priority addition measurements. As shown, these additions could be made with only a slight modification to the "preferred approach" ESP design presented in Figure A-4. The further investigation of these experiments, and their effect on the ESP subsystems and Capsule Bus design, should be conducted as a part of any "growth studies" of the VOYAGER Capsule Bus System.

6.2 SINGLE ESP UNIT VERSUS PHYSICALLY SEPARATED COMPONENTS - The JPL Constraints Document notes the desirability of simplifying the ESP/CB interface to the point where the "CB-ESP physical interface shall consist of a structural field joint and an electrical connector." During our design activity it soon became evident that while a "single ESP unit" approach was generally feasible, it would tend to compromise several of the requirements associated with the science instruments. For example:

Instrument	Requirement (Source)
Accelerometer	Locate "ahead of and near the c.g. of entry capsule" (JPL)
Pressure Transducers	Locate transducers for stagnation point and base region measurements, and near the ports in order to minimize line loss and response lag (McDonnell)
Temperature Probes	Locate Platinum resistance thermometers for stagnation point and base region measurements, and as close to flow field reference point as practicable to prevent any temperature reduction or response lag (McDonnell)
Entry TV	"Camera optical axis must be parallel with roll axis" (JPL)

In addition, a stagnation region source for the mass spectrometer during descent is desired for easier avoidance of contamination, and to permit short time constant sampling.

PROVISIONS FOR ADDITIONAL ESP MEASUREMENTS

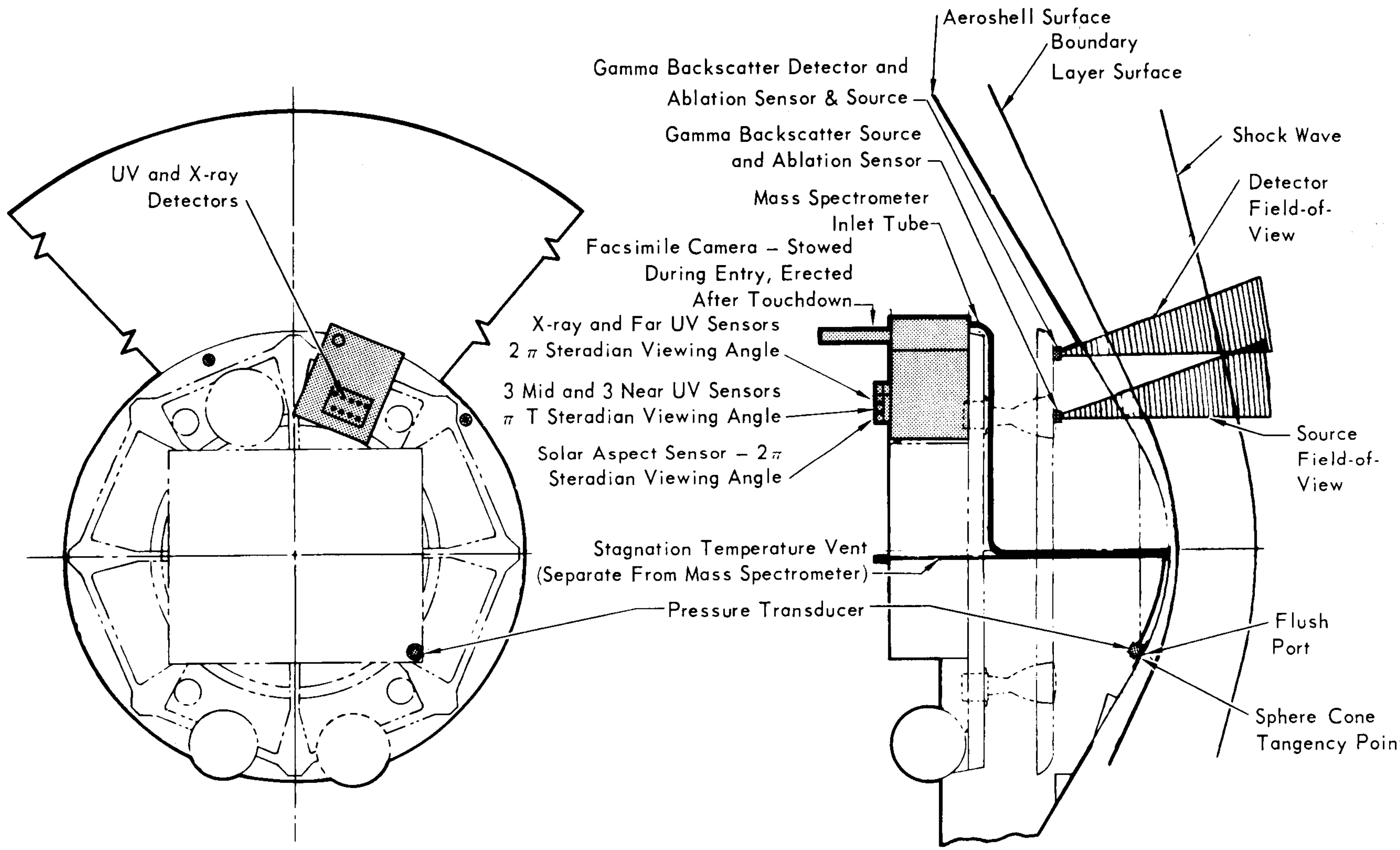


Figure A-9

Considering these requirements, coupled with those for the location of the ESP UHF antenna and for TV window location for simple design and good results, the ESP "preferred approach" that was selected resulted in one primary unit and three smaller packages as indicated below:

	Description	Lateral Location (Ref to CB Roll Axis)	Longitudinal Location (Ref. to CB Nose)
a.	Stagnation point pressure transducer and temperature probe	On roll axis	On CB nose
b.	Dual camera package	℄ of package is parallel to and 32" from roll axis	Center of optical window is 10" aft of nose
c.	Accelerometer Package	on roll axis	Approx. 49" aft of nose (approx. 3.5" ahead of CB c.g.)
d.	ESP Principal Unit (contains all other science instruments plus supporting subsystems)	℄ of unit is parallel to and 45" from roll axis	Unit is 31" in length (incl. UHF antenna) with front face 50" aft of nose

The above noted locations were selected on the basis of ESP design requirements and available physical space (i.e., due to CB and SL equipment location). It is evident that trying to physically locate all of the CB-mounted ESP equipment as one unit could result in all or some combination of the following:

a. Major constraint on the CB and SL equipment packaging.

b. Major compromises in ESP science instrumentation installation requirements.

c. A single ESP unit that would probably be excessively large, in order to even minimally comply with the required instrumentation locations, with a resultant heavier system installed than a comparable system utilizing small, independent instrument packages.

Figure A-10 illustrates some of the compromises involved with attempting to achieve a single unit ESP design. Even here, however, the accelerometer was separated from the "integrated unit" and placed on the roll axis in order to eliminate the measuring inaccuracies that would be incurred by locating the accelerometer a significant distance off the roll axis. As shown in Figure A-10, the stagnation point

SINGLE-UNIT ESP
ACCELEROMETER SEPARATE

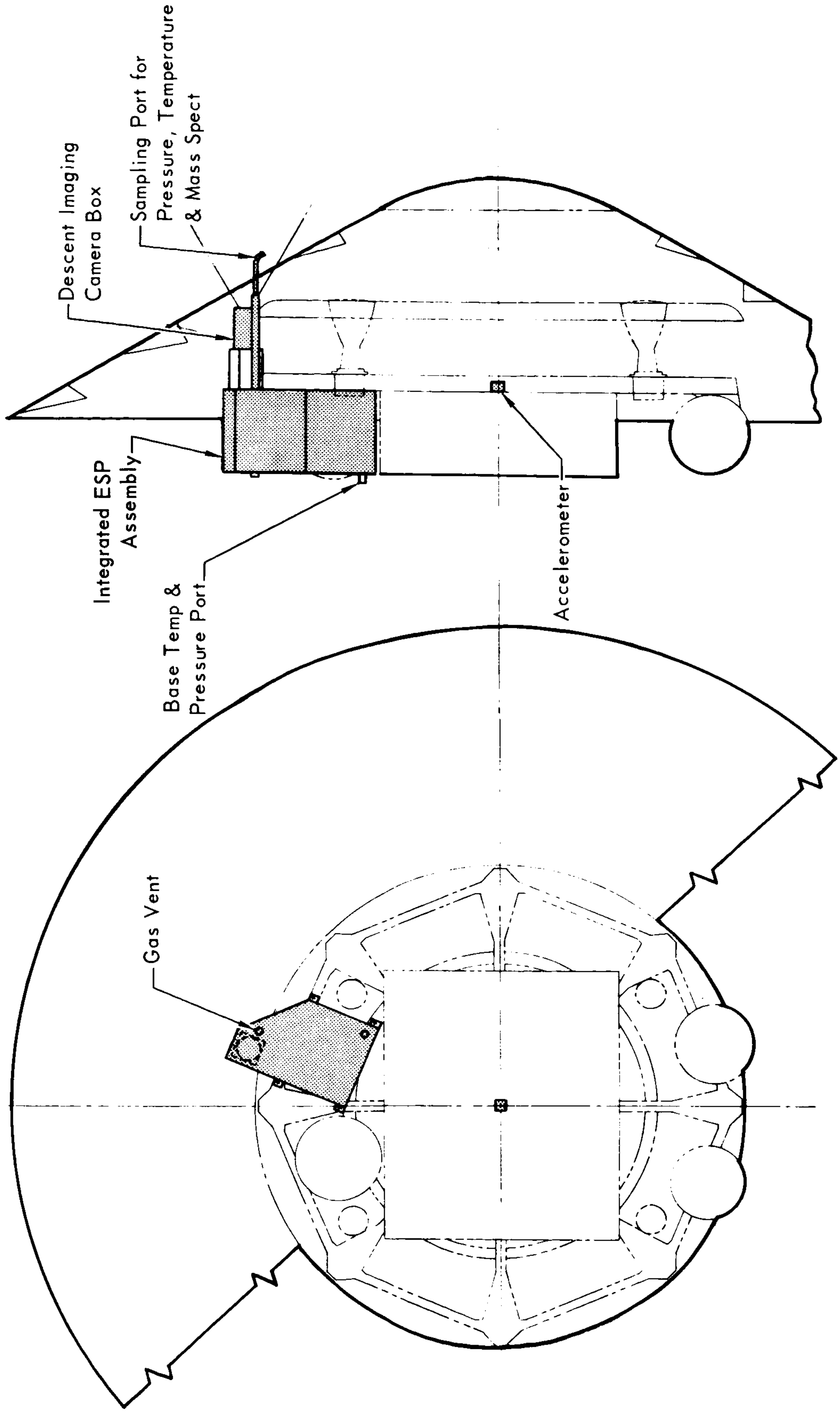


Figure A–10

location for the forebody sampling port (as used in our "preferred approach") has been sacrificed in order to obtain a reasonably sized integrated ESP design. This sacrifice would be undesirable since the stagnation point offers a sampling location having well defined flow field relationships.

On the basis, then, of the aforementioned types of considerations, our Phase B study resulted in the selection of an ESP design which accepted a somewhat more involved ESP/CB interface in order to obtain a superior science data-measuring arrangement.

6.3 SEQUENCING MODE FOR ENTRY TV - The selection of the operational sequence of the entry TV is of major importance due to its influence on the Aeroshell heat shield (i.e., condensation of ablation products on the viewing window can significantly impair its optical qualities) and on the ESP telemetry system (i.e., the entry TV requirements essentially set the telemetry transmission requirements).

In selecting the altitude range during which to obtain imaging, several alternatives are available:

a. Low altitude imaging only
b. Imaging only during periods predicted to permit good imaging and transmission (such as by utilizing a signal from the stagnation temperature probe)
c. Continuous imaging, and attempted transmission, from start of entry (800,000 feet) to near surface

Significantly interrelated with these choices is the selection of the optical window design approach. If imaging occurs only at low altitudes or during selected portions of the trajectory, a protected window can be utilized which is then uncovered after the trajectory period of high temperature/high ablation rate has been completed. Since the possible introduction of ablation products into the atmospheric samples going to the mass spectrometer would prevent the spectrometer from defining the true atmospheric composition of Mars, it has been decided to enhance the data-gathering capability of both the mass spectrometer and imaging installations by employing a ceramic nose cap on the capsule which would also encompass the TV optical window to minimize window condensation. This design decision then makes it possible to utilize the entry TV throughout the entry phase and this is our selected approach. This approach offers the opportunity to obtain limb and wide area coverage, thus increasing the probability of securing some images which will permit correlation with the orbiter TV broad-coverage pictures. In addition, the selected approach of employing uninterrupted sequencing from the start of entry represents a very simple and uncomplicated sequence control operation.

With the decision to obtain imaging throughout the 800,000 foot altitude entry phase, it is evident that both a wide coverage camera and a high resolution camera are required to adequately cover the range of optical requirements. Part E, Section 2.2 discusses the tradeoffs involved and, as indicated therein, the selected system consists of a dual camera arrangement (one camera having a 50° field of view and the other having an 8° field of view) viewing through a single optical window.

In selecting the interval between images, a choice is available between:

a. A minimum interval, which would permit the use of -

- Alternate readout of two sterilizable image storage vidicon cameras with conservative erase times.
- Stereo overlap and good area identification continuity between successive images (even on the parachute with wind).
- Reasonable continuous-transmission image bit rate (without auxiliary storage).

b. Maximum interval, which would permit the use of -

- Maximum reasonable vidicon image storage times.
- Minimum image transmission bit rates

but not of -

- Appreciable stereo overlap and area identification continuity between successive images (unless auxiliary storage is provided).

The desirability of maintaining overlap for stereoscopic viewing reconstruction as well as area-recognition continuity through the parachute descent phase, where winds can produce a significant amount of drift in a short period of time, argues for short intervals between successive images of the order of 5 seconds or less (10 seconds between successive pictures from the same camera). From the standpoint of simplicity, it would be desirable to keep the interval constant throughout the sequence. Communication requirements argue for keeping the interval large, but if we obtain 240,000 bit images from alternate cameras every 5 seconds, this would be compatible with the minimum bit rate of 50,000 bits/sec indicated for the UHF relay link in Reference A-1.

It is this latter approach which we have selected for our imaging system; namely, dual cameras alternately obtaining a 240,000 bit image every 5 seconds and transmitting the data at 50,000 bits/sec through a UHF relay link to the spacecraft.

As previously noted, the selected sequencing approach is to start at entry and continue to near touchdown. This necessitates operation through the "communication blackout" regime. The final major decision with ragard to the entry TV system, then, is concerned with whether to accept loss of the images transmitted during blackout or to record these images onboard the capsule and then transmit them to the spacecraft later in the entry phase.

The duration of the plasma blackout, and the altitudes at which it occurs, is similarly a function of the entry angle, entry velocity, and the atmosphere encountered (VM-1 through VM-10). Secondary effects result from the by-products of the ablative heat shield and the ballistic coefficient of the entry vehicle . Our studies were conducted with a range of entry velocities between 15,000 fps to 13,000 fps, and a range of entry angles between -20° to -10.9°. The results have shown that blackout can start as high as 462,000 feet altitude (VM-9) and terminate as low as 59,000 feet altitude (VM-8). The duration of the blackout can last almost 150 seconds (VM-1, $V_e = 13{,}000$ fps, $\gamma_e = -10.9°$).

The duration of the blackout, and the data rate of the imaging experiment determine the data storage requirements. At 50,000 bps, the storage device must have a capacity of 7.5×10^6 bits. Even if the data rate is reduced, only magnetic tape recording devices appear practical. Of the possible tape recording techniques, an endless loop tape recorder would seem to be the only feasible approach.

An endless loop tape recorder could be sized to provide 150 seconds of storage before the loop is repeated; its weight would be about 4 to 6 pounds. An endless loop recorder would allow re-transmission of the stored imaging data simultaneously with real time transmission. To support the increased data rate, the power of each transmitter must be increased from 40 watts (compatible with 50,000 bps) to 80 watts RF if the real time imaging data rate remains at 50,000 bps. The total increase in ESP weight to support this approach would be approximately 40 pounds.

An alternate approach would be to reduce the data rate of the imaging experiment to about 25,000 bps. This would allow a 50,000 bps ESP radio link to be retained (25,000 bps real time plus 25,000 bps delayed time). The total increase in ESP weight would be 5 to 7 pounds, resulting primarily from the tape recorder. Unfortunately, this approach would require a 10 second interval time between successive images (20 seconds between successive images from same camera). Its worst effect is to reduce the number of low altitude images, which are certainly of high interest.

In summary, after a consideration of the various factors involved, it has been decided (a) not to record the images obtained during blackout, (b) to utilize a 50,000 bps transmission rate, and (c) to employ a 5 second interval between the imaging frames alternately obtained from the two image storage vidicon cameras.

REFERENCES

A-1 JPL RFP No. VO-6-4509, Enclosure 6, "1973 VOYAGER Capsule Systems Constraints and Requirements Document", Revision 2, 12 June 1967.

PART B

OBJECTIVES AND REQUIREMENTS

Primary mission objectives of the Entry Science Package (Section 1) are to obtain atmospheric data during descent and pictures of the Mars surface for transmission to Earth via the Flight Spacecraft. The mission profile and timeline (Section 2) are dependent on the atmospheric entry conditions of the Capsule Bus and the actual Martian atmosphere. To attain the ESP objectives imposes several functional and performance requirements (Section 3) on the Entry Science Package and the Capsule Bus.

SECTION 1

MISSION OBJECTIVES

The primary objective of the VOYAGER Project is to obtain data about the characteristics and history of Mars through experiments and observations performed in orbit about the planet, during flight through the atmosphere, and after landing on the surface. It is desired to better understand the evolution and status of the planetary body, its atmosphere, its chemistry, its environmental characteristics and biology, if any. It is an objective to obtain measurements during entry and descent which will contribute to our understanding of both the atmosphere and surface characteristics of Mars. It is an abjective to both supplement and provide a backup to the mission of the Surface Laboratory. It is an objective to contribute not only to investigation and understanding by the Scientific Community, but to contribute readily usable data to the Engineering Community to facilitate increased effectiveness of operation and design of subsequent missions.

The Planetary Vehicle,which will be sent on an interplanetary trajectory to Mars, will consist of four major systems. These are the Flight Spacecraft, the Capsule Bus, the Entry Science Package, and the Surface Laboratory. Each has a unique function to perform, and all must function properly for the VOYAGER mission to be a complete success. The Flight Spacecraft must maneuver into an orbit about Mars and transmit data back to Earth concerning the environment it finds. The Capsule Bus must separate from the Flight Spacecraft and deorbit, carrying the Entry Science Package and the Surface Laboratory into the atmosphere of Mars and landing them softly on the surface. The Surface Laboratory conducts a program of experiments on the Martian surface under direct control from Earth. The Entry Science Package measures and transmits atmospheric data to the Spacecraft, applicable to a determination of temperature, pressure, and composition versus altitude above the surface, and obtains pictures of the planet during the entry into the atmosphere and descent to the surface. The main objectives of the Entry Science Package are two-fold. First, it will provide data which will reduce the uncertainty about the Martian atmosphere and its properties; and second, it will obtain pictures of the Martian surface during the terminal descent of the Capsule Bus supplementing those obtained by the orbiting Spacecraft and the landed Surface Laboratory. There are several secondary objectives which, if achieved, could contribute to the success of the mission. For example, the ESP could obtain supplementary upper atmosphere measurements and also provide data for deduction of winds encountered during descent. An effort to contribute information for improved operation of subsequent vehicles, and to utilize the UHF link availability for post-touchdown measurements, to back-up Surface Laboratory measurements, appears desirable.

1.1 ATMOSPHERIC PROPERTIES - Present design models of the Martian atmosphere emphasize the uncertainty in such critical atmospheric parameters as density, pressure, composition, and scale height. A primary objective of the Entry Science Package is to obtain atmospheric data during the entry into and descent through the Martian atmosphere. This data will be used in conjunction with the experiment data from the Flight Spacecraft and the diurnal surface atmospheric data from the Surface Laboratory to reduce these uncertainties. The ESP will permit determination of the variation in temperature, pressure, density, and composition with altitude above the surface. The best measure of these atmospheric properties as a function of altitude, at least below the upper atmosphere, will come from the ESP. Present mission planning emphasizes the first VOYAGER flight for ESP use. The success or failure of the ESP in the first two Capsule entries into the Martian atmosphere will have a significant impact upon the overall accomplishments of the first flight. The ESP will make measurements of the density and temperature profiles at altitudes of consequence to the design and operation of subsequent VOYAGER missions, such as those altitude zones of high deceleration and high dynamic pressure. The composition and concentrations of the atmosphere at lower altitudes, important to the definition of Martian meteorology, will be determined. In addition, the ESP could provide data not only on the troposphere and stratosphere but also potentially for the higher atmosphere and ionosphere.

1.2 IMAGING - Pictures of the surface of Mars similar to the first close-up pictures of the surface of the Moon transmitted to Earth by Ranger would be very desirable. A second primary objective of the Entry Science Package is to take pictures of the Martian surface during descent and transmit these pictures to the orbiting Flight Spacecraft for relay to Earth.

If the Capsule Bus does not achieve its mission objective of a soft landing on the Martian surface, the pictures transmitted by the ESP will be the only pictorial record available. The pictures of the approach and terminal portion of the descent to the surface will provide more than a view of the landing site; these will also provide a record of the Capsule Bus descent attitude, the amount of roll and roll rate, and the descent velocity. These data will support the data collected from the engineering data sensors to better evaluate the performance of the Capsule Bus.

The ESP pictures will supplement the orbiter pictures with better resolution of the surface features and characteristics in the vicinity of the landing site. These pictures could also supplement the coverage obtained by the Surface Laboratory. As the Capsule Lander begins its descent, the ESP pictures will aid in pinpointing the landing site location. If Mars does have a cloud cover, the ESP pictures could provide data concerning the amount and some of the characteristics.

SECTION 2

MISSION PROFILE

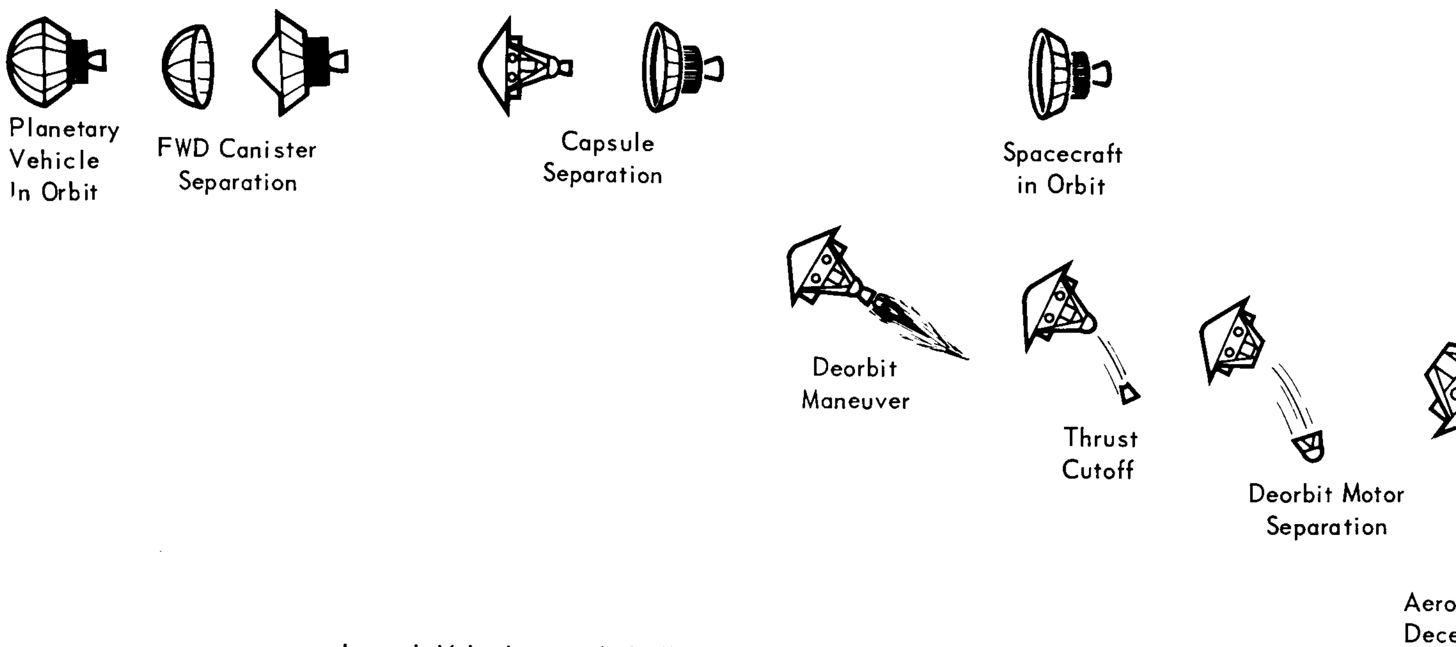
With the Entry Science Package carried within the Capsule Bus, the mission profile of the Capsule governs the flight conditions encountered by the ESP. The sequence of events and experiments performed by the ESP is designed for compatibility with the design range of Capsule Bus atmosphere flight.

2.1 CAPSULE BUS MISSION PROFILE - Within 30 days after the Planetary Vehicle is inserted into Mars orbit, the Capsule Bus is separated from the Spacecraft at a suitable time; a de-orbit motor is fired to bring the Capsule into the Martian atmosphere within the design entry corridor (velocity between 13,000 and 15,000 ft/sec and entry angle between -20° and vacuum graze); the drag of the Capsule reduces the velocity to a speed and altitude compatible with deployable aerodynamic decelerators; and final descent velocity is controlled by terminal propulsion. After landing, the Spacecraft remains in view of the Capsule for a minimum of 5 minutes. From atmospheric entry until the Spacecraft passes out of view, the Entry Science Package transmits data to the orbiting Spacecraft for relay to Earth. The major sequential events for the Capsule Bus from separation to landing are shown in Figure 2-1.

2.2 ENTRY SCIENCE PACKAGE MISSION PROFILE - The optimum time for initiation of sensing and transmission of entry science data is strongly influenced by the total power available. The operation of the ESP could commence at any time after the Capsule Bus begins its descent. However, because of the paucity of critical scientific data to be obtained during the long extra-atmospheric descent, no attempt will be made to obtain and transmit data prior to entry. Battery power must be conserved for image transmission at lower altitudes.

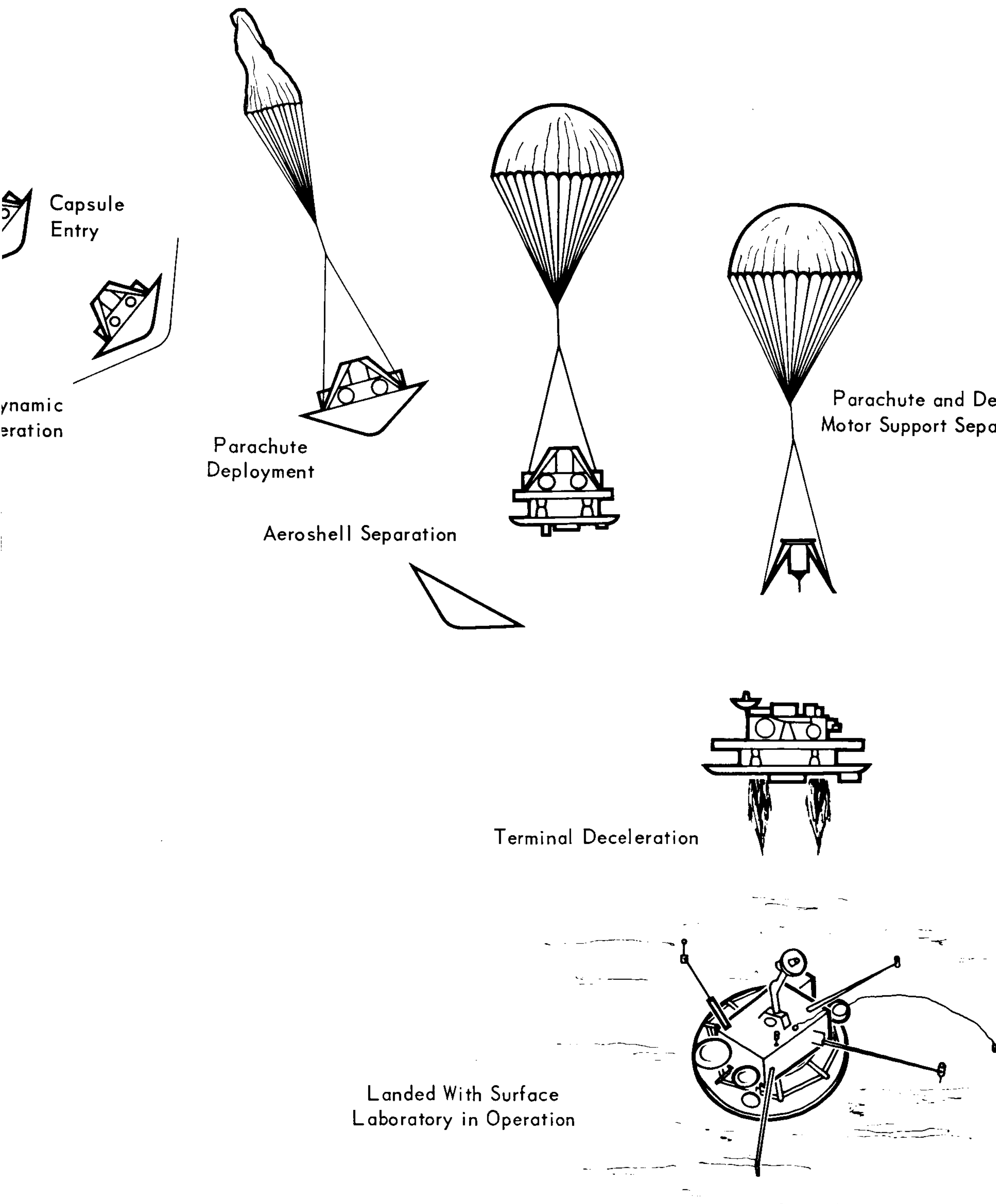
After the atmosphere is entered, data measurements will be made and imaging performed. Figure 2-2 shows the effect on altitude and time of the Capsule Bus entry conditions. With a dense atmosphere (VM-9), the Capsule decelerates at high altitudes. The blackout region (See Vol. II, Part B, Section 5.5.13), where the image data are lost, also occurs at high altitudes. With a shallow atmosphere (VM-8), this phenomenon occurs at a lower altitude. All data except TV are relayed at least twice, with a time delay for the repeat transmission.

MAJOR VOYAGER MISSION EVENTS



Launch Vehicle:	Liftoff Earth Orbit Heliocentric Transfer Injection
Planetary Vehicle:	Separation Midcourse Corrections Cruise Mars Orbit Insertion Orbit Operations
Spacecraft:	Capsule Separation Orbit Operations
Capsule:	Deorbit Descent Entry
ENTRY SCIENCE PACKAGE OPERATIONS	Atmospheric Flight Aerodecelerator Aeroshell Separation Terminal Propulsion Landing Post-Landing
Surface Laboratory:	Landed Operations

Figure 2-1


Capsule Entry
ynamic
eration
Parachute Deployment
Aeroshell Separation
Parachute and Deorbit Motor Support Separation
Terminal Deceleration
Landed With Surface Laboratory in Operation

ENTRY SCIENCE PACKAGE
ATMOSPHERIC PROFILE

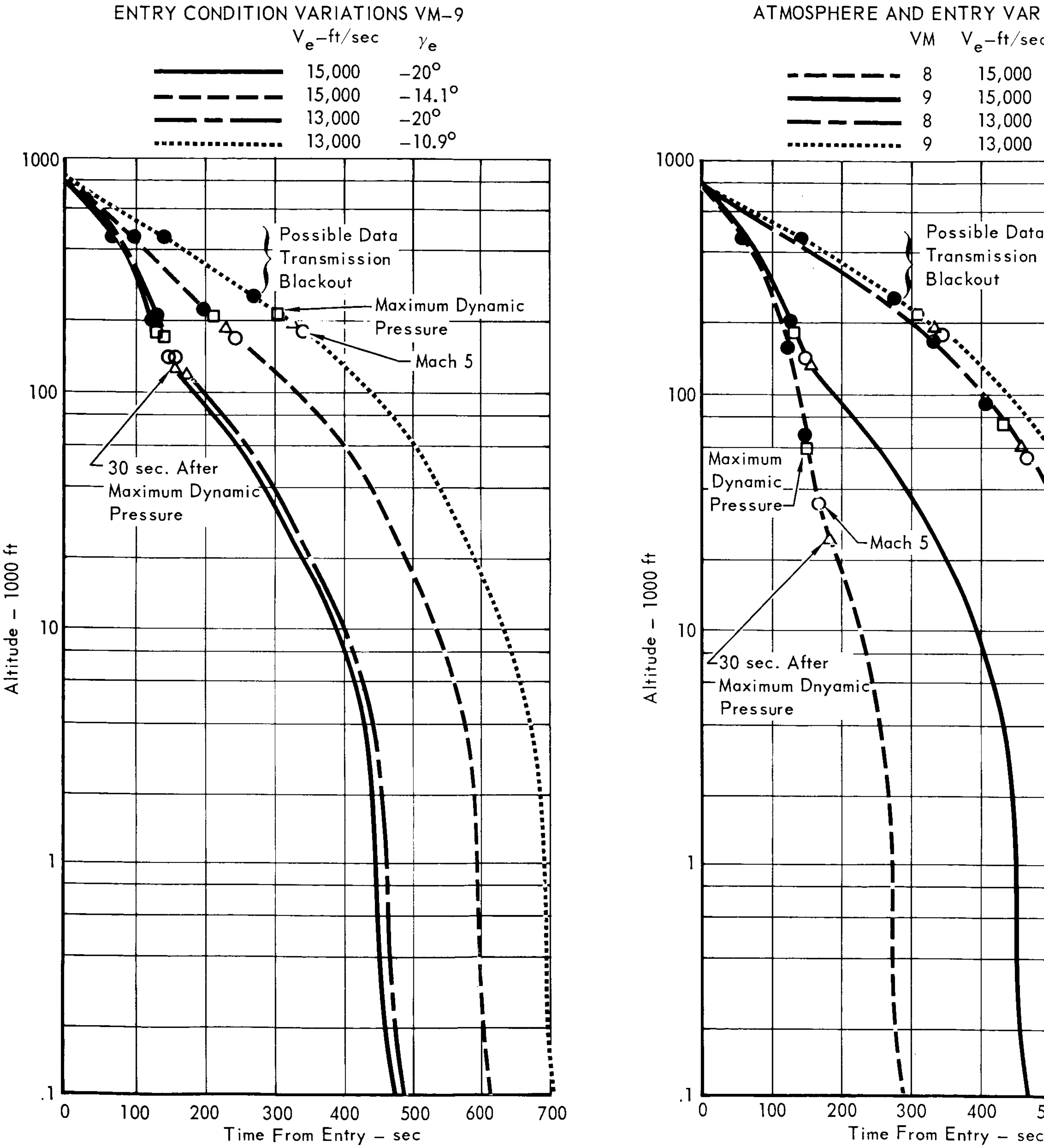
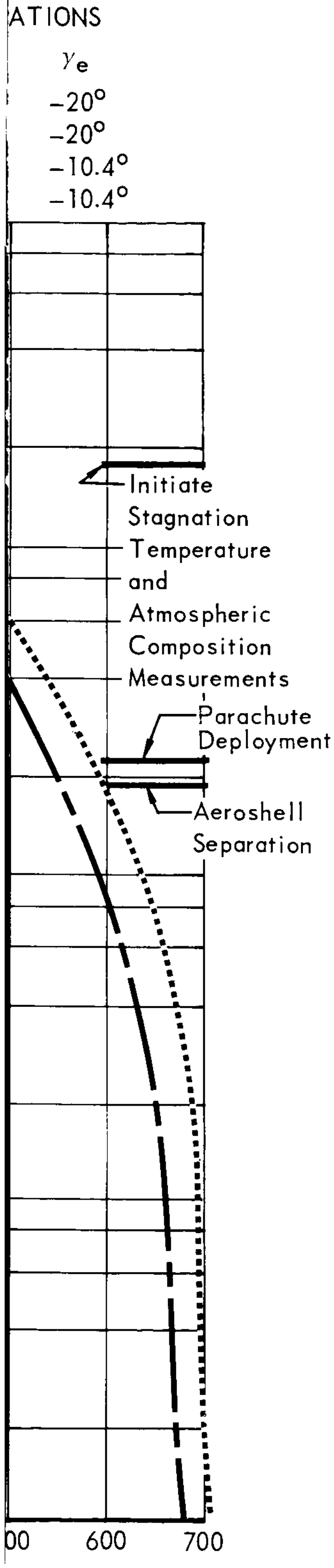
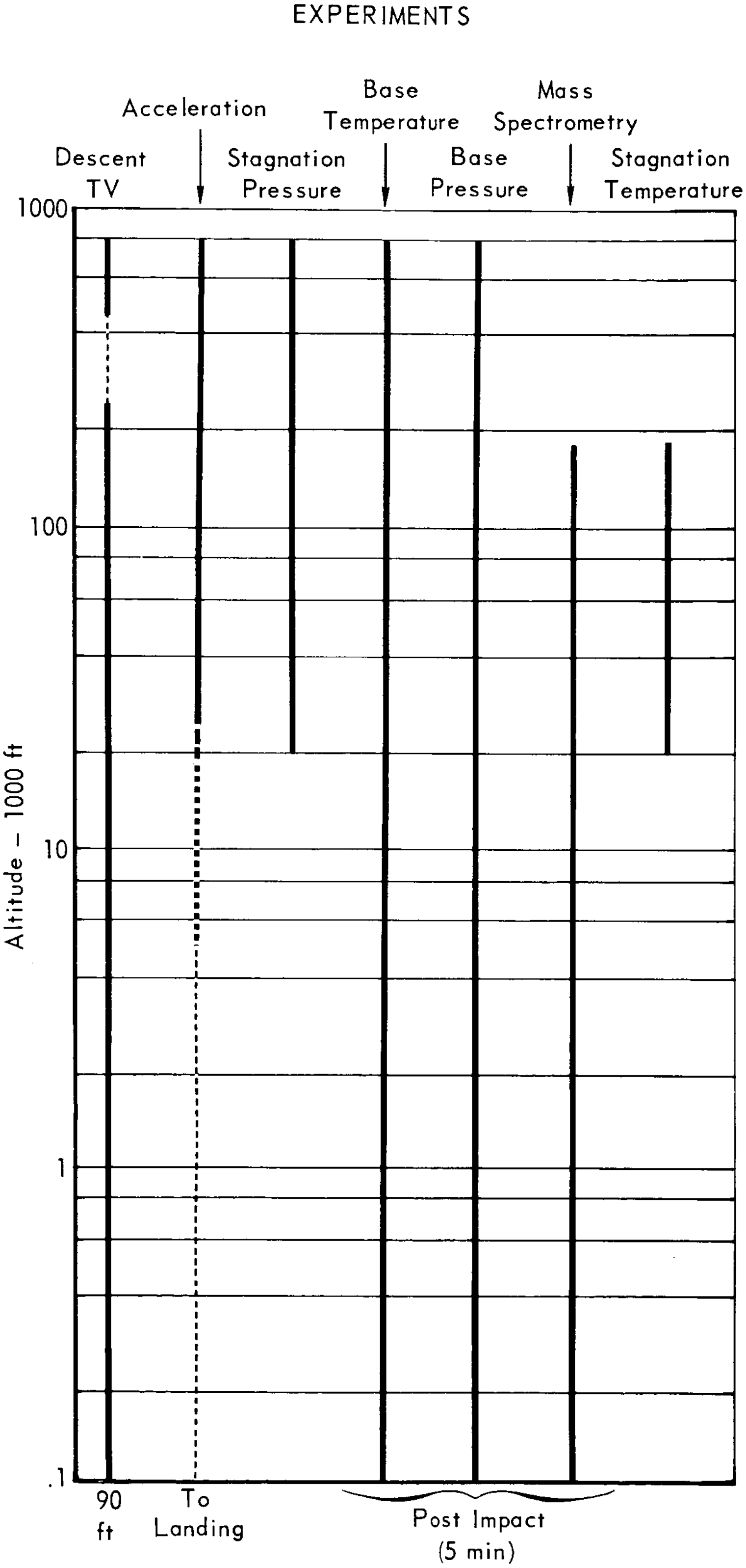


Figure 2-2


ATIONS
γ_e
-20°
-20°
-10.4°
-10.4°
Initiate Stagnation Temperature and Atmospheric Composition Measurements
Parachute Deployment
Aeroshell Separation
00
600
700


EXPERIMENTS
Descent TV
Acceleration
Stagnation Pressure
Base Temperature
Base Pressure
Mass Spectrometry
Stagnation Temperature
Altitude – 1000 ft
1000
100
10
1
.1
90 ft
To Landing
Post Impact (5 min)

High speeds may affect the operating temperatures of some sensors. Consequently, the stagnation temperature measurements and mass spectrometer sampling are not initiated until the capsule decelerates to about Mach 5. The peak dynamic pressure is used as a reference point for timing the start of these measurements. After the parachute deployment and Aeroshell separation, terminal propulsion rocket plumes will affect the measurements of the stagnation temperature and pressure sensors.

Transmission of the descent data from the ESP is continued after impact to provide complete data records of the non-imaging data. After the ESP landing, about 2 minutes of view time to the Spacecraft is required for this communication link. The de-orbit conditions for the Capsule Bus were constrained so as to provide a conservative minimum of 5 minutes of post-landed view. During this additional relay time, landed images could possibly be transmitted, utilizing the high bit rate of the ESP.

Figure 2-3 shows the variation of velocity with altitude in a dense atmosphere for the range of Capsule Bus entry conditions. The abrupt breaks in the curves are due to added retardation - parachute and terminal propulsion.

Although the ESP must be compatible with the flight conditions of the Capsule Bus, the ESP does place a restriction on the operation of the Capsule. To obtain good imaging during descent, the Capsule must land in an area which is between 15° and 30° from the terminator. This restriction on the lighting during descent must be met by the selection of orbit size and location, point and time of Capsule separation, and de-orbit. The discussion of these limits and of the choice of landing sites is presented in Volume II, Part B, Section 3.

2.3 ENTRY SCIENCE PACKAGE SUBSYSTEM TIMELINE - The 1973 VOYAGER Entry Science Package timeline illustrates the start, stop, and operating times of the various subsystems and major assemblies of the Entry Science Package. These are shown in Figure 2-4. Important events and their associated times of occurrence during the Entry Science Package equipment operation are indicated by triangle and explained in the legend at the bottom of the charts.

VELOCITY PROFILE
VM-9 ATMOSPHERE

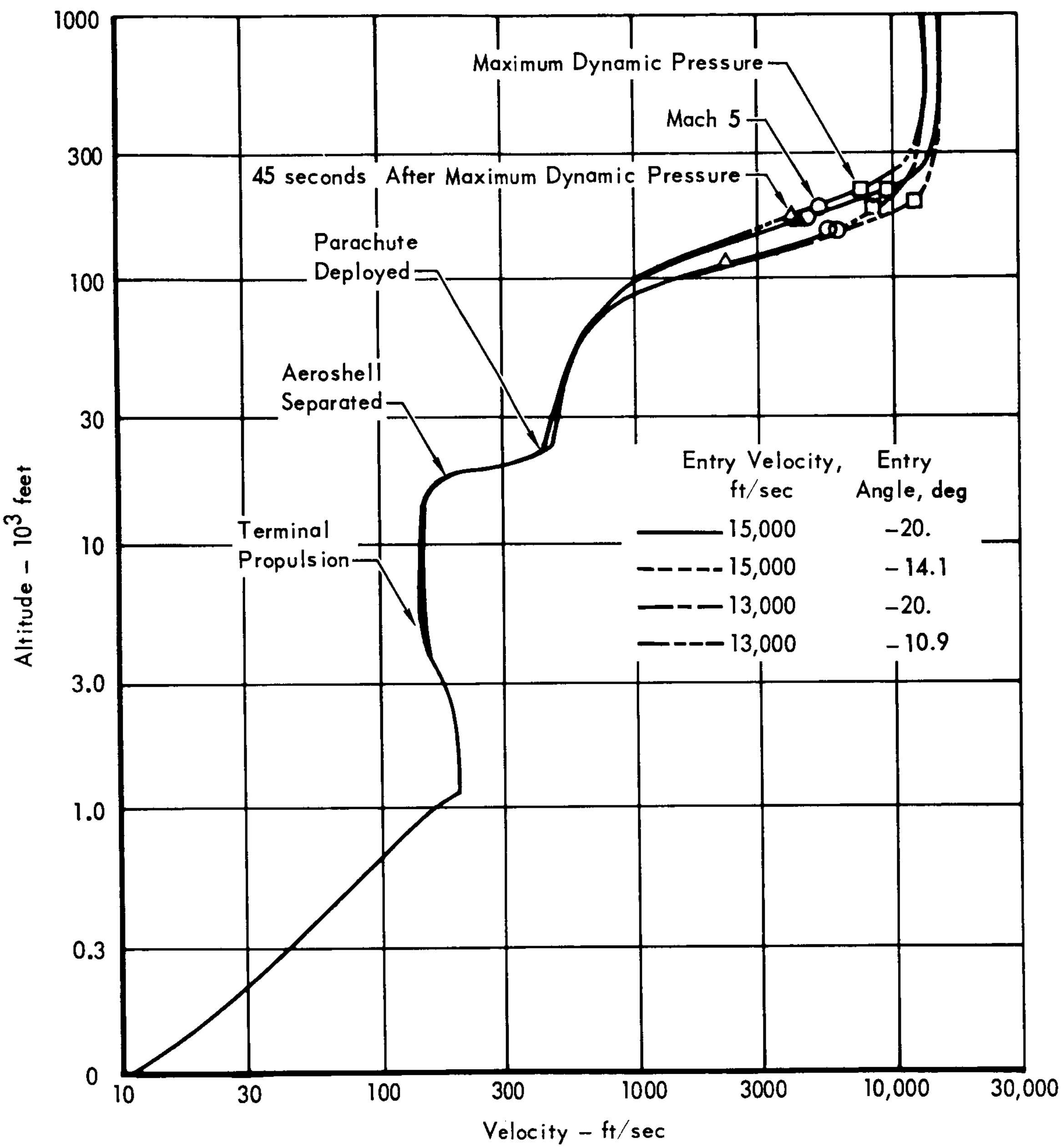
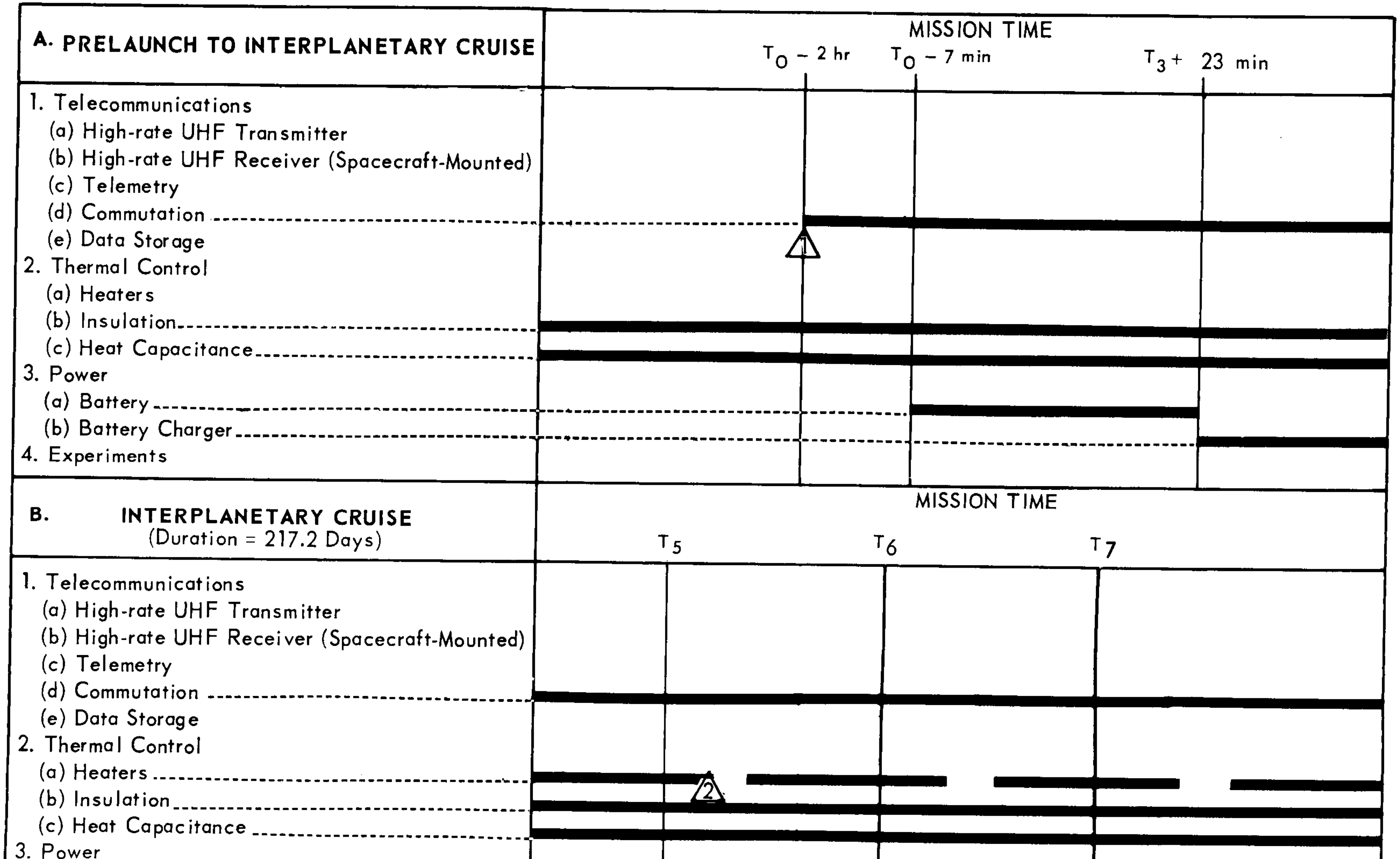
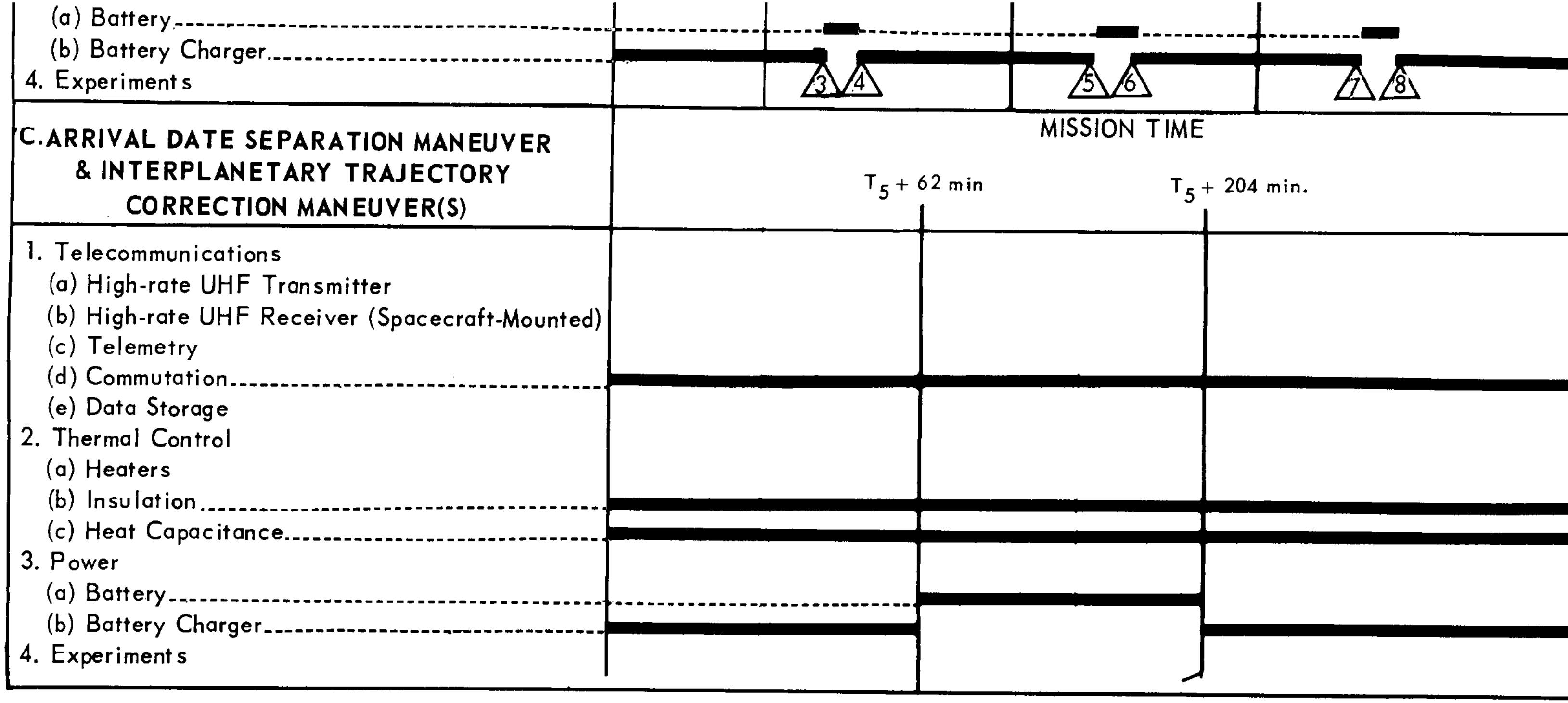


Figure 2-3

ENTRY SCIENCE PACKAGE EQUIPMENT OPERATION TIMELINE



2-6-1

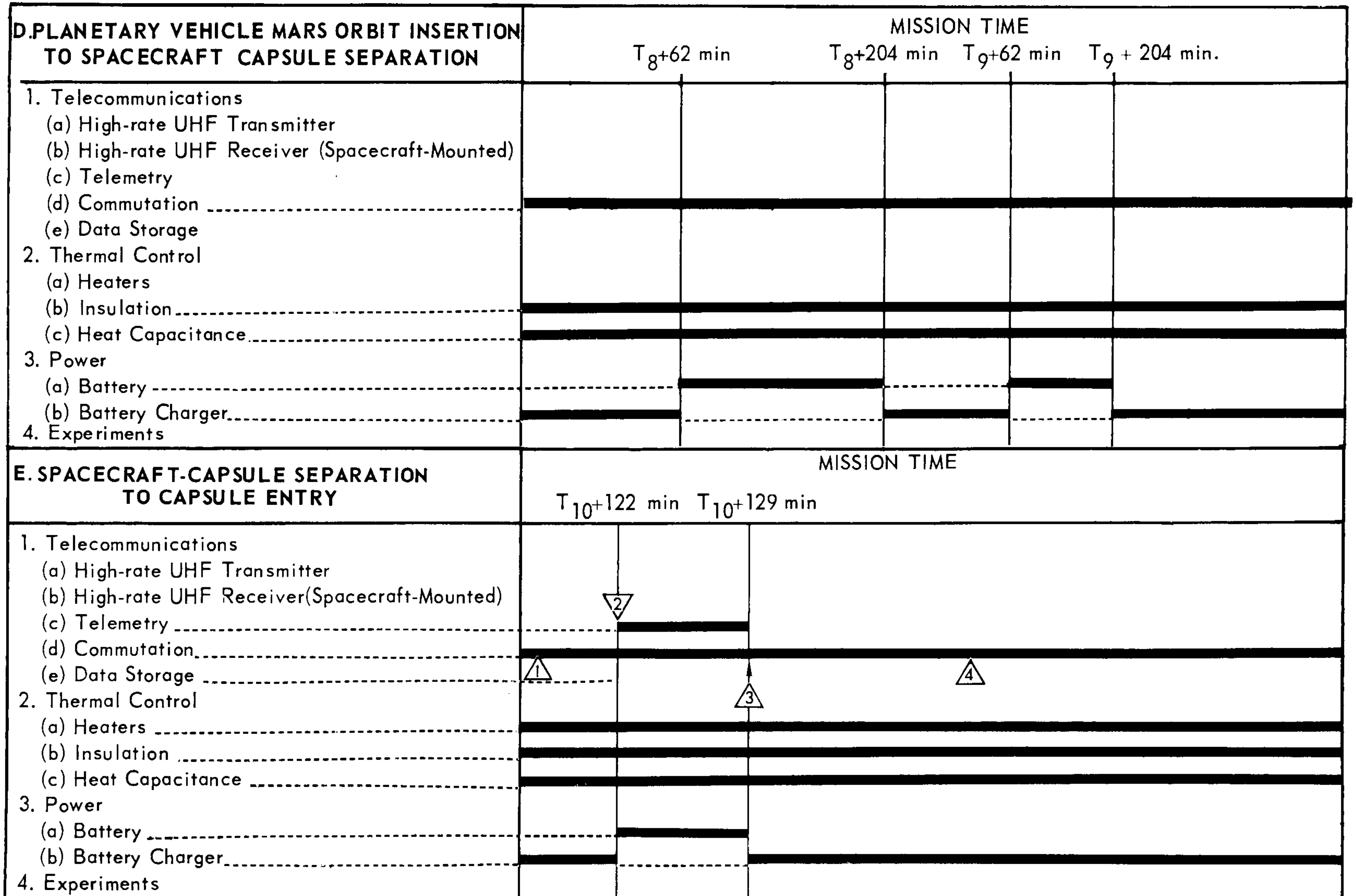


Notes:
1. Cruise Commutator on Cruise Mode
2. Heaters Operate Intermittently – Thermostatically Controlled
3. T_5 + 62 min
4. T_5 + 204 min
5. T_6 + 62 min
6. T_6 + 204 min
7. T_7 + 62 min
8. T_7 + 204 min

T_0 – Liftoff
T_3 – Separation of Forward Planetary Vehicle
T_5 = Arrival Date Separation Maneuver (Injection + 6.8 Days)
T_6 – First Midcourse Correction Maneuver (Injection + 30 Days)
T_7 – Second Midcourse Correction Maneuver (Mars-Arrival – 7 to 30 Days)

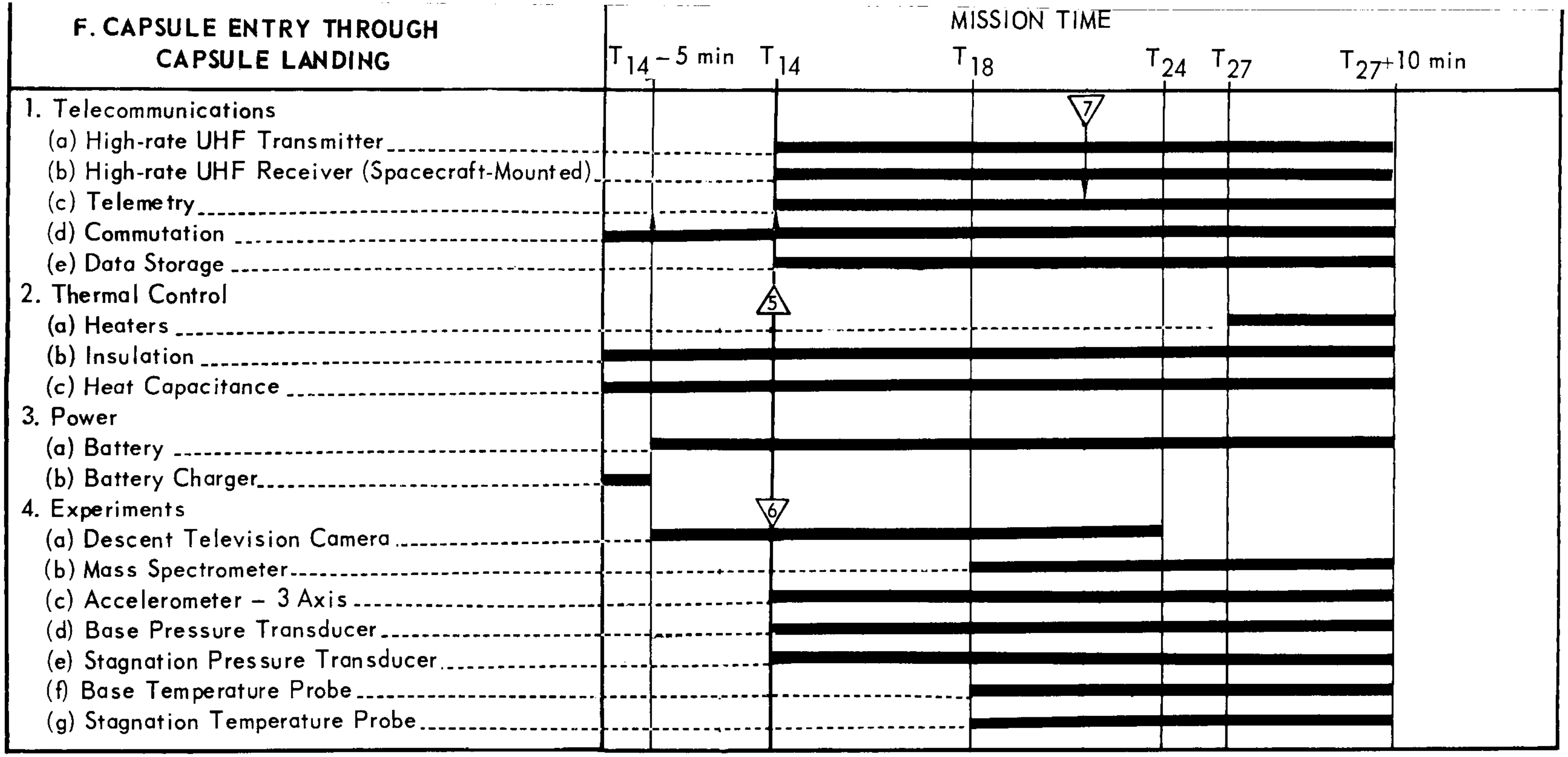
Figure 2–4

ENTRY SCIENCE PACKAGE EQUIPMENT OPERATION TIMELINE (Continued)



2-7-1

Figure 2



NOTES:

1. Switch Cruise Commutator to CB Telemetry Control Mode (T_{10} + 7 sec)
2. Switch Telemetry Subsystem to Checkout Mode (T_{10} + 122 min)
3. Switch Cruise Commutator to Cruise Mode (T_{10} + 129 min)
4. Switch Cruise Commutator to CB Telemetry Control Mode (T_{12} – 56 min)
5. Switch Telemetry Subsystem to Entry Mode (T_{14})
6. Initiate Descent Television Camera Sequencing (T_{14})
7. Switch Telemetry Subsystem to Terminal Descent Mode (T_{20})

T_8 – Mars–Orbit Insertion
T_{10} – Pre–Separation FC Checkout
T_{12} – FS–FC Separation
T_{14} – Mars Atmospheric Entry
T_{18} – Sense Mach 5
T_{20} – Separate CB Aeroshell
T_{24} – Release Descent Television Cameras
T_{27} – Touchdown on Sequence of Mars

-4 (Continued)

2-7-2

SECTION 3

ENTRY SCIENCE PACKAGE REQUIREMENTS

The Entry Science Package shall collect and transmit data to the Flight Spacecraft for relay to Earth on the atmospheric properties and surface features of Mars. Four complete units and one Proof Test Model shall be delivered in November 1972, available for launch on 20 June 1973. The combined Surface Laboratory and Entry Science Package weight shall be a maximum consistent with established Capsule Bus, Surface Laboratory and Entry Science Package design margins. To the extent practicable, the Capsule Bus and Entry Science Package shall be mutually independent, separable and self supporting. The CB-ESP interfaces shall anticipate changes, including possible deletion, in subsequent opportunities.

3.1 Functional Requirements - The Entry Science Package shall be an automatic device to:

a. Perform all entry science measurements.
b. Transmit all entry science data to the Flight Spacecraft for relay to Earth.

The entry science functions are assumed to include:

a. Descent imaging to obtain a sequence of views starting with images of the planet limb and high altitude oblique views of the surface, followed by images of decreasing area and increasing vertical aspect and resolution.
b. Measurement of time history of acceleration for use in determining the atmospheric density profile.
c. Measurement of time history of interpretable pressures for supplemental use in determining the atmospheric properties profile.
d. Measurement of time history of interpretable temperatures at low Mach number for supplemental use in determining the atmospheric properties profile.
e. Use of mass spectrometer to determine composition of the lower atmosphere, including possible information on water vapor content.

The Entry Science Package shall include provision for in-flight checkout and monitoring.

3.2 PERFORMANCE REQUIREMENTS - The Entry Science Package shall accommodate at least 45 lb of science instruments and science support equipment (science data handling, control and other equipment required solely for the support of the science instruments). The ESP shall occupy volume and use the weight allocation assigned to the Surface Laboratory.

The Entry Science Package shall be capable of transmitting at least 5×10^6 bits of science and engineering data during entry and atmospheric descent. Atmospheric data collected during periods of ionization (blackout) shall be stored for relay prior to ESP shutdown.

The imaging requirements on the ESP and for accommodation on the Capsule Bus are:

a. Pitch and yaw attitude rates to be less than approximately 4°/sec.

b. Condensates on the viewing window, and luminous articles in the flow pattern past the window, to be minimized.

c. Viewing window to be of high temperature resistant optical material; to provide greater than 50° field of view; to be located to facilitate camera viewing (axis parallel to CB axis).

d. Camera to use dual one-inch vidicons with 200 x 200 picture elements; 6 bits/element; format size of 0.44 in.

e. Camera resolution to be equal to or better than that required "to obtain resolutions identifiable with orbiter TV down to 1 meter per line pair." This resolution to be achieved or exceeded in last image even in the event of failure of the terminator aerodynamic and propulsion decelerators

f. Cycle time to be adequate for continuity of area identification.

The requirements of the ESP for measuring atmospheric properties and for accommodation on the Capsule Bus are:

a. Atmospheric density, pressure, and temperature profiles, each with a precision on the order of $\pm$ 1 percent.

b. Sensors to be located for ease of interpretation of measurements, considering flow field relationships, effect of angle of attack, and accelerometer corrections.

c. Contaminants in the sensor influent gages to be minimized.

d. Capsule Bus to provide stability, aerodynamic predictability, adequate data gathering time, and low interference with measurements, as compatible with Lander mission requirements and constraints.

PART C

DESIGN CRITERIA AND CONSTRAINTS

The requirements and constraints that have a major influence on the design of the Entry Science Package (ESP) are identified in this section. Some are taken directly from the JPL documents and some are self-imposed and are the result of system analysis. They include General Constraints, Interface Constraints, Environmental Requirements, Structural Design Criteria, Electromagnetic Compatibility, and Experiment, Equipment, and Subsystem Constraints.

SECTION 1

GENERAL CONSTRAINTS

The constraints that are most important in the design of the Entry Science Package are:

a. The single short launch opportunity for 1973 as shown in Figure 1-1
b. A conservative engineering approach
c. Accommodation of at least 45 lb of science instruments and science support equipment
d. Independent, separable, and self-supporting operation
e. Fully automatic operation
f. Planning so that no single failure will have a catastrophic effect on the mission
g. Compatibility of Capsule Bus and mission profile
h. Reliability requirements for design simplicity, flight-demonstrated practices, conservative margins, and selective redundancy.

SECTION 2

INTERFACE CONSTRAINTS

Interface requirements and constraints are tabulated in Figure 2-1.

VOYAGER MISSION CONSTRAINTS

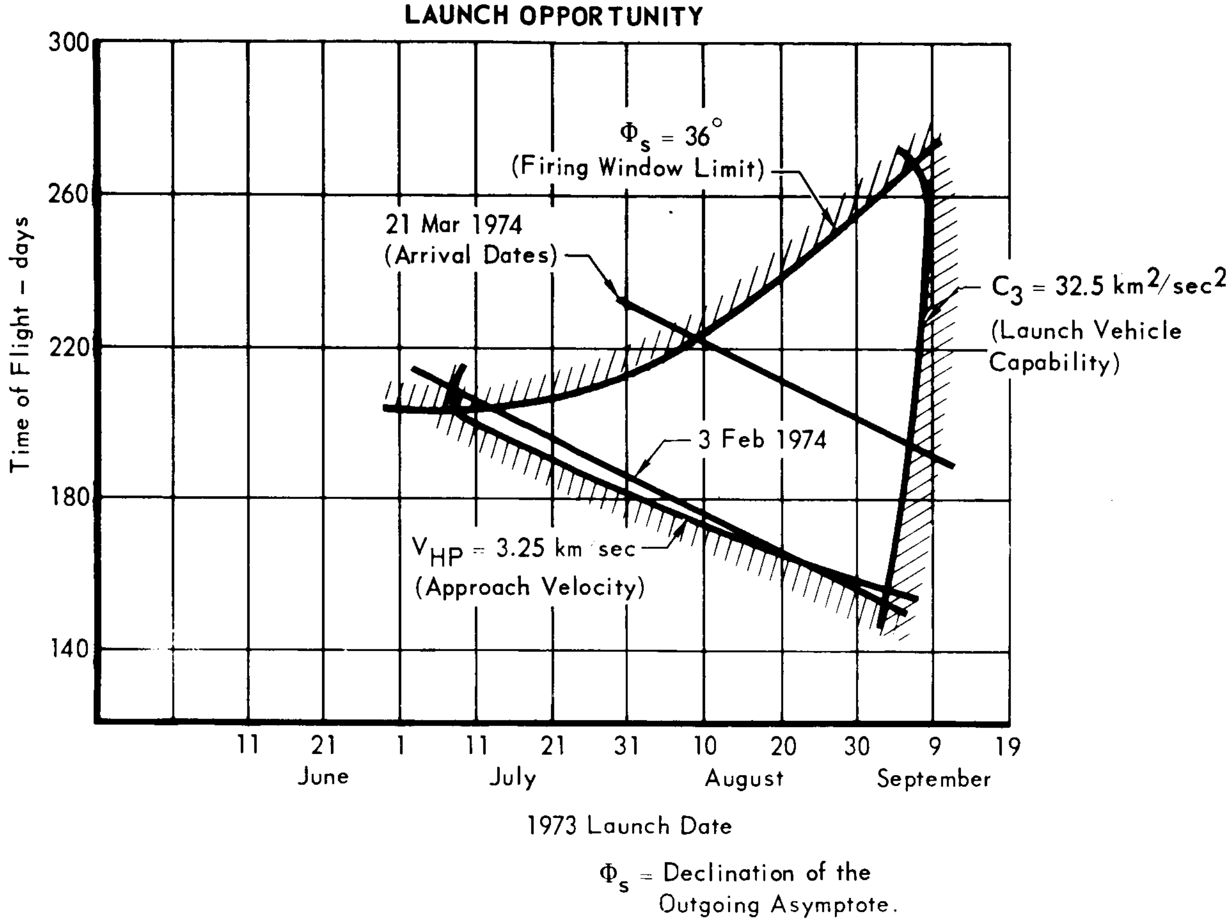


Figure 1–1

SYSTEM INTERFACE REQUIREMENTS AND CONSTRAINTS
ENTRY SCIENCE PACKAGE (ESP)

ESP TO OTHER SYSTEM INTERFACES	GENERAL	ELECTRICAL	STRUCTURAL/MECHANICAL
CB	1. Standardization for future mission. 2. Provisions for post 1973 opportunity ESP changes (including deletion of ESP)	ELECTRICAL POWER 1. CB shall distribute SC power to ESP during cruise with SC on/off control. 2. Switching conditions nor single failure mode will allow ESP to power SC. 3. CB shall provide ESP to SC short circuit protection so that ESP failure will not inhibit CB or SL battery charging. No Fuses. 4. Backup SL battery power shall be provided to ESP via CB in event of ESP power failure. SEQUENCING 5. Initiation of touch down commands shall be supplied to ESP by CB. DATA 6. ESP engineering data routes thru CB to SC during cruise. Appropriate SC Sync. and control signals to be provided to ESP by CB. 7. ESP engineering and science data with Sync. signal shall be routed thru CB to SC during in-flight checkout prior to FC separation. 8. ESP engineering data and low rate science data shall be provided to CB during entry for transmission to SC via CB-SC relay link. 9. CB engineering data is provided ESP during entry for transmission to SC via ESP-SC relay link. COMMANDS 10. SC commands are routed thru CB to ESP. INFLIGHT CHECKOUT 11. Control signals from CB test programmer shall be routed thru CB to ESP. EMC 12. Conducted or radiated interference generated by ESP or CB shall not degrade performance of PV. CABLING 13. Interface connectors shall be located with straight and free engagement of contacts and are accessible without disturbing other cabling. 14. ESP connectors are rigidly mounted. 15. CB connectors are on flexible cabling, having manual disconnects. 16. All interface circuits shall be insulated (100 megohm minimum between each conductor and all other conductors and shields connected in parallel, and between each conductor and conductor shells.) (20 megohm minimum between all conductors and their associated shields.) 17. ESP receptical shall have socket contacts (compatible with all VOYAGE environments.) 18. Interference of signals within the cabling network and interface connectors shall be minimized (by isolation per signal type by physical separation – with interface connectors and within each connector.)	1. TV camera looks through aeroshell during entry and descent. 2. ESP remotely mounted packages and probes. 3. TV camera field of view parallel to CB roll axis. 4. TV camera ejected prior to landing. 5. Accelerometer as close as possible to CB c.g.
SL	[Same as Above]	1. Backup SL battery power provided to ESP via CB/ESP interface.	N/A
SCS	[Same as Above]	1. ESP science and engineering data to the SC via ESP to SC RF relay link during entry.	[To be derived with SC contractor]
LOS	[Same as Above]	1. ESP equipment compatibility with launch pad environments.	N/A
MOS	[Same as Above]	1. ESP data shall be descriptive of ESP status for mission control	N/A

Figure 2–1

THERMAL	SCIENCE	OSE
1. Minimize heat transfer paths across field joints. 2. Electrical power required for ESP heaters – power requirement depends on ESP equipment temperatures. 3. Controlled temperature environment required prior to landing. 4. Heating from rocket motors, base region during entry. 5. Temperature monitoring throughout mission.	1. Test patterns for descent imagers must be aligned if they are included in ESP. 2. Angle-of-attack excursions below Mach 5.0 are limited to $\pm 20^\circ$ for accurate stagnation temp. data. 3. Descent imagers require that roll rates not exceed 50°/second and pitch and yaw rates 4°/second. 4. CB pressure coefficients determination constrain ESP data interpretation. 5. Aeroshell design must accommodate ESP stagnation sensors and imager windows. . 6. Altimeter antenna and stagnation region assembly must operate on a non-interference basis. 7. Stagnation temperature sensor vent tube must be routed through the CB in order to obtain mass spectiometer samples. 8. CB must provide Δt update to pressure switch in stagnation assembly. 9. ESP accelerometer must be located near CB c.g. and have capability to be aligned along CB body axis. 10. Descent imager viewing access must be provided through the aeroshell. 11. Wire bundles for aeroshell mounted ESP equipment must be severed prior to aeroshell separation.	1. The ESP control room OSE shall be compatible with the CB control room. 2. The ESP TM shall be compatible with the CB ground station. 3. The ESP contractor shall provide an interface simulator for integrated flight capsule tests. 4. The ESP shall contain sufficient test point access for integrated flight capsule tests.
N/A	[ESP data obtained on surface will serve as backup]	N/A
[To be derived with SC contractor]	[SCS & ESP Science Data may be mutually supporting]	1. The SC flyaway umbilical shall contain pins for handling Flight Capsule test data, critical analog parameters and RF coax cables for PV tests. 2. The SC contractor shall provide interface simulator for RF and hardline compatibility testing prior to mate. 3. The SC contractor ground stations shall receive, extract, and route Flight Capsule TM data to the Flight Capsule STC.
N/A	1. Safety inputs for radiation and pressure areas. 2. Special handling procedures for science instruments.	THE LOS PROVIDES 1. Ground power compatibility. 2. Facility Cabling (A2A lines) 3. OSE Test Areas.
N/A	1. Mission contingency. 2. Atmosphere profile and trajectory parameter reconstruction computer inputs.	N/A

SECTION 3

ENVIRONMENTAL REQUIREMENTS

Major attention has been given in this study to the environmental requirements that are unique to VOYAGER, as stipulated by the JPL Environmental Predictions Document and Capsule Systems Constraints and Requirements Document. These can be placed in the following categories:

a. Decontamination and sterilization

b. Long term storage in a deep space environment

c. Mars Atmospheric characteristics

d. Mars surface characteristics.

3.1 DECONTAMINATION AND STERILIZATION

a. Ethylene oxide (ETO) decontamination requires compatibility certification of parts and materials

b. Dry-heat sterilization time and temperature cycles, $135^{o}C$ for 92 hours and $125^{o}C$ for 60 hours, require careful selection of parts and materials for compatibility.

3.2 STORAGE IN DEEP SPACE ENVIRONMENT

a. Temperature approaching $-459^{o}F$ as a radiation sink

b. Pressure of 10^{-14} torr or less.

3.3 MARS ATMOSPHERIC CHARACTERISTICS - Martian surface environments are:

a. Ambient temperatures from $+120^{o}$ to $-190^{o}F$

b. Ambient pressure from 5 to 20 millibars
The altitude/pressure profiles are shown in Figure 3-1

c. Atmospheric density from 6.4×10^{-6} to 3.8×10^{-5} gm/cm^3
The altitude/density profiles are shown in Figure 3-2

d. Mars free stream winds, wind gusts, and wind gradients as indicated for the worst design case are shown in Figures 3-3 and 3-4.

3.4 SURFACE CHARACTERISTICS

a. Slopes up to 34 degrees from local vertical and up to 100 meters long; and 10 degree slopes up to 2 km long.

b. Surface roughness ranging from sand particles with 10 micron diameters to rocks with 5 inch diameters.

c. Surface albedo for sunlight of 0.05 to 0.35 with an emissivity of 0.85 to 1.00.

ATMOSPHERIC PRESSURE
MODELS VM-1 THROUGH VM-10

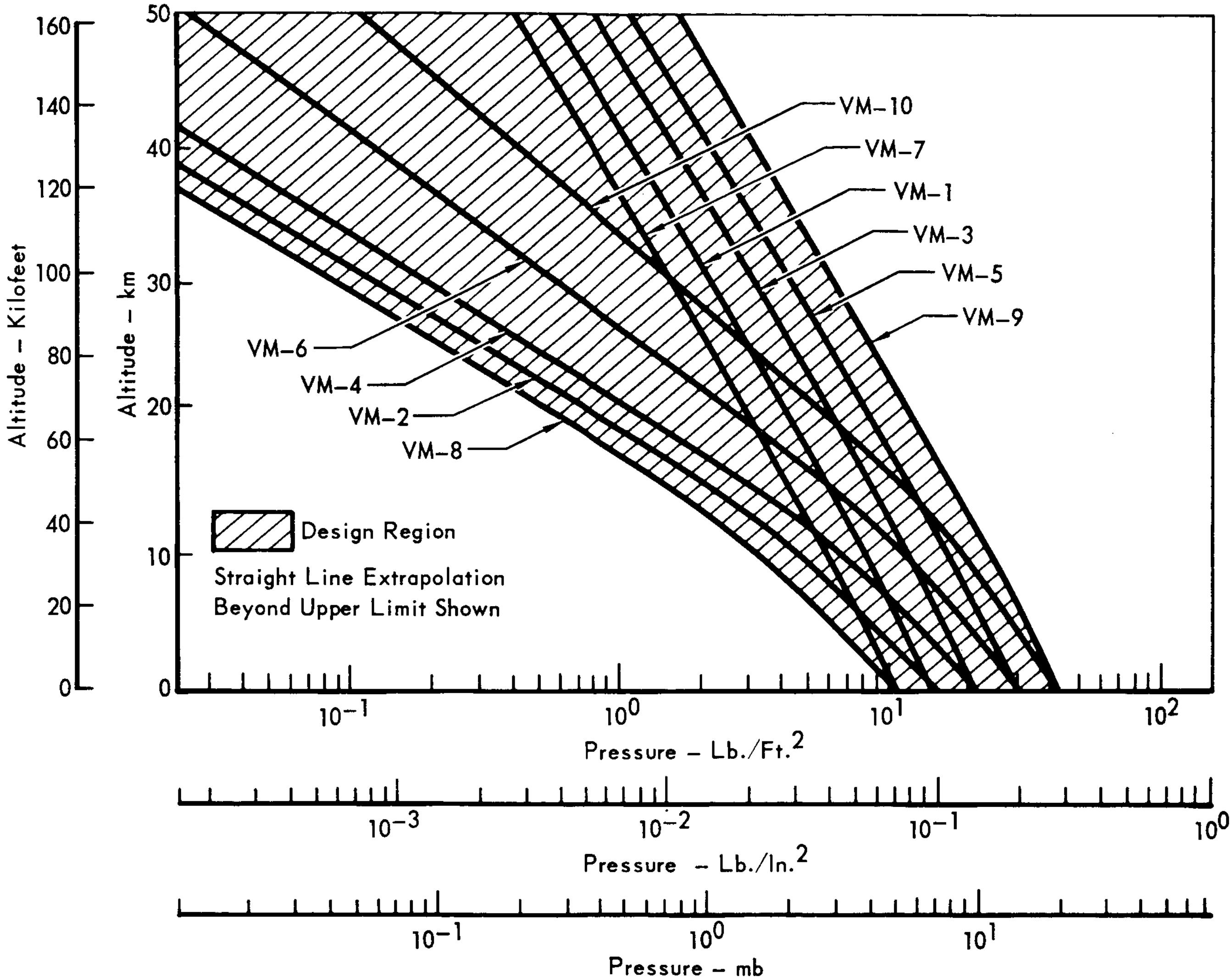


Figure 3-1

ATMOSPHERIC DENSITY
MODELS VM-1 THROUGH VM-10

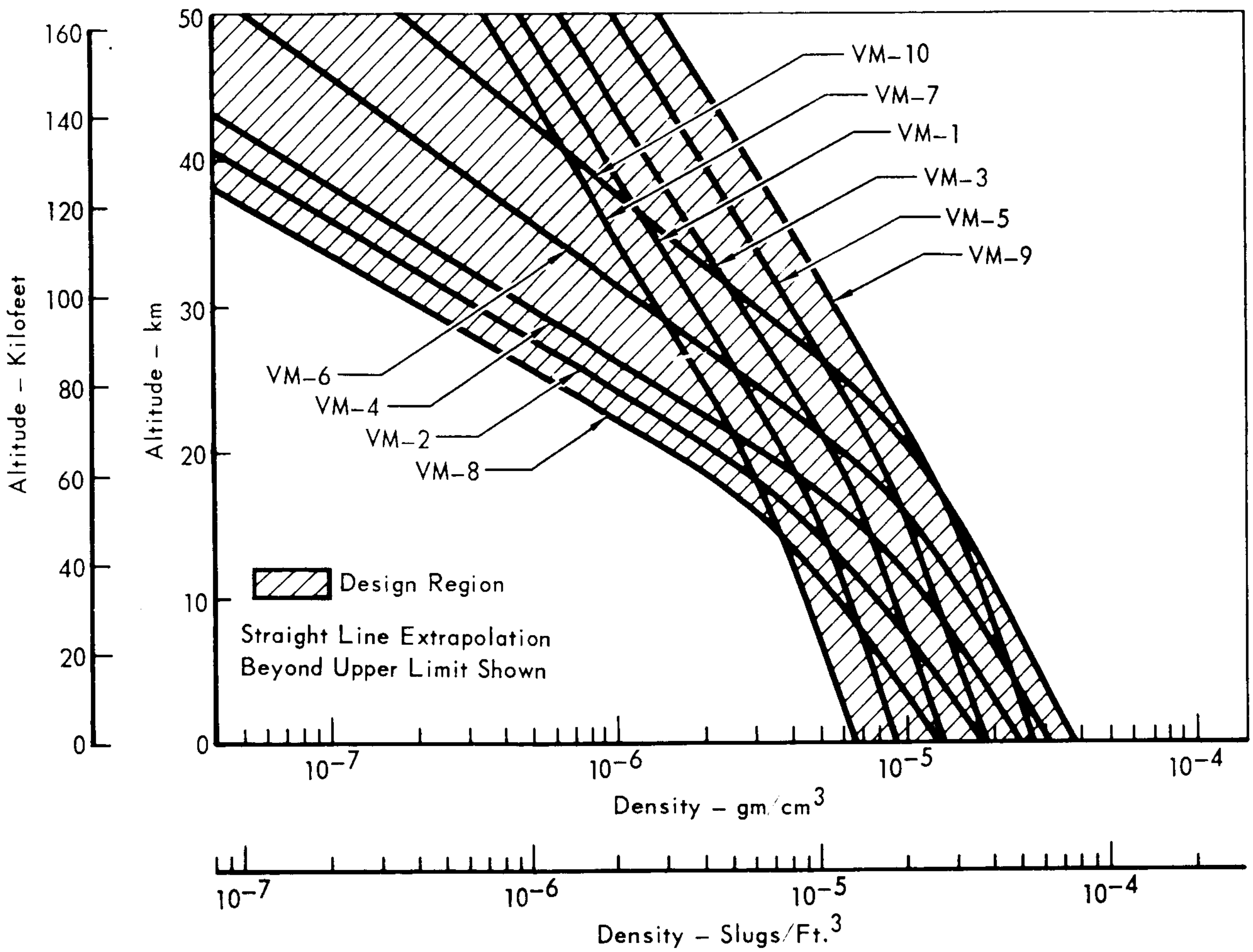


Figure 3-2

ENTRY WIND PROFILE
VM-7 AND VM-8 ATMOSPHERE

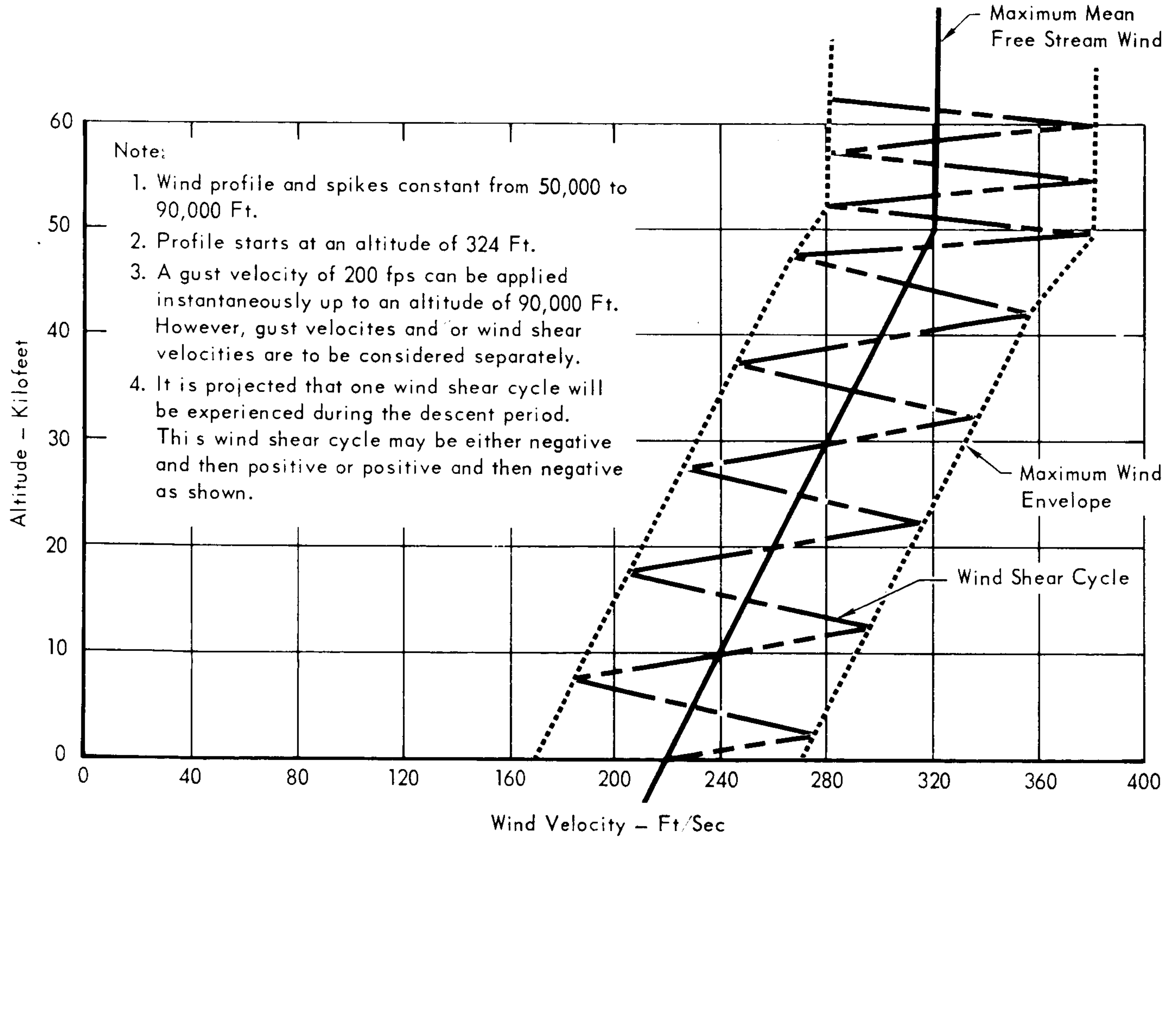


Figure 3-3

SURFACE BOUNDARY LAYER PROFILE
VM-7 AND VM-8 ATMOSPHERE

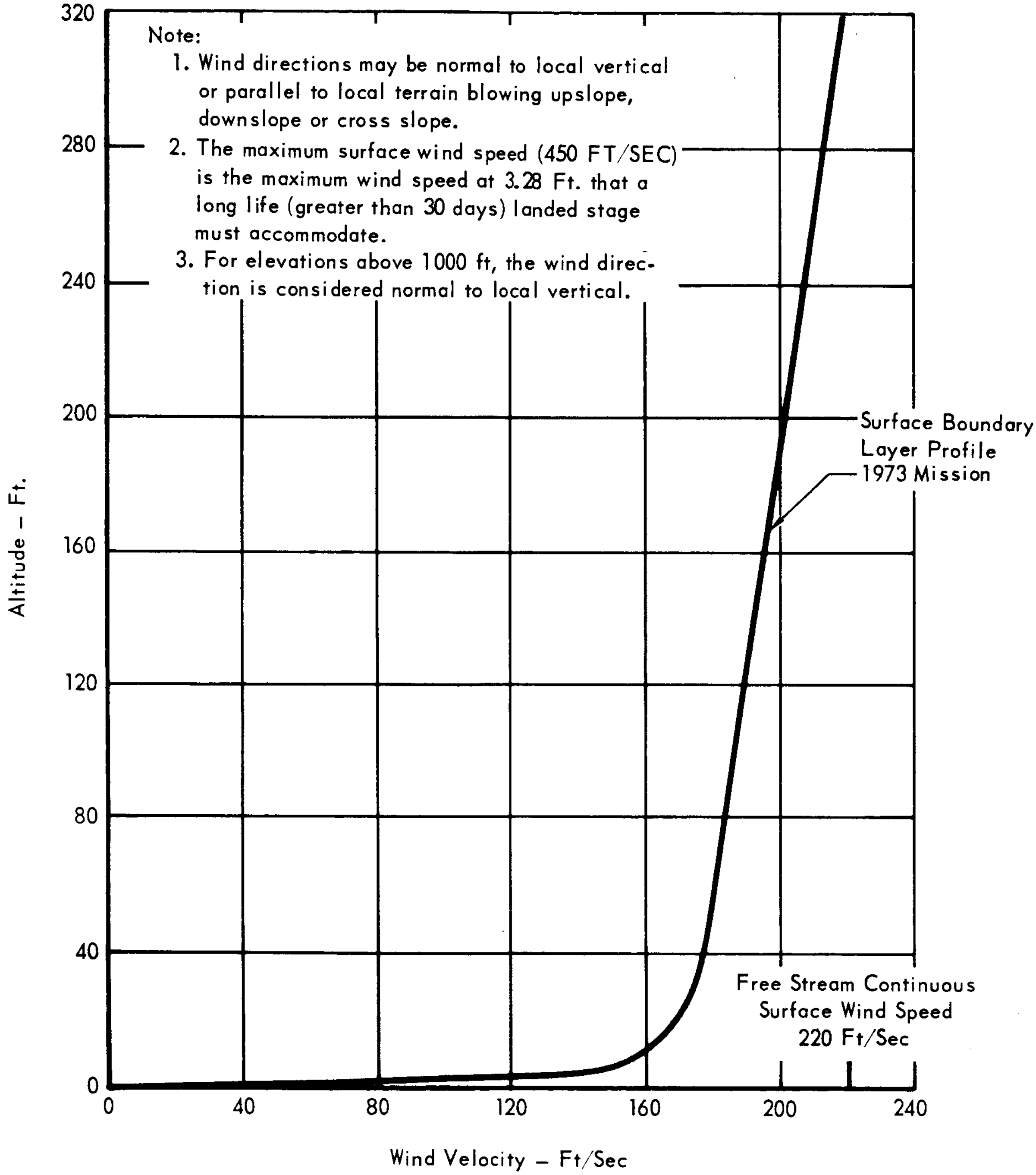


Figure 3-4

3.5 SOURCE OF ENVIRONMENTAL REQUIREMENTS - The data for the ESP environmental specifications were obtained from the following sources:

a.	SE 003 BB 011-1B28, dated 20 October 1966	Draft Voyager Environmental Predictions Document
b.	SE 003 BB 002-2A21, dated 12 June 1967	Voyager Capsule System Constraints and Requirements Document
c.	VOL-50503-ETS, dated 12 January 1966	Environmental Specification (Decontamination and Sterilization)
d.	MIL-STD-810A, dated 23 June 1964	Environmental Test Methods for Aerospace and Ground Equipment
e.	DA AR 705-15, C1, Dated 14 October 1963	Research and Development of Material - (Operation of Material under Extreme Conditions of Environment)

Appendix A in Volume II, Part A, gives the more specific environmental design requirements governing the ESP design. These requirements were selected to assure survival and satisfactory performance under any reasonable combination of environmental conditions cited in the source documents.

Induced shock and vibration environments are referenced in Part C, Section 4. In addition, the dynamic thermal and pressure environments may be correlated with the range of stagnation temperatures and pressures shown in Part D, Section 4.

SECTION 4

STRUCTURAL DESIGN CRITERIA

The structural criteria for the ESP are based on the data presented in Volume II, Part 4, Section 2 of the CB. These data were expanded to support the ESP structural analyses. Criteria of significance are summarized here.

4.1 STRENGTH

a. The structure must withstand limit load with the structure at predicted temperature.

b. The structure must withstand limit load at design temperature or ultimate load at predicted temperature, whichever is more critical. Effects of temperature gradients are accounted for by adding the thermal stress associated with predicted temperature to the stress which results from limit mechanical loads, and multiplying the resulting stress by the factor of safety for mechanical loads.

c. The predicted structural temperatures and heating rates are based on dispersed trajectories. The design temperature for radiative structure is the initial entry temperature plus the predicted temperature rise multiplied by a temperature uncertainty factor of 1.15. The design temperature for ablative protected structures is the initial entry temperature plus the predicted temperature rise multiplied by a temperature uncertainty factor of 1.25.

4.2 STIFFNESS

a. Structural stiffness must prevent unintentional contact between ESP components or between the ESP and other Flight Capsule segments.

b. Structural stiffness must minimize dynamic coupling with other segments of the Flight Capsule.

4.3 FACTORS OF SAFETY - A factor of safety of 1.25 must be maintained for all phases except ground handling and transportation, which may be hazardous to personnel. In these cases, the factor of safety must be 1.5.

4.4 MISSION PHASE LOAD FACTORS - A summary of the maximum load factor requirements for the ESP is presented in Figure 4-1. These data are rigid body values at the c.g. of the ESP.

MAXIMUM LOAD FACTORS FOR THE ENTRY SCIENCE PACKAGE

MISSION PHASE	*LIMIT LOAD FACTOR (EARTH g)	
	LONGITUDINAL	LATERAL
Launch	+ 4.9	± .10
Entry	−31.0	±4.75
Landing	−10.0	±10.0
	−14.0	0

* Rigid body load factors at Entry Science Package c.g.

Figure 4-1

SECTION 5

ELECTROMAGNETIC COMPATIBILITY

Interference-free operation of all VOYAGER components is ensured by employing appropriate design controls and electromagnetic interference (EMI) tests at both the equipment and system levels.

5.1 EQUIPMENT LEVEL INTERFERENCE - The following basic EMC design criteria apply to all ESP support subsystems and electrical and electronic equipment units of the science instruments.

a. Conductor Shielding - A conductor shield must not be used as a conductor. Conductor shielding must not be used where separation or circuits or twisting of conductors is more effective. Individual return conductors must be included in the same shield with the "hot" conductor. Interconnections between RF circuits that do not employ waveguide must use coaxial cable, or balanced-shielded RF cable.

b. Types of Grounds and Returns - In each equipment item, separate buses must be provided for ac, dc, and signal returns for audio frequency and secondary dc power circuits. A minimum dc isolation of 1 megohm must be maintained between primary power circuits.

c. Audio Frequency Grounds - A minimum dc resistance of 1 megohm must be maintained between all audio frequency input or output circuits (0-150kHz) and the equipment case. Audio shields must be grounded at only one point.

d. Radio frequency Grounds - Radio frequency circuits (above 150 kHz) may use bonded enclosures or structural members to obtain ground reference. Conductor shields, including coaxial outer conductors, must be continuous and grounded at both ends and all convenient intermediate points.

5.2 SYSTEM LEVEL INTERFERENCE - The following design criteria delineates the additional system level requirements over and above the equipment requirements for the interconnected ESP.

a. Grounding - A separate central ground point (CGP) must be located in the Entry Science Package and electrically referenced to the Capsule Bus CGP when mated.

b. Conductor Shield Grounds - Shields of audio-frequency conductors (0 to 150 kHz) must not be grounded at more than one point. Shields of radio-frequency conductors (above 150 kHz) must be grounded at each end and all convenient intermediate points.

c. Electroexplosive Device (EED) Firing Circuits - EED circuits must employ balanced, shielded, twisted pairs, and must be completely isolated from other electrical circuits. Each shield must be grounded at both ends and at all convenient intermediate points. Each EED case must be bonded to the vehicle structure.

d. Interfaces - Signal circuits whose source and load are referenced to a particular central ground point will be designed to maintain audio-frequency (0 to 150 kHz) isolation between the other Flight Capsule central ground points at all except operating frequencies.

e. Power Distribution - All electrical power must be distributed via a twisted pair made up of the "hot" and return lines and must contain no ground loops.

f. OSE/ESP Interface - All OSE power supplies feeding the ESP must be grounded at the ESP CGP. All OSE operational, calibration, and test circuitry associated with the ESP, (except RF circuits), must be isolated from the OSE facility ground system and terminated at the ESP CGP.

g. Electrical Bonding - Both the primary structure and the secondary structural elements which support electrical and electronic equipment will be electrically bonded. Electronic/electrical equipment must be bonded directly to the structure through metal-to-metal contact or bonding straps.

h. Circuit Classifications - Each wire must be assigned a classification code or category which is based on the characteristics of the signal it carries. To ensure against interference by wire-to-wire coupling, wires must be routed only in cables made up of like or compatible categories.

SECTION 6

EXPERIMENT, EQUIPMENT, AND SUBSYSTEMS CONSTRAINTS

These constraints are grouped in accordance with their effect on descent imaging, atmospheric properties determination and supporting subsystems.

6.1 DESCENT IMAGING - The imaging equipment, supporting subsystems, and Capsule Bus accommodations must be designed taking into account the constraints on imaging which do or would arise from the following.

a. Effect of attitude rates
b. Effect of any ablative vapor condensation on window
c. Effect of shock optics
d. Effect of gaseous flow field luminosity
e. Blackout period image loss
f. Contrast attenuation from Mars atmosphere
g. Effect of radio subsystem bandwidth limits
h. Effect of parachute/aeroshell separation/terminal thruster ignition altitude/velocity sequence
i. Thruster plumes
j. Effect of lateral winds on V/H ratio and consequently on imaging scene overlap for area identification continuity and stero imaging
k. Weight
l. Mounting provisions for camera ejection before lander touchdown.

6.2 ATMOSPHERIC PROPERTIES DETERMINATION - Instrumentation equipment for atmospheric properties determination, the supporting subsystems, and Capsule Bus accommodations must be designed taking into account the constraints on accuracy and measurement range which do or would arise from the following.

a. Effect of spacecraft orientation and deorbit attitude and impulse on entry trajectory
b. Effect of entry conditions and altitude measurement uncertainties
c. Effect of any mixing of ablative products with atmospheric samples
d. Effect of angle of attack oscillations on acceleration corrections and $C_L(x)$ uncertainties
e. Angle of attack error effect on stagnation pressure and temperature probes calibration

f. Effect of angle of attack errors relative to flight path

g. Effect of attitude rates on temperature and pressure measurements

h. Effect of capsule C_D and C_L (α) and parachute C_D uncertainties

i. Altitude/velocity for parachute deployment, aeroshell separation, and terminal thruster ignition

j. Effect of vehicle base pressure and temperature uncertainties

k. Time/space variation in the atmosphere and local surface elevation differences

l. Effect of mass and c.g. uncertainties

m. Structural vibration effects on accelerometer

n. Instrumentation state of art.

6.3 SUPPORTING SUBSYSTEMS - Equipment designs for the supporting subsystems are influenced by the constraints which result from the following.

a. Telecommunications
   - o Effect of atmospheric plasma blackout
   - o Multipath transmission effects
   - o Effect of doppler shift during entry
   - o Entry geometry effects
   - o Onboard prestored sequences

b. Electrical Power
   - o Heat sterilization effect on batteries
   - o Long term storage effect on batteries
   - o Minimum functional interface with the capsule bus and spacecraft
   - o No single point failure mode in the power subsystem

c. Thermal Control
   - o Limited weight for insulation and electric power for heaters
   - o Effect of atmospheric heating
   - o Effect of cold cruise environment

d. Packaging, Structural, and Cabling
   - o Electromagnetic compatibility on cabling
   - o Provide a simple structural field point and an electrical connector interface
   - o Maximum degree of independence from surface laboratory and capsule bus
   - o Accommodation to future mission requirements, including possible deletion.

SECTION 1

CONFIGURATION

A general description of the preferred design configuration of the Entry Science Package is given below. Figure 1-1 shows the installation of the major ESP components. These include the ESP principal unit, the stagnation point instrument head, the entry TV camera unit, and the accelerometer unit.

1.1 ESP INSTALLATION - The ESP is comprised of imaging equipment, atmospheric properties determination equipment, and science instrument support equipment which may be classed according to physical location as follows:

a. Sensors and UHF antenna having specific location constraints and installed in several locations on the Capsule Bus,

b. Mass spectrometer and ESP supporting subsystems without specific location constraints and integrally housed in a single container on top of the Capsule Lander base platform, and

c. Flight Spacecraft mounted equipment.

1.1.1 Location Constrained Equipment - The location constrained equipment includes the entry TV cameras, total temperature and base temperature sensors, total pressure and base pressure transducers, mass spectrometer sample gathering port, accelerometer, and UHF antenna.

The entry TV cameras are mounted in a container installed below the lower surface of the Capsule Lander foot pad. The fields of view are aligned parallel to the roll axis and directed through an optical window in the Aeroshell.

The ports for sensing total temperature and pressure and for procuring gas samples for the mass spectrometer are contained in a single instrument head installed in the Aeroshell nose cap at the stagnation point. The insulated total pressure transducer is clamped directly to the instrument head.

The base pressure transducer is mounted on a bracket which supports the UHF antenna above the ESP equipment container. The base temperature sensor is installed in the base pressure sampling tube.

The three axis accelerometer is installed in the hub fitting of the Capsule Lander base platform.

1.1.2 Non-Location Constrained Equipment - The non-location constrained equipment includes the mass spectrometer, telemetry equipment, instrumentation equipment, high rate radio subsystem, data storage subsystem, battery, battery charger, and power

PART D

SELECTED DESIGN CONCEPT

This part describes first the general arrangement for the entry science package in the Capsule Bus. Next are the entry science and supporting subsystem descriptions, which can be supplemented by reading the functional descriptions of Part G, as well as the further discussions of Part E. The mututal accommodations required between the entry science package and the Capsule Bus is discussed, and followed by sections on mission operations and implementation.

ENTRY SCIENCE PACKAGE INSTALLATION

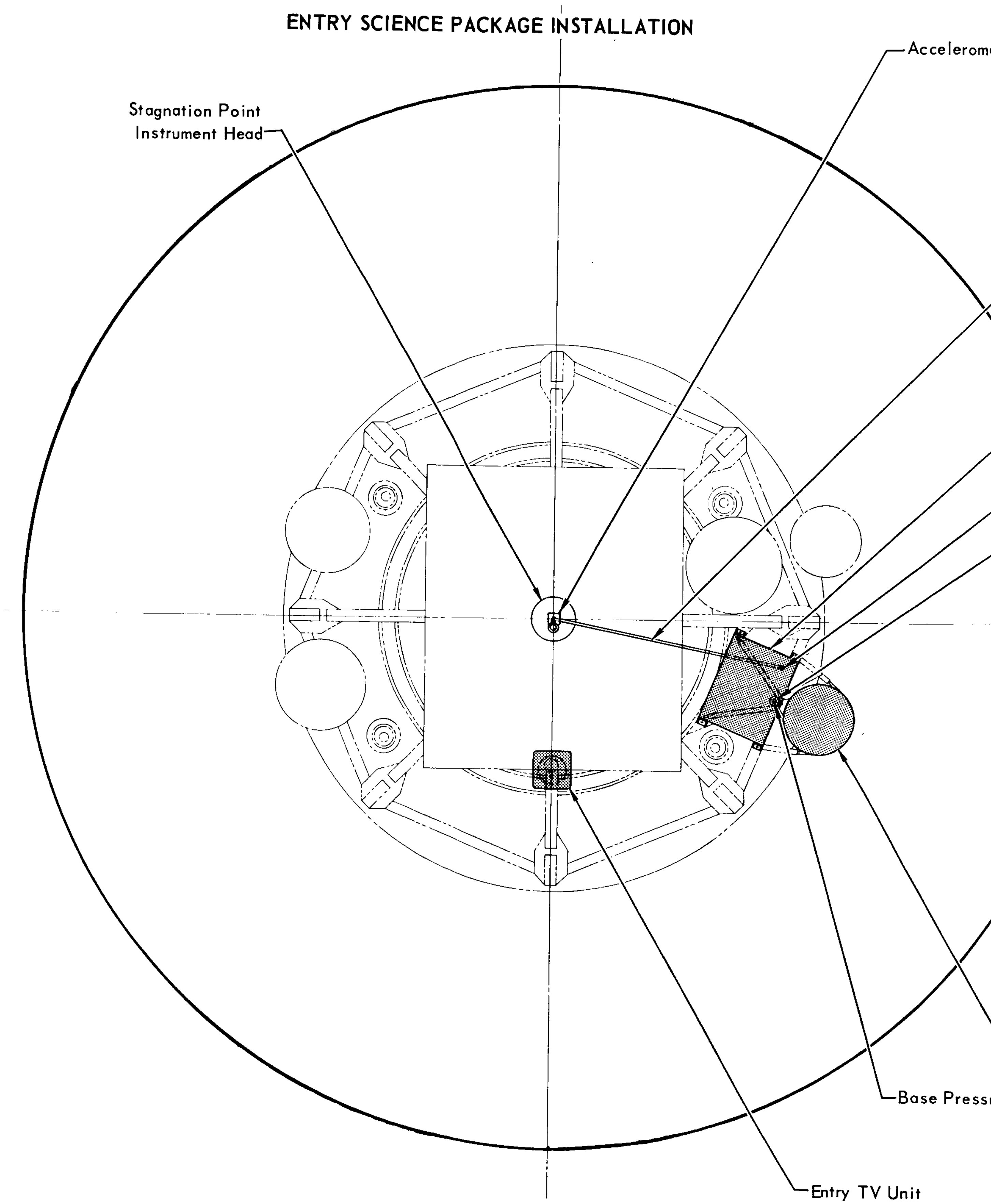
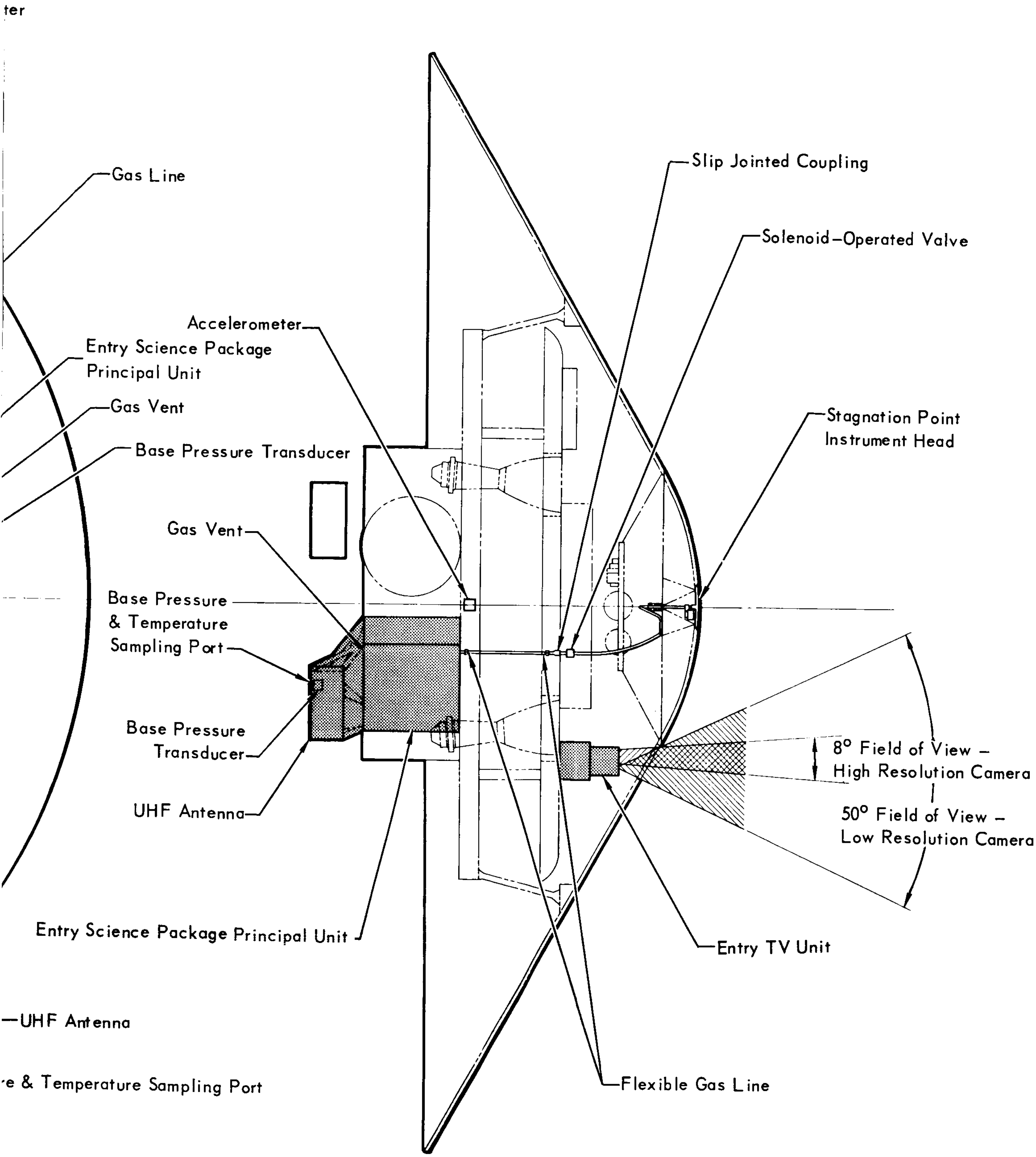


Figure 1-1



ter
Gas Line
Accelerometer
Entry Science Package Principal Unit
Gas Vent
Base Pressure Transducer
Gas Vent
Base Pressure & Temperature Sampling Port
Base Pressure Transducer
UHF Antenna
Entry Science Package Principal Unit
—UHF Antenna
·e & Temperature Sampling Port
Slip Jointed Coupling
Solenoid-Operated Valve
Stagnation Point Instrument Head
8° Field of View – High Resolution Camera
50° Field of View – Low Resolution Camera
Entry TV Unit
Flexible Gas Line

switching and logic. These components are housed in the ESP equipment container which is attached by four bolts to the top of the Capsule Lander base platform. See Figure 1.1-1.

1.1.3 Flight Spacecraft Mounted Equipment - The Flight Spacecraft mounted equipment includes a high-rate antenna subsystem, high-rate radio subsystem and high-rate data storage subsystem. This equipment provides the Flight Spacecraft portion of the ESP communication link and shall be discussed in further detail in Paragraph 1.2.3.

1.2 ESP EQUIPMENT - The ESP equipment may be categorized according to function as follows:

a. Entry Television (TV),

b. Atmospheric Properties Determination, and

c. Science Instrument Support Subsystems.

The major equipment items in the ESP are tabulated in Figure 1.2-1.

1.2.1 Entry TV - The entry TV is comprised of two cameras housed in a single container. Each camera contains a vidicon and lens with associated exposure components. One camera provides low resolution imaging with a 50° field of view while the other camera provides high resolution imaging with an 8° field of view. The container is mounted below the bottom of the Capsule Lander foot pad by means of a pyrotechnic thruster. The container is aligned tc locate the cameras' fields of view parallel to the roll axis by means of three adjustable compression struts as shown in Figure 1.2-2.

A fused silica, optical window maintains structural integrity in the Aeroshell while providing a clear field of view for the cameras during entry. The inside surface of the window is protected by a flexible boot which spans the gap between the camera lenses and the window. A protective cover fastened to the Sterilization Canister prevents contamination of the outside surface of the window during sterilization and interplanetary cruise.

The cameras operate continuously throughout the entry phase and during the descent phase until the Capsule Lander is within approximately 90 feet of the Martian surface. The pyrotechnic thruster is then initiated. At the same time a pyrotechnic disconnect separates the electrical wire bundle allowing the thruster to propel the camera container away from the Capsule Lander.

1.2.2 Atmospheric Properties Determination - The equipment components associated with atmospheric properties determination are the total pressure transducer, total temperature sensor, base pressure transducer, base temperature sensor, mass spectrometer and accelerometer.

ENTRY SCIENCE PACKAGE PRINCIPAL UNIT

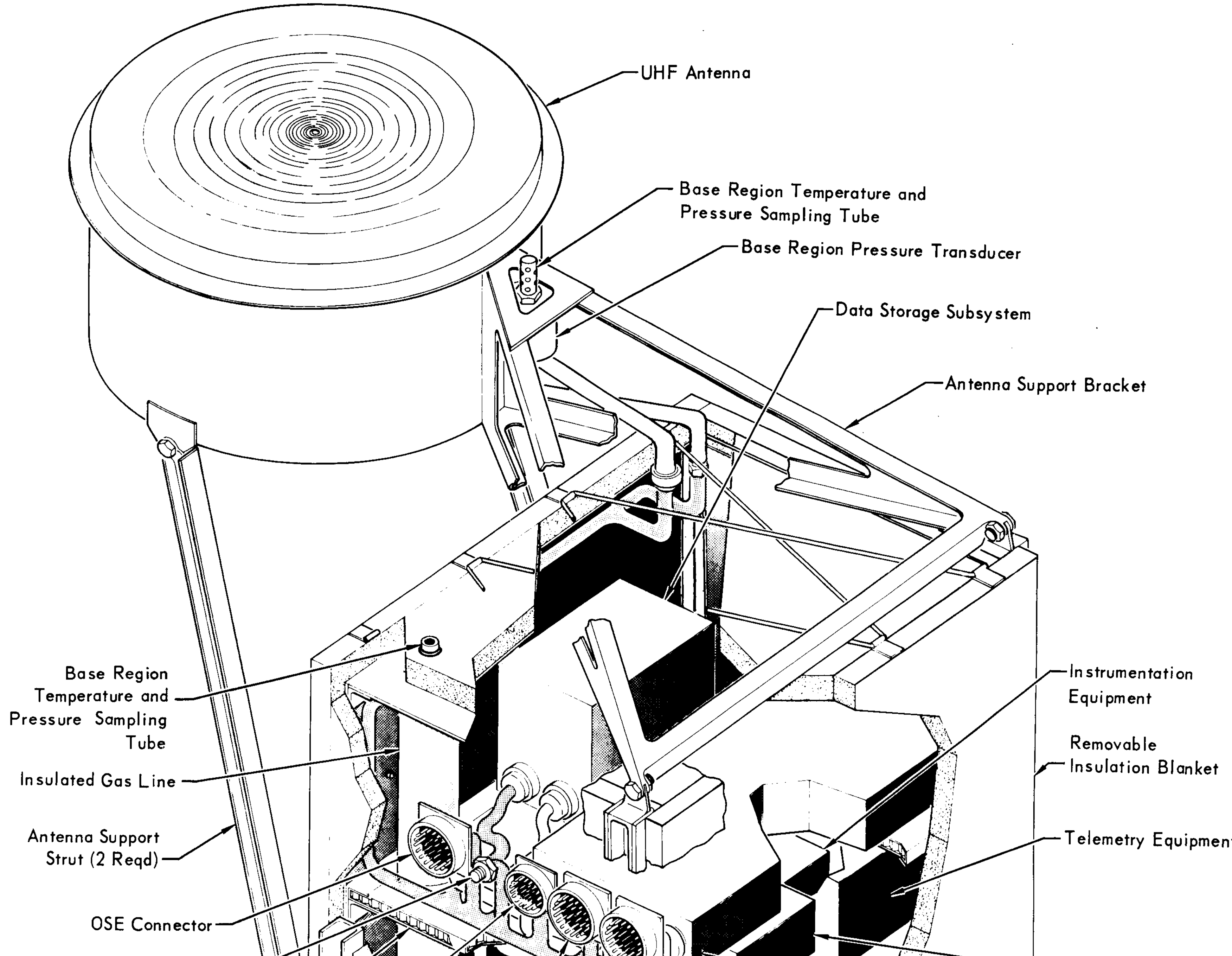
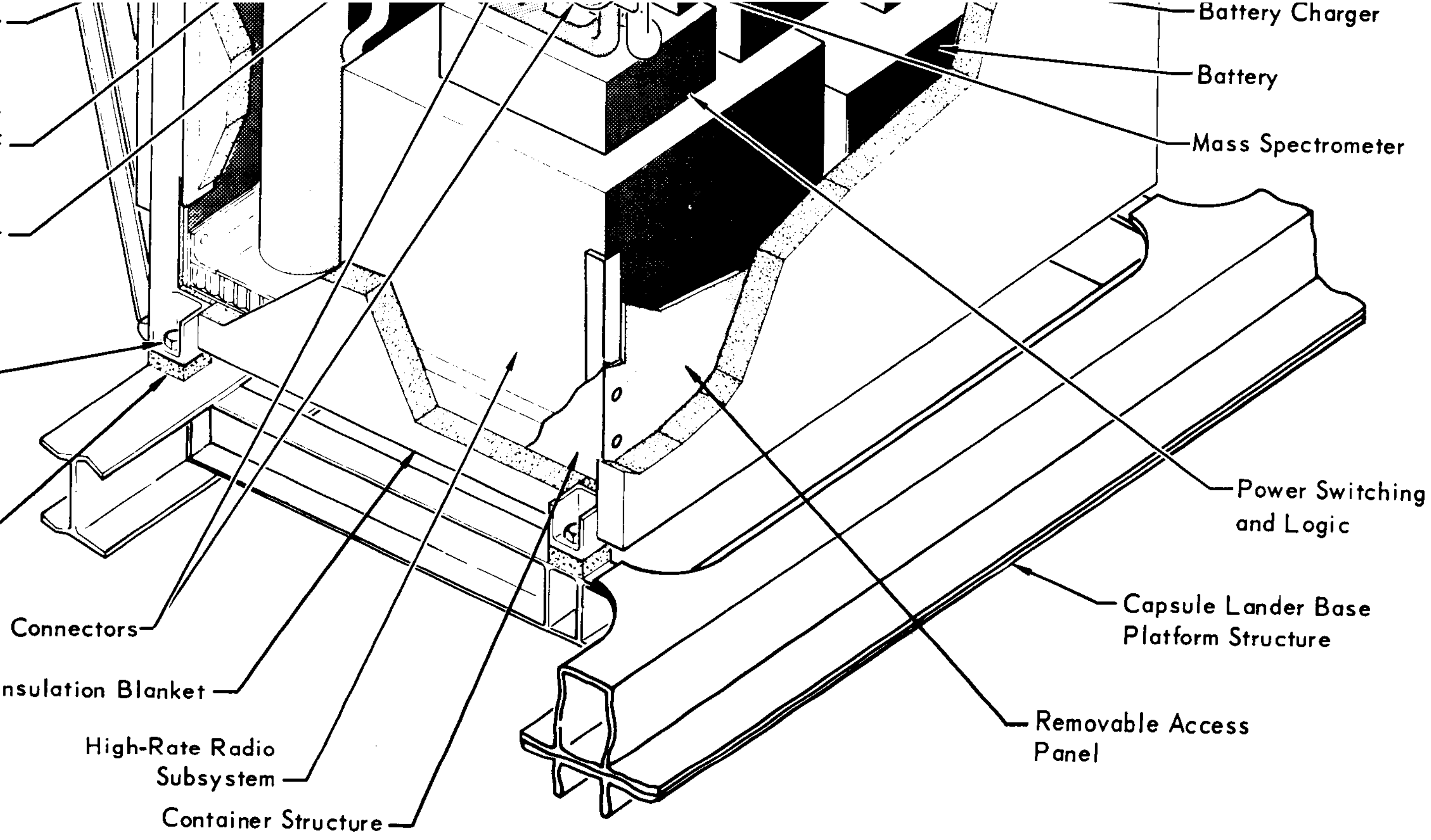


Figure 1.1-1


OSE Coaxial Connector
Honeycomb Equipment Shelf
Entry TV Connector
Container Mounting Bolt (4 Reqd)
Structural Insulation Block
CB/ESP Connectors
Fixed Insulation Blanket
High-Rate Radio Subsystem
Container Structure
Battery Charger
Battery
Mass Spectrometer
Power Switching and Logic
Capsule Lander Base Platform Structure
Removable Access Panel

ESP EQUIPMENT LIST

ESP EQUIPMENT CATEGORIES	MAJOR EQUIPMENT ITEMS	FUNCTION	INSTALLATION PARAMETER
Entry Television	Television Cameras	Visual imaging of Martian surface during entry and descent.	Optical axis parallel to Capsule Bus roll axis. Close to Aeroshell to minimize size of optical window.
Atmospheric Properties Determination	Total Pressure Transducer	Sense gas pressure in the stagnation region during entry and descent.	Close to sampling port to minimize lag time. Sampling port in instrument head at stagnation point.
	Base Pressure Transducer	Sense gas pressure in the base region during entry and descent.	Close to sampling port to minimize lag time. Sampling port in base region.
	Total Temperature Sensor	Sense gas temperature in the stagnation region during descent.	In instrument head at stagnation point.
	Base Temperature Sensor	Sense gas temperature in the base region during entry and descent.	In base region.
	Mass Spectrometer	Analyze gas samples in the Martian atmosphere during descent.	In ESP principal unit with remote sampling part free from contamination by ablator outgassing
	Accelerometer	Measure deceleration during entry to obtain data on atmospheric density.	In front of and in close proximity to C.G. of entry configuration.
	Telemetry Equipment	Monitors ESP status during interplanetary cruise and feeds this data through hardline to the flight spacecraft. Arranges and formats ESP science and engineering data during entry and descent and feeds data to high-rate radio subsystem.	Installed in ESP Principal Unit
	Instrumentation Equipment	Processes engineering data signals and feeds them to the telemetry equipment.	Installed in ESP Principal Unit
	High-Rate Radio Subsystem	Receives ESP science and engineering data from telemetry subsystem and transmits this data to orbiting Flight Spacecraft	In proximity of UHF antenna to minimize line loss.

Figure 1.2-1

Science Instrument Support Subsystems	Data Storage Subsystem	Delays ESP Non-video science data and ESP engineering data by 50 seconds and 150 seconds. Interleaves real-time and 2 delayed data streams for transmission via ESP radio link.	Installed in ESP Principal Unit
	UHF Antenna	Radiates entry and descent science data in direction of orbiting Flight Spacecraft	120° unobstructed field of view with centerline parallel to Capsule Bus roll axis.
	Battery	Furnishes electrical power to ESP equipment during entry and descent and during short periods of interplanetary cruise when Flight Spacecraft power is not available.	Installed in ESP Principal Unit
	Battery Charger	Maintains full charge on ESP battery through Flight Spacecraft power source.	Installed in ESP Principal Unit
	Power Switching and Logic	Turns ESP on and off. Transfers ESP to internal power when flight spacecraft power is not available. Senses ESP battery failure.	Installed in ESP Principal Unit
	High-Rate Antenna Subsystem (In Flight Spacecraft)	Receives signals from ESP UHF antenna and delivers them to Flight Spacecraft mounted high-rate radio subsystem.	Installed in Flight Spacecraft with field of view directed toward Capsule Bus
	High-Rate Radio Subsystem (In Flight Spacecraft)	Demodulates signals from high-rate antenna subsystem and feeds the reconstructed bit stream and synchronization pulses to the Flight Spacecraft mounted Capsule Bus data distribution unit.	Installed in Flight Spacecraft.
	High-Rate Data Storage Subsystem (In Flight Spacecraft)	Stores ESP data receives from flight spacecraft mounted capsule bus data distribution unit at the received bit rate and plays back data upon command at the Flight Spacecraft transmitter bit rate.	Installed in Flight Spacecraft.

ENTRY TV CAMERA STRUCTURAL/MECHANICAL INSTALLATION

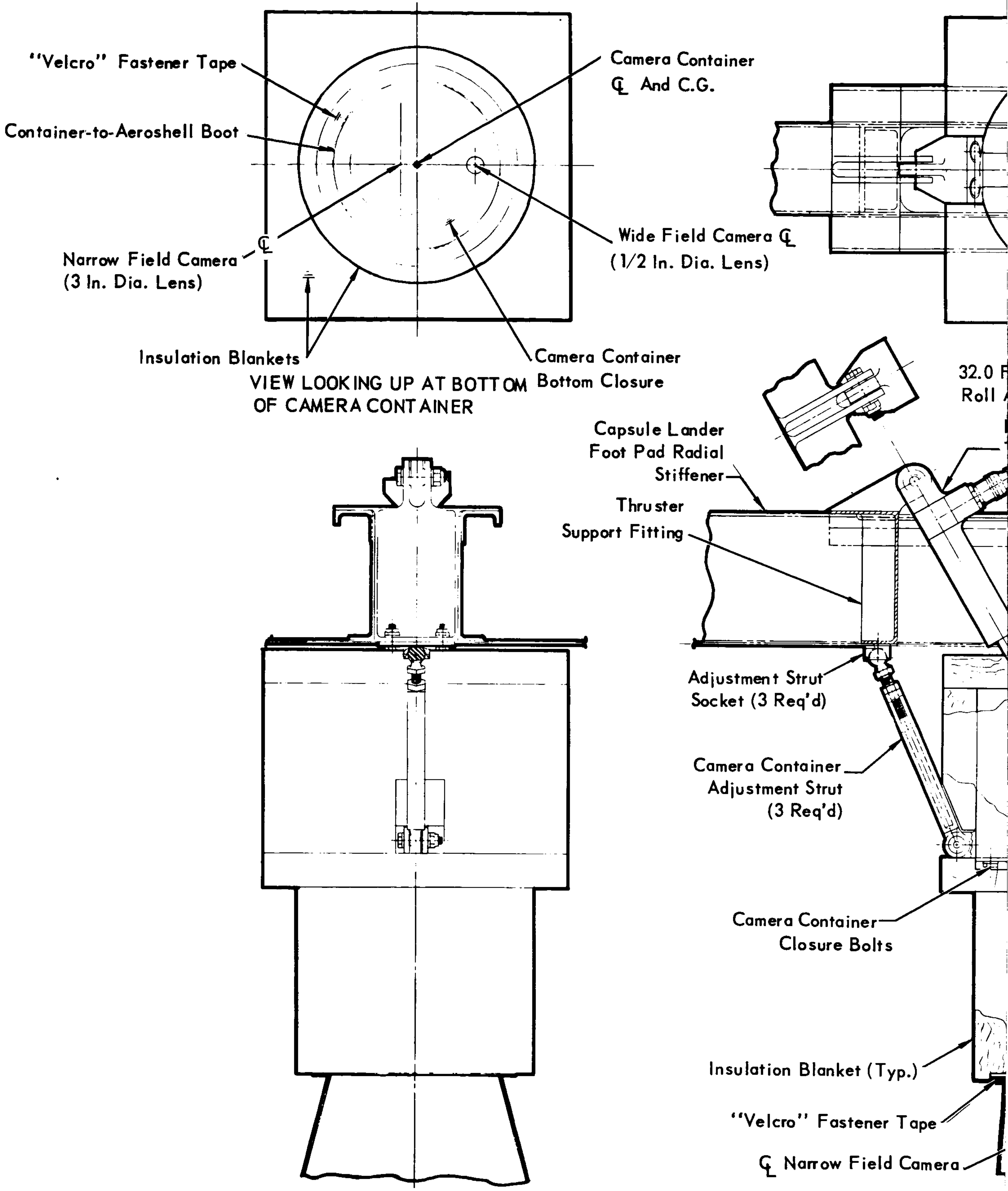
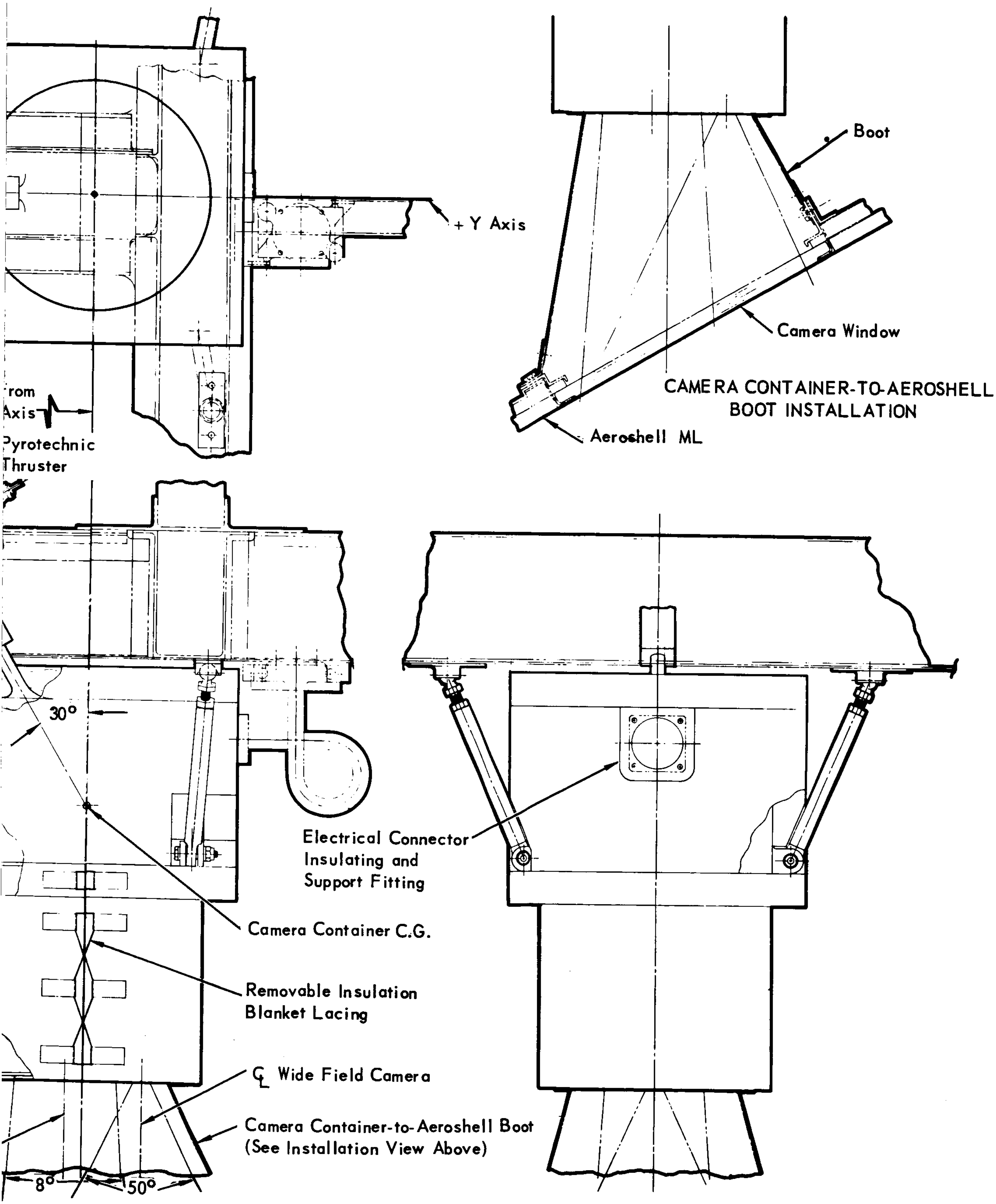


Figure 1.2-2


+ Y Axis
Boot
Camera Window
CAMERA CONTAINER-TO-AEROSHELL
BOOT INSTALLATION
Aeroshell ML
Pyrotechnic
Thruster
30°
Electrical Connector
Insulating and
Support Fitting
Camera Container C.G.
Removable Insulation
Blanket Lacing
℄ Wide Field Camera
Camera Container-to-Aeroshell Boot
(See Installation View Above)
8°
50°

A beryllium instrument head is installed in the center of the Aeroshell nose cap to obtain gas samples for the mass spectrometer and to sample the temperature and pressure of the gas in the stagnation region. See Figure 1.2-3. The inlets are provided by a center hole and concentric annular slot. The total temperature sensor is installed in the center hole, with an outlet bleed vented to the capsule base region. The annular slot provides the source for the gas pressure samples. A rigid tube connected to the top of the instrument head carries the bleed gas to the base region and makes gas samples available through a molecular leak to the mass spectrometer located in the ESP equipment container. A 1/4-inch port connects the outer sampling slot to the total pressure transducer, which is clamped to the instrument head.

A solenoid actuated valve mounted on the vent tube below the Capsule Lander foot pad prevents gas flow through the tube until after peak aerodynamic heating. The tube is routed through the foot pad via a slip jointed coupling. At Aeroshell separation, the portion of the tube below the foot pad stays with the Aeroshell and the open coupling on the bottom of the foot pad serves as the gas sampling port during the terminal descent phase. The wire bundle providing the electrical interface between the stagnation point instrument head and the ESP equipment container is disconnected by a pyrotechnic disconnect on the bottom of the foot pad at the time of Aeroshell separation.

The base pressure transducer is mounted on a bracket which supports the UHF antenna above the ESP equipment container as shown in Figure 1.1-1. The base temperature sensor is installed in the base pressure sampling tube, which protrudes through the thermal curtain into the base pressure region.

Sensing the deceleration of the Capsule during entry is essential in determining the Martian atmospheric density profile. To accomplish this, a three axis, dual range accelerometer is installed in the hub fitting of the Capsule Lander base platform as shown in Figure 1.2-4. This installation provides a location as close as possible to the c.g. of the entry configuration without violating the envelope of the Surface Laboratory.

1.2.3 Science Instrument Support Subsystems - All of the equipment in the ESP which is not directly associated with entry science is identifiable as "scinece instrument support equipment." This includes the instrumentation equipment, telemetry equipment, high rate radio subsystem, data storage subsystem, battery, battery charger, and power switching and logic within the ESP equipment container and the UHF antenna which is mounted above the container. This container with all equipment including the antenna installed comprises the ESP principal unit. See Figure 1.1-1. Also included

RADAR ALTIMETER CONICAL MONOPOLE ANTENNA
AND
ENTRY SCIENCE STAGNATION POINT INSTRUMENT HEAD

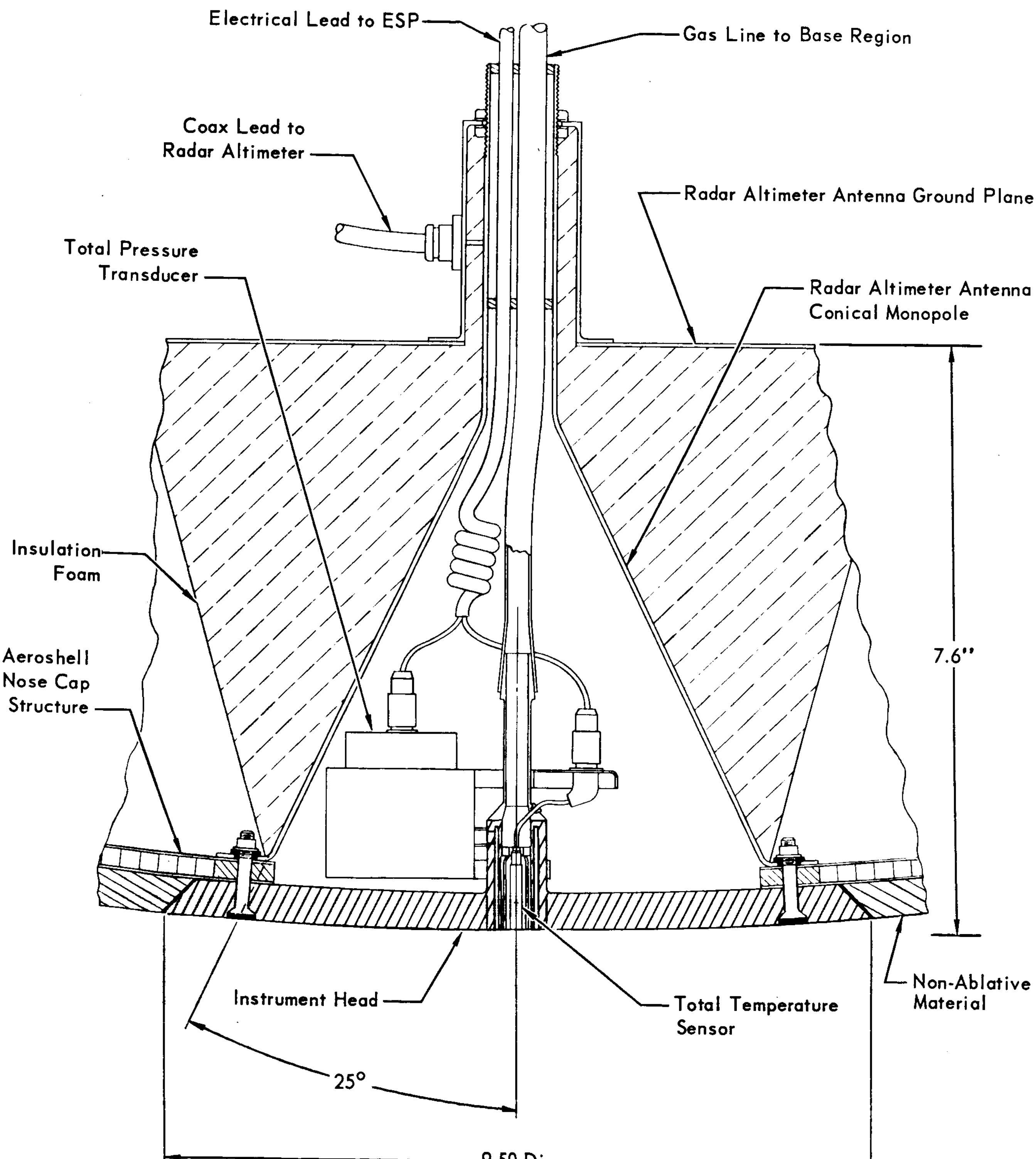


Figure 1.2-3

ESP ACCELEROMETER INSTALLATION

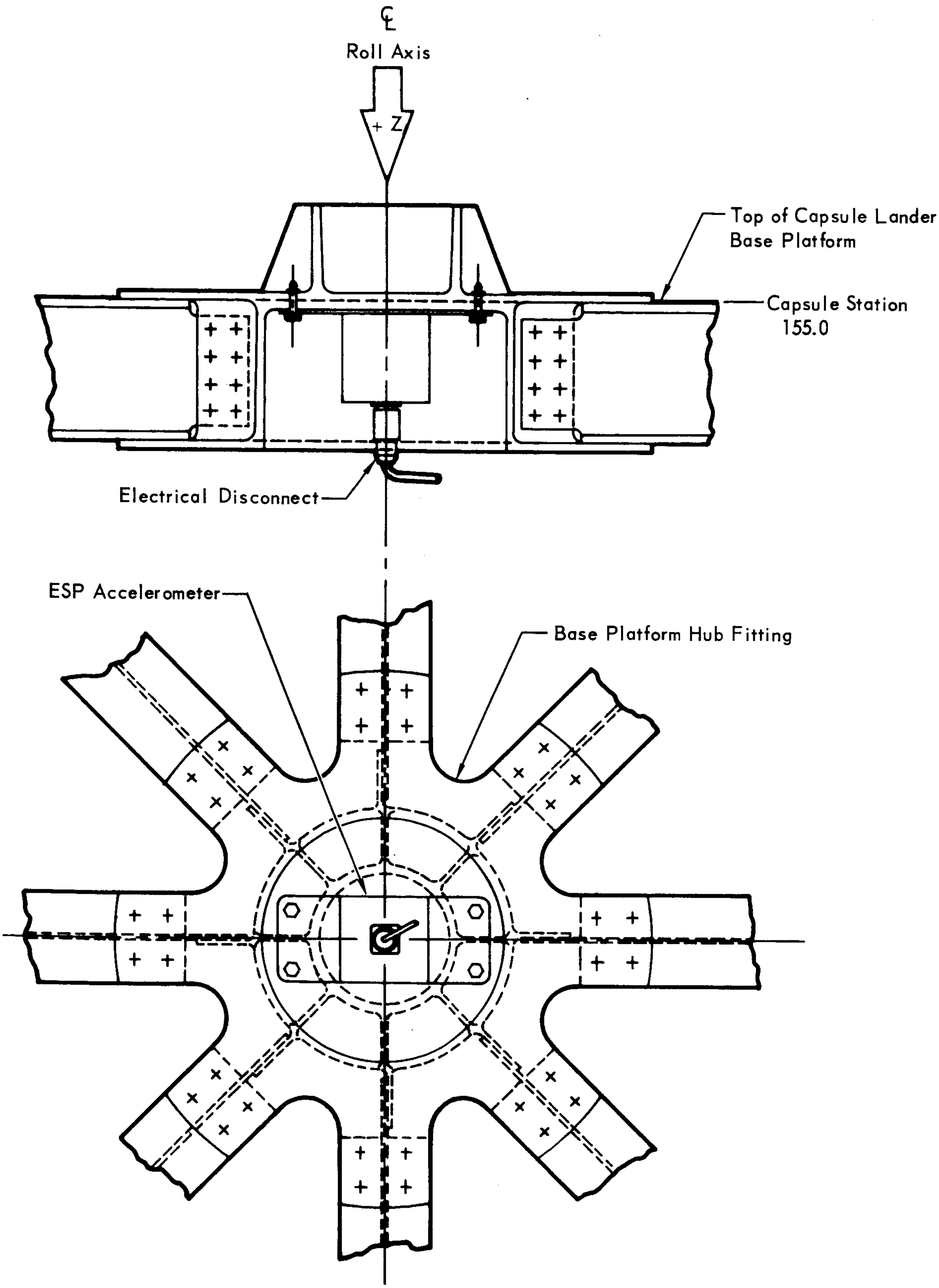


Figure 1.2-4

in the science instrument support subsystem is the Flight Spacecraft mounted ESP equipment, which is a portion of the ESP communication link.

The instrumentation equipment, which includes a signal processing unit and power supply, processes engineering data signals and delivers them to the telemetry equipment. The instrumentation equipment is mounted on the lower shelf of the ESP equipment container.

The telemetry equipment arranges and formats ESP science and engineering data during the entry and descent phases and feeds these data to the radio subsystem. During interplanetary cruise, the telemetry equipment monitors ESP status and delivers these data through the Capsule Bus interface to the transmitter in the Flight Spacecraft. The telemetry equipment is mounted on the upper shelf of the ESP equipment container.

The high-rate radio subsystem receives ESP science and engineering data from the telemetry equipment during the entry and descent phases and transmits them to the orbiting Flight Spacecraft. The high-rate radio subsystem is mounted adjacent to the telemetry equipment on the upper shelf of the ESP equipment container.

The data storage subsystem, also mounted on the upper shelf of the container, retains ESP science and engineering data (except for entry TV data) during the communication black-out period, and then delivers these data to the radio subsystem after termination of the black-out period.

The battery, battery charger, and power switching and logic boxes make up the ESP power subsystem. These are installed in the lower section of the container; the battery being mounted on the lower shelf and the battery charger and the power switching and logic mounted to the bottom side of the upper shelf.

The manually activated, 320 watt-hour battery furnishes electrical power for operating the ESP equipment during entry and descent. The battery is also required to power the telemetry equipment and ESP heaters for short periods during interplanetary cruise when Flight Spacecraft power is not available.

The battery charger maintains a full charge on the ESP battery utilizing power from the Flight Spacecraft bus through the Capsule Bus interface. The power switching and logic turns the ESP on and off, transfers the ESP to internal power when Flight Spacecraft power is not available, and senses ESP battery failure. It receives its command signals through the Capsule Bus interface.

The UHF antenna is a cavity-backed spiral, 15.0 inches in diameter and 7.0 inches deep. It is supported above and outboard of the ESP equipment container by a bracket and two struts which attach to the container. This location provides the antenna with a 120° field of view and places the antenna aperture through the thermal curtain. The mounting of the antenna provides for alignment parallel to the roll axis within 1.0°.

The equipment mounted in the Flight Spacecraft includes a high-rate antenna subsystem, high-rate radio subsystem and high-rate data storage subsystem.

The high-rate antenna subsystem is comprised of an axial-mode, helix antenna mounted on a boom which is perpendicular to the orbital plane. See Figure 1.2-5. The boom is rotatable to properly position the antenna with respect to the Capsule Bus. The function of the high rate antenna subsystem is to receive signals from the ESP UHF antenna and deliver them to the spacecraft-mounted high-rate radio subsystem.

The high-rate radio subsystem is comprised of two receivers, a diversity combiner, and a bit synchronizer. Its function is to demodulate the signals from the high-rate antenna subsystem and deliver the reconstructed bit stream and synchronization pulses to the spacecraft-mounted, Capsule Bus data distribution unit.

The high-rate data storage subsystem is a combination tape recorder and data rate converter. Its function is to store ESP data received from the spacecraft-mounted, Capsule Bus data distribution unit at the received bit rate, and play back data upon command at the spacecraft transmitter bit rate.

1.3 INTERFACES - The ESP mechanical interfaces with the Capsule Bus and Flight Spacecraft are described in Figure 1.3-1.

1.4 WEIGHT DATA - Nominal weight, weight uncertainties, and substantiating data are presented in the following sections. The nominal weights are presented in Section 1.4.1 reflecting designs which meet the system requirements specified in this report. Empirically derived contingencies are included in the predicted nominal weights to reflect the preliminary status of the design. Statistical variations in estimation techniques, material properties, and contingencies assigned were considered in the uncertainty analysis (Section 1.4.2).

As long as requirements and criteria are not changed, the weight can be expected to fall within the limits specified. Specific allowances for changes in system requirements (growth) are not included.

1.4.1 Entry Science Package Weight Summary - Group weight summaries are presented in Figure 1.4-1 and detail weight summaries in Figure 1.4-2 for capsule bus mounted equipment. Spacecraft mounted equipment, part of the telecommunication subsystem, is as follows:

ENTRY SCIENCE PACKAGE RECEIVING ANTENNA LOCATION

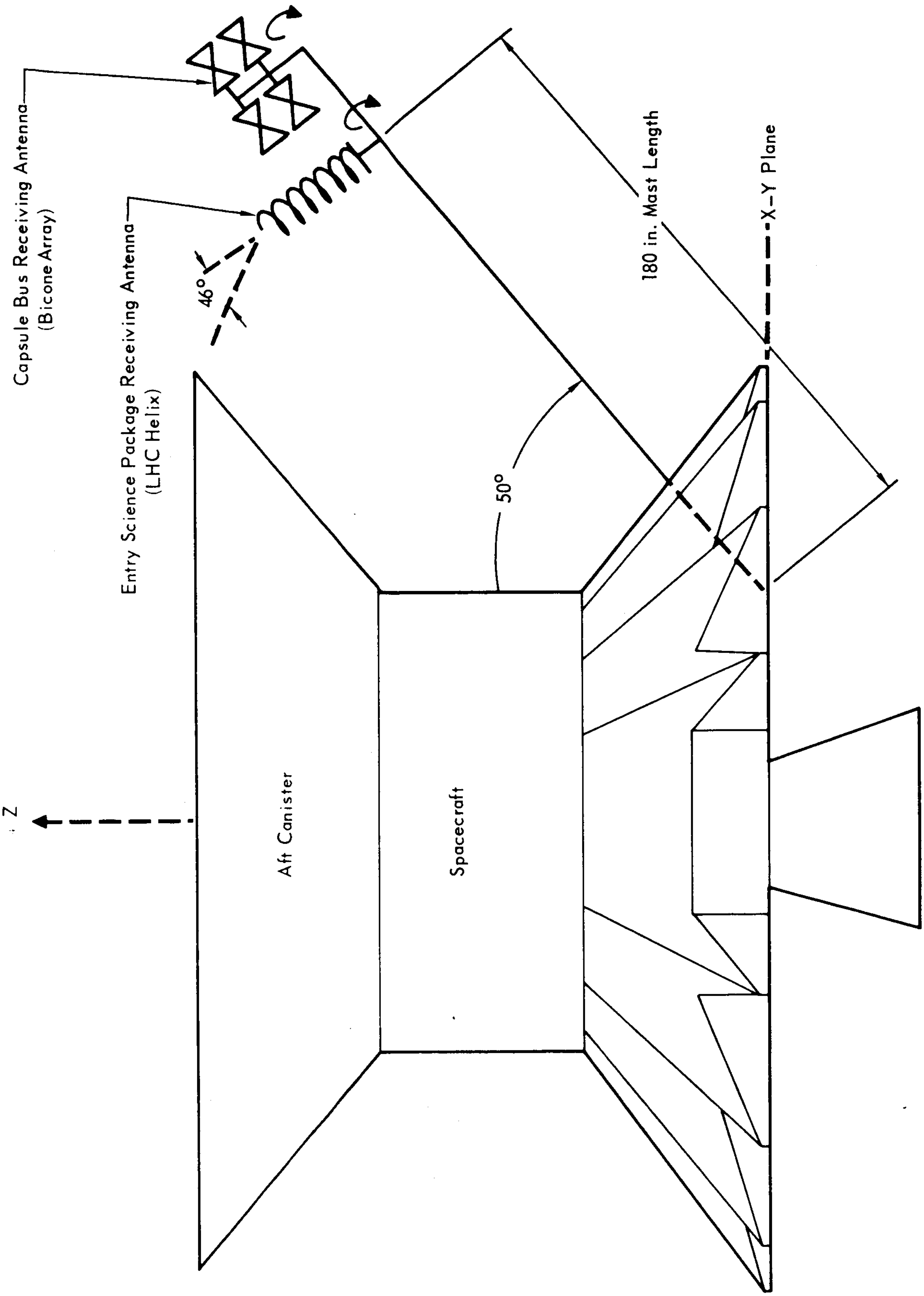


Figure 1.2-5

ESP MECHANICAL INTERFACES

INTERFACE	COMPONENT INVOLVED	DESCRIPTION
ESP/CB	ESP Principal Unit	4-Bolt attachment to top of Capsule Landed base platform.
ESP/CB	Stagnation point instrument head.	12-Bolt attachment through Aeroshell nose cap structure and conical monopole section of radar altimeter antenna.
ESP/CB	Entry TV Unit	Attached by means of a pyrotechnic ejector and 3 compression struts to Capsule Lander impact foot pad. Flexible cover for lens and window protection joins Entry TV Unit and Aeroshell.
ESP/CB	Accelerometer	4-Bolt attachment to inside of hub fitting in center of Capsule Lander base platform.
ESP/CB	Gas sampling tube	Routed from stagnation point instrument head to ESP container; clamped to Aeroshell, impact foot pad and base platform.
ESP/FSC	High-rate antenna subsystem	Mounted on boom protruding from Flight Spacecraft.
ESP/FSC	High-rate radio subsystem	Mounted in Flight Spacecraft.
ESP/FSC	High-rate data storage subsystem	Mounted in Flight Spacecraft.

Figure 1.3-1

ENTRY SCIENCE PACKAGE GROUP WEIGHT SUMMARY
(CAPSULE BUS MOUNTED EQUIPMENT)

	BEFORE AEROSHELL SEPARATION	AFTER AEROSHELL SEPARATION
Structure	14.3	14.3
Thermal Control	5.0	5.0
Telecommunications	55.0	55.0
Electrical Power	22.5	22.5
Experiments	27.0	25.5
Wiring and Mounting Provisions	56.8	55.7
Total Entry Package Weight	180.6	178.0

Figure 1.4-1

ENTRY SCIENCE PACKAGE DETAIL WEIGHT SUMMARY
(CAPSULE BUS MOUNTED EQUIPMENT)

	BEFORE AEROSHELL SEPARATION	AFTER AEROSHELL SEPARATION
Structure (Box Assembly)	14.3	14.3
Thermal Control (Insulation & Heater)	5.0	5.0
Tele-communications	(55.0)	(55.0)
Radio Subsystem	26.0	26.0
Antenna Subsystem	6.0	6.0
Data Storage Subsystem	7.5	7.5
Telemetry Subsystem		
Telemetry Equipment	9.0	9.0
Instrumentation Equipment	6.5	6.5
Electrical Power	(22.5)	(22.5)
Battery	15.0	15.0
Battery Charger	1.5	1.5
Power Switching and Logic Unit	6.0	6.0
Experiments – Entry and Descent	(27.0)	(25.5)
Entry TV	14.0	14.0
Atmospheric Profile		
Accelerometers	2.0	2.0
Pressure Transducers	2.0	1.0
Temperature Probes	1.0	.5
Mass Spectrometer	8.0	8.0
Wiring and Connectors	(36.9)	(36.0)
Tele-communications	22.0	22.0
Electrical Power		
Battery and Charger	.8	.8
Distribution	6.0	6.0
Experiments	8.1	7.2
Mounting Provisions	(19.9)	(19.7)
Shelf Mounted Items	18.0	18.0
Individually Mounted Items	1.9	1.7
Total ESP Weight	180.6	178.0

Figure 1.4-2

UHF Antenna	1.0 lbs.
Radio Subsystem	9.0 lbs.
Data Storage Subsystem	8.0 lbs.
Directional Coupler and Attenuator	1.3 lbs.

The weights shown are predicted nominal values. Contingencies are included, but weight uncertainties are considered separately in Section 1.4.2. The nominal weight of items which are included in the preferred design to improve mission success and provide standardization is presented in Figure 1.4-3.

1.4.2 Uncertainty Analysis - A weight uncertainty analysis was conducted to define, for the present design detail, a plus and minus tolerance on the nominal weight (or estimated weight plus contingency) for the entry science package. The actual measured weight at completion of design and manufacture would be expected to fall within this tolerance band a specified percentage of the time. Uncertainty decreases with the progress of the detail design, since greater detail is possible in the analysis and more of the contingency items are specifically analyzed. This is illustrated in Figure 1.4-4.

System weight uncertainty was obtained by assigning each individual component a one sigma (standard deviation) uncertainty value and statistically combining these uncertainties (root-sum square). This method involved the following assumptions:

a. component uncertainties are independent
b. uncertainties have a normal distribution
c. the net weight uncertainty (σw) is

$$\sigma w = \sqrt{\sum_{i=1}^{n}\left(\Delta w_i\right)^2}$$

where ΔW_i = weight uncertainty of ith component.

The individual uncertainties were obtained by one of two methods. For an empirical estimate, the tolerance intervals of the correlated data points were used. For estimates based on preliminary design analysis, values were derived from 130 data points from previous McDonnell programs. The results are shown in Figure 1.4-5.

Entry Science Package uncertainties for launch, entry, and touchdown weights are shown in Figure 1.4-6.

1.4.3 Qualification of Data - The approaches used to predict the nominal weights presented in Section 1.4.1 are discussed in this section. These approaches can be summarized as follows:

a. Weight is estimated, or calculated, based on preliminary design analyses. Empirically derived contingencies are then added, considering the status of

UTILIZATION OF WEIGHT MARGINS

–ITEMS INCLUDED IN THE BASELINE DESIGN–

SYSTEM	Δ WT	ITEM
Entry Science Package	2.1	Total Weight Margin Utilized
	(2.1)	Total Redundancies
Telecommunications	.5	Redundant ESP Cruise Commutator and Switching
	.3	Interleaving of ESP Data on CBS Radio Link
	.1	Redundant Cruise Encoder
Electrical	1.2	ESP Backup Need Relay

Figure 1.4–3

IDEALIZED VARIATION OF PREDICTED MASS PROPERTY WITH DESIGN STATUS

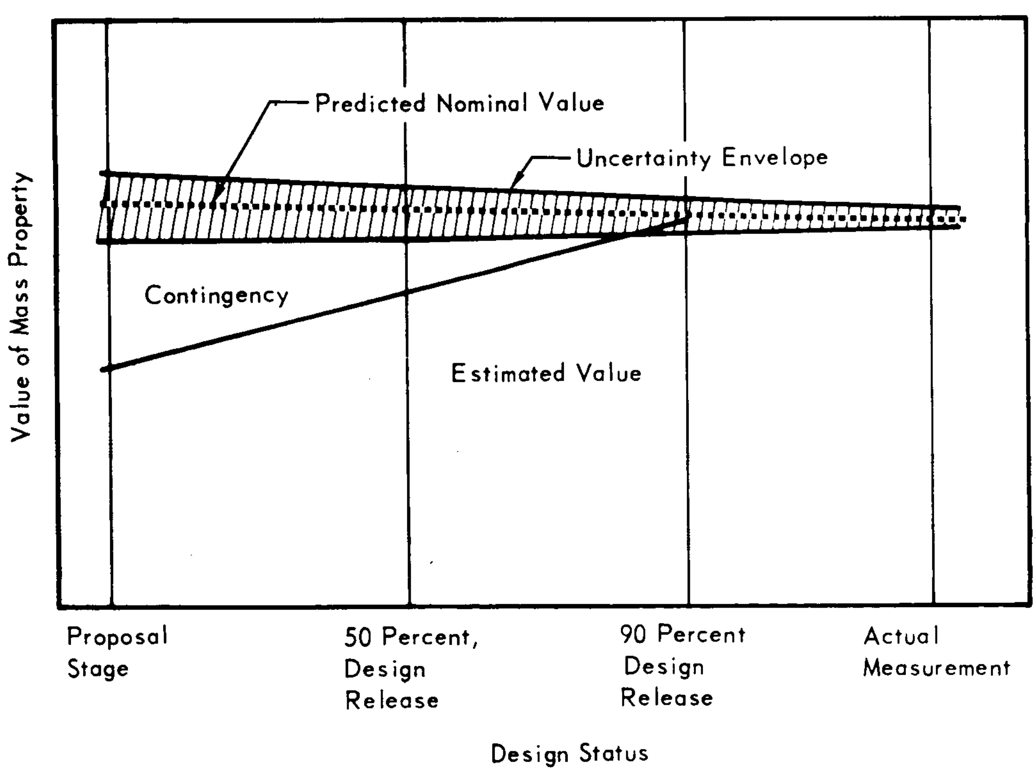


Figure 1.4–4

design at the time of the estimate, i.e., preliminary layout, detail layout, "off-the-shelf", slightly modified, etc. (See Figure 1.4-7).

b. Weight is estimated using semi-empirical weight estimation techniques. Since these techniques include correlations with actual hardware, no additional contingency is added.

c. Weights specified in the 1973 VOYAGER Capsule Systems Constraints and Requirements Document were used. No contingency factors were added.

Weight estimate for a particular subsystem may involve a combination of these approaches.

In this analysis the structure, thermal control, telecommunicating and electrical power weights were derived using Approach "a." The Experiment weights were derived using Approach "c." Wiring and Mounting Provision Weights were derived using Approach "b."

TYPICAL UNCERTAINTIES

Category	NUMBER OF SAMPLES	UNCERTAINTY (1σ)
Structure		
Detailed Layout-Stress Analyzed	42	.18
Preliminary Layout-Little or No Analysis	17	.43
Equipment		
Off the Shelf or Slightly Modified	43	.19
New Development	13	.26
System Installation	15	.45

Figure 1.4-5

MASS PROPERTY UNCERTAINTIES – BASELINE
1 STANDARD DEVIATION Δ'S

CONDITION	ENTRY SCIENCE PACKAGE
Launch	±17.2
Entry	±17.2
Touchdown	±17.2

Figure 1.4-6

TYPICAL CONTINGENCY FACTORS

DESCRIPTION	CONTINGENCY FACTOR
Structure	
Detailed Layout-Stress Analyzed	1.00
Preliminary Layout-Little or No Analysis	1.28
Equipment	
Off the Shelf or Slightly Modified	1.09
New Development	1.33
System Installation	1.32

Figure 1.4-7

SECTION 2

PREFERRED ENTRY EXPERIMENTS

The purpose of the entry experiments is (1) to obtain topographical photographs of the area surrounding the landing site during descent, (2) to determine the atmospheric density and temperature altitude profile, and (3) to measure the atmospheric composition including water vapor. The characteristics of the preferred instruments required to achieve this purpose are presented in Figure 2-1. These instruments are within the constraints listed in the JPL VOYAGER Capsule System, Constraints and Requirements Document, 12 June 1967.

2.1 IMAGING - The preferred entry TV observations utilizes a dual vidicon camera system for accomplishing two basic objectives, landing site identification and detailed surface investigation. The site location can be satisfied by a sequence of overlapping images starting at mid to high altitudes with initial resolution comparable to that obtained from the orbiter (See Figure 2.1-1). Study of surface phenomena is best obtained in the terminal phases of flight when 1-3 foot ground resolution is available. Secondary goals relate to the collection of information on the planetary limb, surface photometry, and atmospheric attenuation.

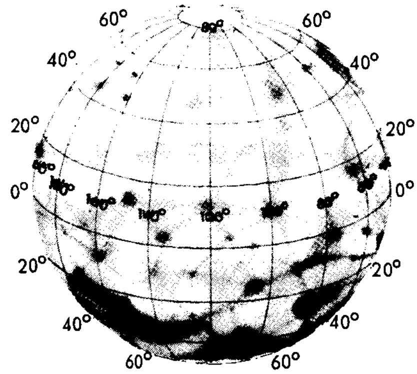
These observations are severely limited by a prolonged period of communications blackout and by the period of high heating. The wide range of atmospheres which can be encountered forces a design configuration which performs well both at high and low altitudes. Good pictures are necessary early in the descent in case blackout blocks information transfer at the lower more favorable altitudes. At these lower altitudes difficulties are experienced in achieving desirable resolution while still being able to observe the landing location. Special provisions have been made to avoid optical degradation from flow field effects induced by peak entry heating. Terminal imaging must contend with the obscurations and oscillations resulting from parachute deployment, Aeroshell separation, and descent engine ignition.

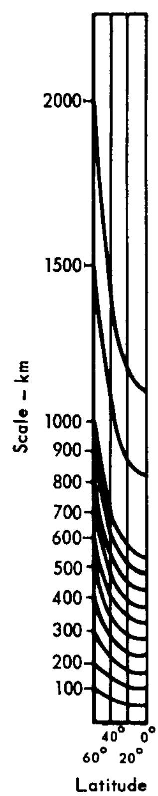
Instrumentation - Imaging is accomplished with a dual vidicon camera installation (see Figure 2.1-2). One camera is equipped with a 3.2 inch lens and provides resolution adequate for both high altitude and terminal altitude pictures. It has an 8-degree field of view when used with a 0.44 inch format slow scan vidicon.

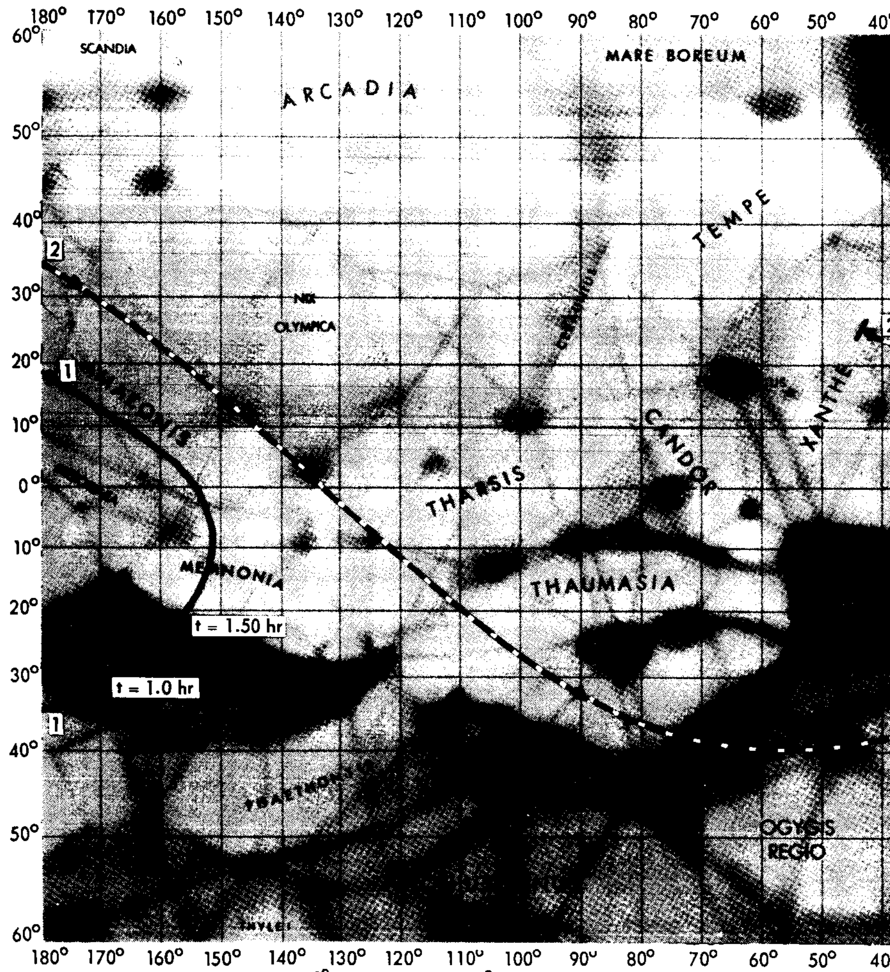
PREFERRED ENTRY SCIENCE INSTRUMENTS CHARACTERISTICS SUMMARY

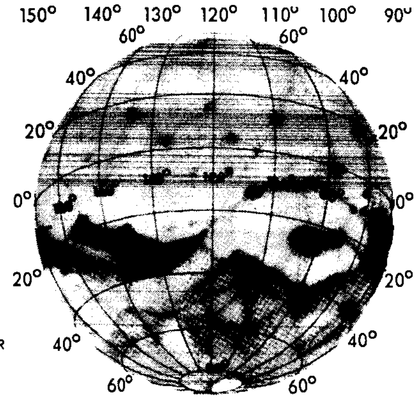
	WEIGHT (lb)	VOLUME (in.^{3})	POWER (watts)	MAX. ENERGY (watt-hr)	TOTAL DATA (k Bits)
Stagnation Region Instruments					
Temperature Transducer	0.5	1.7	0.01	.0025	8.7
Pressure Transducer	1.0	6.3	1.4	0.35	8.7
Base Region Instruments					
Pressure Transducer	1.0	6.3	1.4	0.42	8.7
Temperature Transducer	0.5	1.7	0.01	.003	8.7
Accelerometer	2.0	9.6	4.0	1.2	65.4
Mass Spectrometer	8.0	200	7.0	2.1	47.5
Vidicon Cameras	14.0	700	20.0	5.4	240/Frame
Totals	27.0	925.6	33.8	9.48	

Figure 2-1









Base 101
Base Information Aug 1962

Basic Map Published by
AERONAUTICAL CHART AND INFORMATION CENTER
UNITED STATES AIR FORCE
ST. LOUIS 18, MISSOURI

TRY TV FRAME SEQUENCE

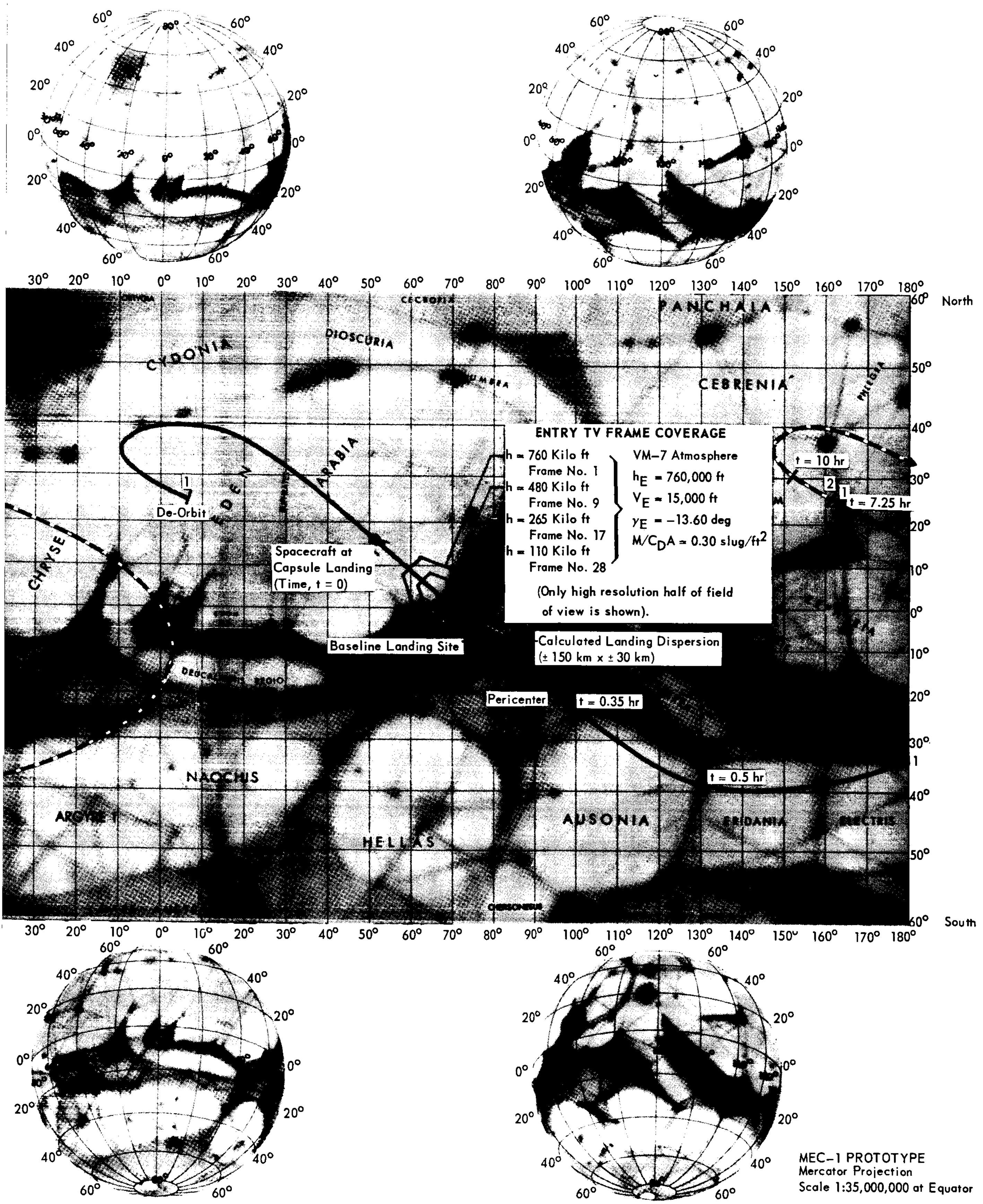


Figure 2.1–1

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PREFERRED DESIGN CONFIGURATION FOR DESCENT IMAGING SYSTEM

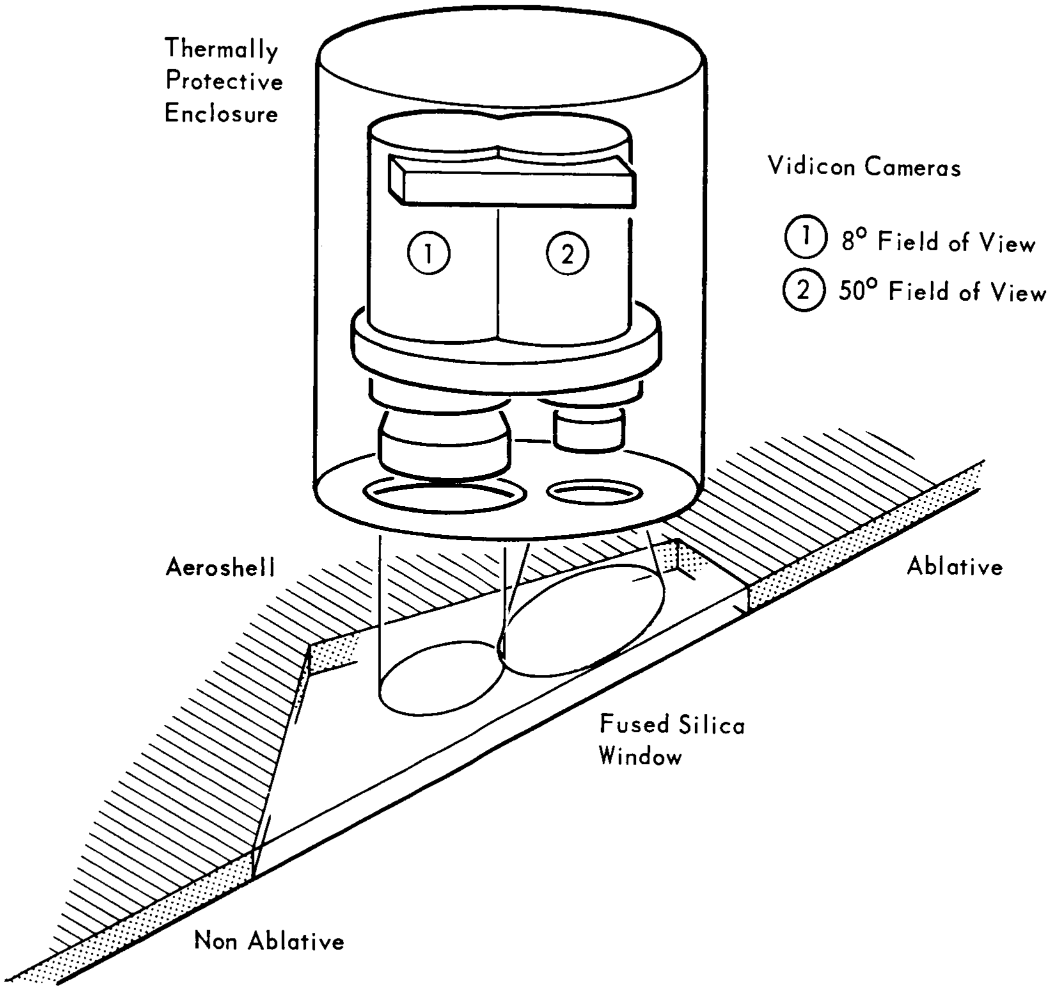


Figure 2.1-2

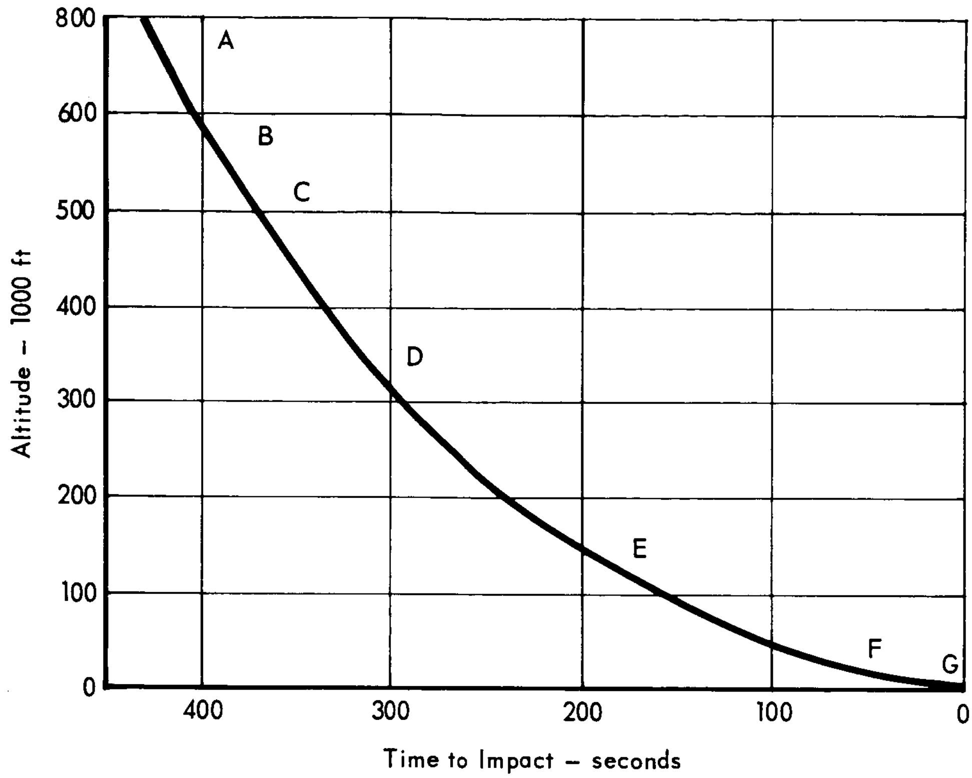
The second camera has a 50 degree field of view and uses a 0.5 inch focal length. It is designed to obtain the primary landing area pictures at altitudes approximating 100,000 feet. The cameras are housed in a thermally protective container and mounted on the bottom of the CBS landing foot pad. This package meets all the necessary physical and environmental constraints. Viewing is conducted through a fused silica window in the Aeroshell. A nonablative nose cap extends out from the apex of the Aeroshell to this window freeing the optical path from ablative contamination.

Operational - The picture sequence starts at 800,000 feet. Cameras operate alternately generating between them one frame every five seconds. A spectral filter is periodically introduced in each optical path to enhance scene contrast. Actual exposure takes only 5-10 milliseconds; readout, however, requires an additional 5 seconds. Between the two units a steady, approximately 50,000 bit/second (6 bit gray scale) data rate is provided. During the 5 seconds of dead time on each camera following readout, the photoconductor is optically erased in preparation for the next frame. Pictures are taken continuously, through communications blackout, and terminate when the camera package is pyrotechnically removed from the CBS at the 90 foot altitude. A typical profile is outlined in Figure 2.1-3.

Performance - Primary landing area pictures are obtained between 100,000 and 200,000 feet with the 50 degree camera. Backup imaging is secured above 465,000 feet with the 8 degree camera. This unit also supplies one meter ground resolution at altitudes 3300 feet and below insuring useful surface evaluation even in the event of parachute and descent engine failure. The continuous picture sequence with its high degree of frame to frame overlap makes available: (a) stereoscopic modeling, (b) photogrammetric surveying, (c) surface and atmospheric photometry (the planet limb, sky, and dark space background appear in early frames), and (d) recording of the effects of entry and descent capsule operations on the images. This satisfies both the primary and secondary objectives without compromising any of the rest of the CBS.

2.2 ATMOSPHERIC DENSITY AND TEMPERATURE PROFILE DETERMINATION METHOD - The measurements used in the determination of the atmospheric profile are in such large quantity and of such diverse types that a statistical procedure is required to process them. Of even more consequence is the method used to ascertain the nature of their influence upon this determination. The following is a description of our preferred maximum likelihood method.

TYPICAL OPERATIONAL/PERFORMANCE PROFILE FOR THE DESCENT IMAGING SYSTEM



A – Imaging Initiation at 800,000 ft

B – Early Photometric Pictures
(Large Ground Areas and Planet Limb)

C – Backup Images for Landing Site Location
(Ground Resolution GR = 1000–2000 ft, Narrow Field)

D – Communications Blackout

E – Primary Imaging for Landing Site Location
(GR = 750–1000 ft, Wide Field)

F – Detailed Surface Imaging (GR – 3 ft)

G – Image Termination

Figure 2.1–3

2.2.1 Quantities Measured - The determination method can utilize essentially all trajectory and predictably related atmospheric property data which is available, both from instruments comprising the Entry Science Package (ESP) and from the instrumentation of the Capsule Bus System (CBS).

The ESP measurements can be summarized as follows:

a.	Stagnation point temperature (from M = 5 on down to Aeroshell separation)	1 sample per second
b.	Capsule base region temperature (through measurable region to the surface)	1 sample per second
c.	Capsule base region pressure 0 to 0.75 psia (through measurable region to the surface)	1 sample per second
d.	Stagnation point pressure 0 to 3 psia (from 800,000 foot altitude down to Aeroshell separation)	1 sample per second
e.	3-axis accelerometer (from 800,000 foot altitude on down)	2 samples per second
f.	Descent images (800,000 feet down, except for communications blackout region)	
g.	Mass spectrometer (M = 5 down to surface)	1 sample per 10 seconds

Additional ESP measurements considered for the alternate payload are:

a. γ-ray backscatter for direct measurement of density.

b. Differential pressure of stagnation/wall at sphere-cone tangency point for determination of dynamic pressure independent of free-stream conditions.

c. Higher altitude mass spectrometer readings.

d. UV and X-ray absorption.

e. Speed of sound experiment

f. Ram spectrometer

g. Wind velocity measurements

h. Doppler measurements between spacecraft and capsule.

The Capsule Bus System measurements include:

a. Altimeter (from 200,000 feet down)

b. Three gyros (all the way, 2 samples per second)

c. Accelerometer

d. Landing radar

e. Internal temperature and pressure sensors

2.2.2 Calculation Technique - The preferred method for calculating the atmospheric density profile from the measurements made by the science instruments during entry is discussed in this section. A linear estimator is used to incorporate all of the useful data into the determination of the atmospheric profile. To determine this profile it is necessary to determine also the trajectory along which these properties are measured. Thus, in addition to estimating density and pressure, we must also estimate the trajectory determining parameters. These include entry conditions and also the actual deceleration history. The acceleration at every calculation sample point is, therefore, included in the state vector of quantities to be estimated.

The estimating procedure is described briefly by the sequence in Figure 2.2.2-1. The state vector for the estimation will also include some systematic errors in the acceleration and the assumed drag coefficient so that the column vector representing the state to be estimated is:

$$\begin{bmatrix} r \\ V \\ \gamma \\ P \\ K_A \\ K_D \end{bmatrix} = \begin{bmatrix} \text{radial distance from planet center} \\ \text{velocity} \\ \text{flight path angle} \\ \text{pressure} \\ \text{accelerometer bias scale factor} \\ \text{drag coefficient scale factor} \end{bmatrix}$$

The dynamic equations describing the system which are used in Steps II and III, indicated in Figure 2.2.2-1 are:

$$\begin{bmatrix} \dot{r} \\ \dot{V} \\ \dot{\gamma} \\ \dot{P} \\ \dot{K}_A \\ \dot{K}_p \end{bmatrix} = \begin{bmatrix} V \sin \gamma \\ -\frac{G\sigma}{r^2} \text{SIN}\ \gamma + (a + K_A b_A) \cos \alpha - (a_N) \sin \alpha \\ (\frac{r}{V} - \frac{G\sigma}{r^2 V}) \cos \gamma + a_N \cos \alpha - (a + K_A b_A) \sin \alpha \\ -\frac{G\sigma}{r^2} \quad \frac{2a}{V} \quad \frac{m}{K_D C_D A} \quad \sin \gamma \\ 0 \\ 0 \end{bmatrix}$$

ESTIMATION PROCEDURE FOR ATMOSPHERIC RECONSTRUCTION

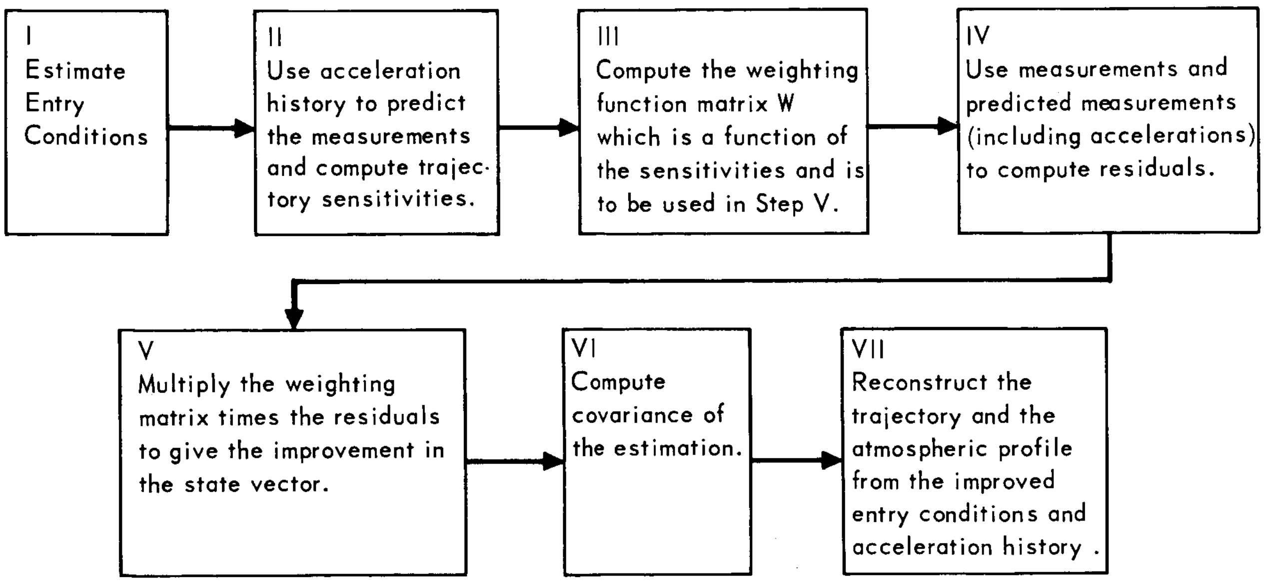


Figure 2.2.2–1

The three-axis accelerometer measurements and the gyro data mentioned earlier are used to resolve the angle of attack, using predicted lift-to-drag characteristics of the Aeroshell. This occurs prior to using the data for trajectory smoothing.

Although the attitude degrees of freedom could conceivably be included as state variables in the estimation, this would complicate the procedure considerably, and is undesirable. The uncoupled nature of the attitude motion allows the smoothing of the acceleration data to be carried out in an independent calculation.

The trajectory and the sensitivity matrix for this system of equations are computed simultaneously.

The various measurements are functions of the components of the state vector. These are used in Step II of Figure 2.2.2-1. Some of these are:

a. Altitude $h_m = r - r_♂$

b. Pressure $P_m = P + K_p a \frac{m}{K_D C_D A}$ Where K_p is the recovery coefficient.

c. Stagnation Temperature

$$T_s = T + K_T \frac{\Gamma - 1}{2} \frac{V^2}{\Gamma R}$$

$$= \frac{P V^2 K_D C_D A}{R 2 a m} + K_T \frac{\Gamma - 1}{2\,\Gamma} \frac{V2}{R}$$

Where:

K_T is the recovery coefficient

R is the gas constant

Γ is the ratio of specific heats

The mass spectrometer measurements are used to determine the correct values of R and Γ for these formulae. The sensitivities of these measurements to the state are also computed.

The trajectory sensitivities and the measurement sensitivities are used to calculate the sensitivity of the state components to each measurement quantity (including acceleration). These are arranged into a matrix denoted by M.

The maximum likelihood weighting matrix depends upon this matrix M and also upon both the expected noise characteristics of the instruments (the Q matrix) and the frequency content of the acceleration history to be extracted. This frequency content and also the a priori covariance of the initial trajectory state is contained in a Matrix R. The larger the correlation terms involved in R, the smoother will be the predicted atmospheric profile. The simplest weighting matrix W (Step III) is computed from the relation:

$$W = RM^T (MRM^T + Q)^{-1}$$

A schematic diagram of this data filtering is shown in Figure 2.2.2-2.

(Hypothetically, it is better to constrain the estimation so that, if no noise existed on the measurements and if the actual acceleration history were a fourth-order polynomial, the predicted acceleration would be this same polynomial.)

2.2.3 Expected Results - This program results in time profiles of the atmospheric density, pressure, and other state variables which are then transcribed into functions of altitude. These profiles can then be theoretically corrected based upon meteorological considerations, to more closely correspond to variations with respect to altitude directly above the landing site.

The expected accuracy of the determination at each calculation sample point is a by-product of the method. This expected accuracy is provided as a sequence of covariance matrices which corresponds to the determination accuracy at points along the trajectory.

These accuracy matrices are necessary in ascertaining the relative importance of the different types of data and in determining the sensitivity of the determination to measurement accuracies. This is done by varying the parameters (such as measurement noise variances) involved in hypothetical determination situations. Also, measurement schedules can be varied to help in optimization of the overall data gathering system. These aspects are discussed in a more quantitative manner in Part E, Section 2 of this report.

2.3 INSTRUMENTS REQUIRED FOR DENSITY AND TEMPERATURE PROFILE DETERMINATION - Density and temperature profile determinations require data from both ESP science instruments and CB engineering instruments.

The ESP instrument requirements include stagnation and base region temperature transducers, stagnation and base region pressure transducers, and a tri-axis accelerometer.

The CB instruments required include the radar altimeter, the rate gyros, the guidance accelerometers, and the landing radar. Correlation of data from all of these sources will provide the information required to reconstruct the atmospheric density and temperature profiles by the method described in Section 2.2 above.

2.3.1 Temperature Sensors - Two temperature probes will be used on the ESP to measure the stagnation and base temperature. Two probes are required because the vehicle configuration changes over the portion of the mission during which temperatures are measured. One is located at the stagnation region because the data

SCHEMATIC OF ESTIMATION PROCEDURE FOR ATMOSPHERIC RECONSTRUCTION

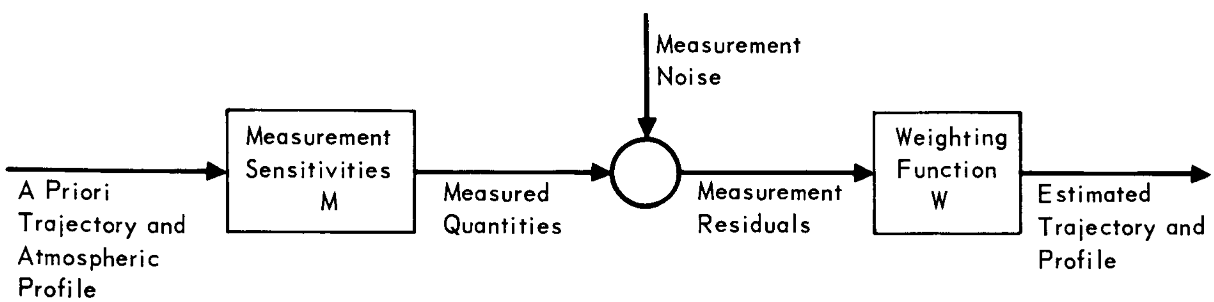


Figure 2.2.2-2

obtained in this area has the greatest degree of interpretability. The aerodynamic and thermodynamic coefficients' behavior are more fully known in this area of the vehicle than at any other. However, when the Aeroshell separates at between 15,000 and 19,000 feet altitude, the stagnation temperature sensor separates too; so a second sensor is located in the base region of the vehicle primarily to obtain temperature data from separation to the surface. Figure 2.3.1-1 illustrates the stagnation temperature/pressure transducer configuration and Figure 2.3.1-2 illustrates the base region temperature/pressure transducer configuration.

2.3.1.1 <u>Instruments</u> - The temperature transducers are total temperature probes using platinum resistance elements and resistance bridge output circuits. The stagnation sensor is designed to accept gas flows over an angle-of-attack range of $\pm 20°$ with little or no effect on the sensed temperature's relation to the free-stream temperature. In addition, the design of this sensor includes features which minimize the recovery error over the Mach number range from 0 to 5. The range of this sensor is 150°K to 1200°K.

In the base region, the gas flow velocity will be below Mach 0.3 during the period when measurements are desired. Below this velocity, the recovery error effects are negligible so that special aerodynamic considerations are not of prime importance in the design of this sensor. The range of this sensor is 150°K to 330°K.

Both sensors incorporate features to protect the sensing element from solar radiation, radiated heat, and conducted heat. Figure 2.3.1-3 presents the instruments requirements and characteristics. Part F, Section 1.4 of this volume presents a detailed functional description of the temperature probes.

2.3.1.2 <u>Operation</u> - The purpose of these transducers is to obtain temperature measurements during certain portions of the entry profile for use in reconstructing atmosphere free-stream temperature and density profiles by the method described in Part D, Section 2.2 of this volume.

In order to accomplish the purpose of these instruments, the operating procedure described below is to be followed. Three hundred seconds prior to entry, the ESP science instruments are powered and data is read out from each temperature probe at a rate of 1 sample/second. A valve in the stagnation temperature transducer is closed during this period and the peak heating period which follows to keep the temperatures at the sensing element within design limits. Approximately 30 seconds after the peak stagnation pressure has been reached, the valve is opened to permit gas flow over stagnation transducer. Useful data is obtained from this

STAGNATION REGION PRESSURE – TEMPERATURE ASSEMBLY CONFIGURATION

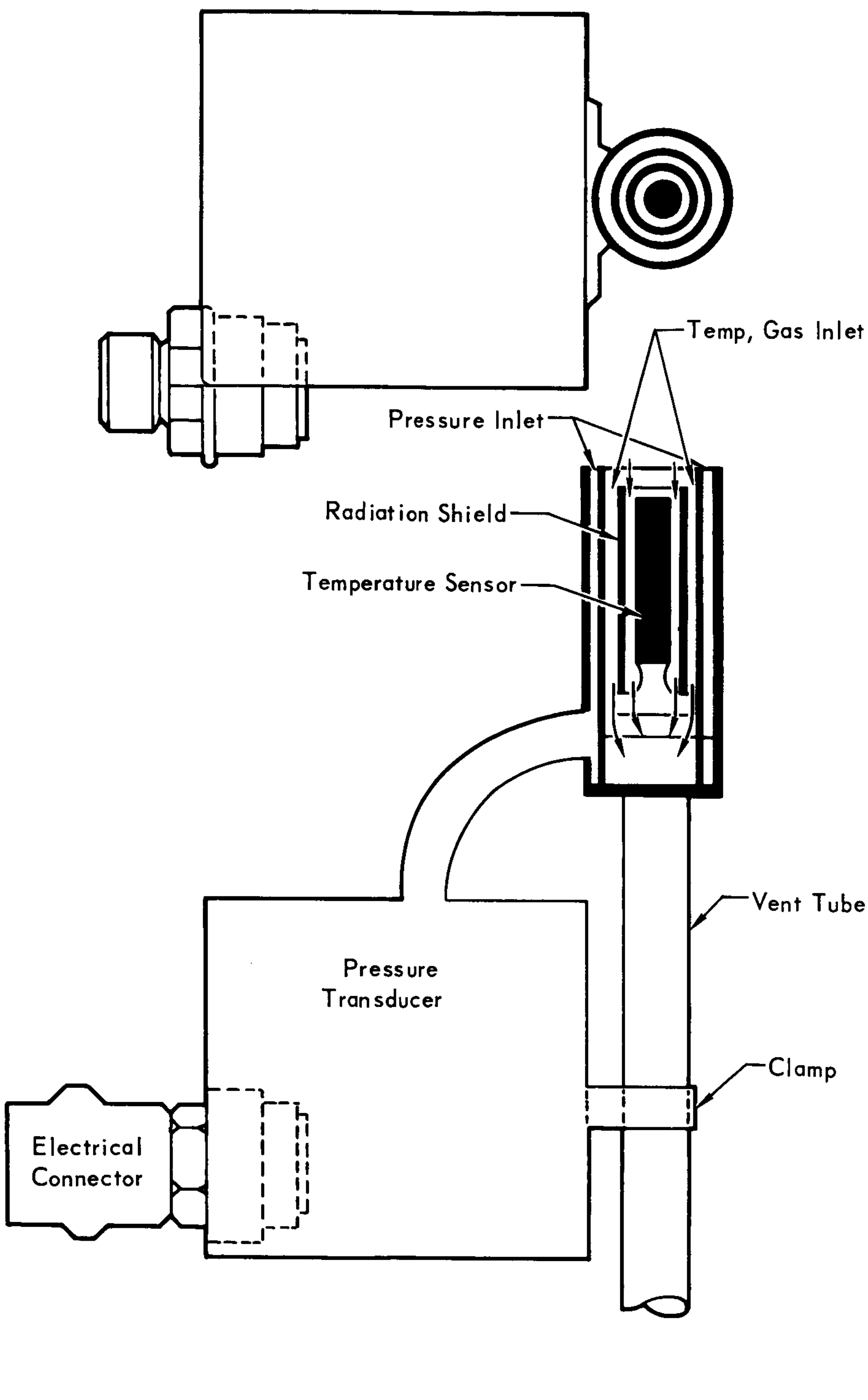


Figure 2.3.1–1

BASE REGION TEMPERATURE – PRESSURE ASSEMBLY CONFIGURATION

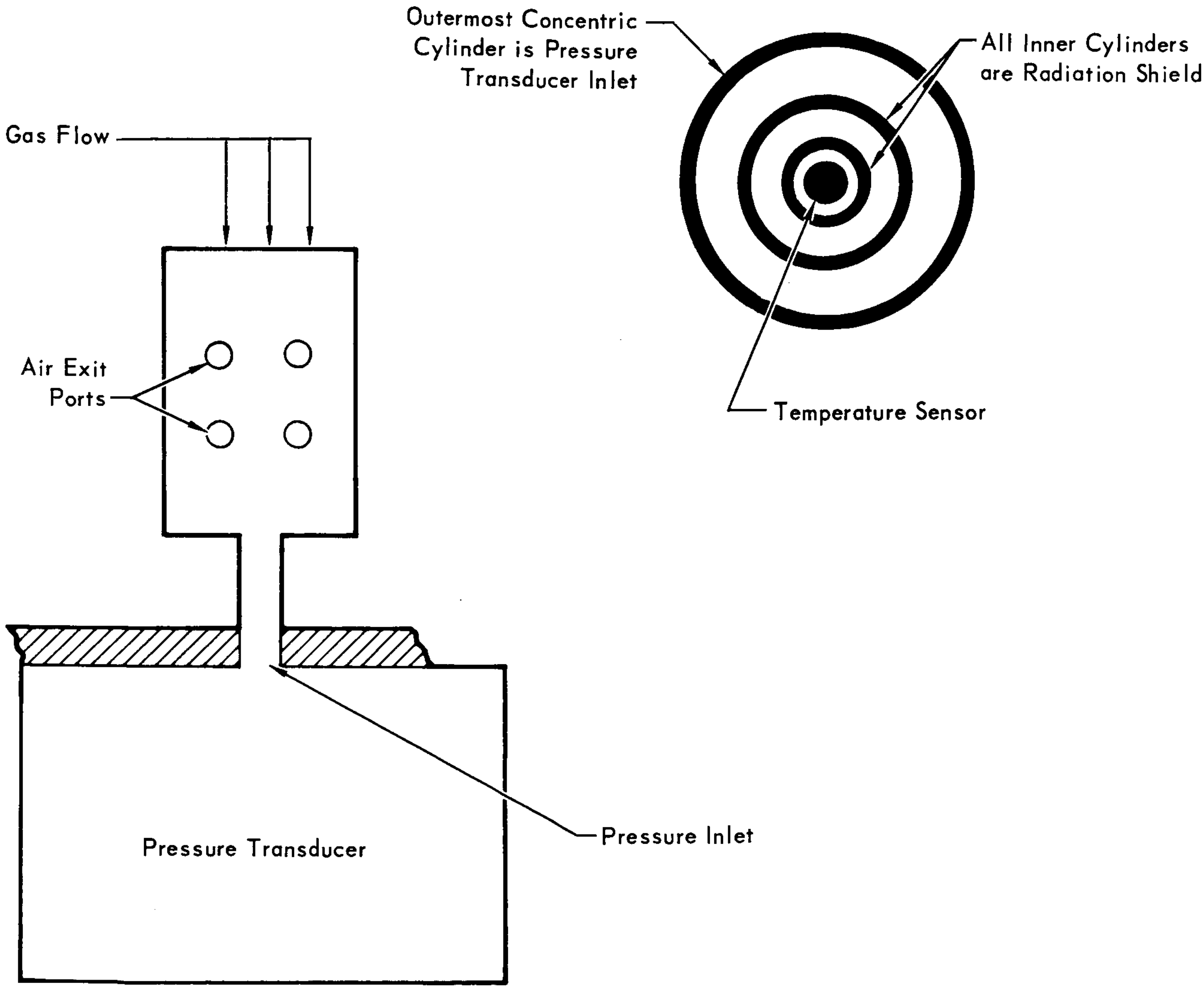


Figure 2.3.1–2

ESP TEMPERATURE TRANSDUCER REQUIREMENTS AND CHARACTERISTICS

INSTRUMENT	REQUIREMENTS AND CONSTRAINTS	VOLUME	WEIGHT	POWER
Stagnation Temp Transducer	• Survive Entry Heats prior to use • Accept gas flow over 20° angle-of-attack excursions • Experience minimum effects from conducted and radiated heat • Be vented to base	1.7 cu in.	0.5 lb	10^{-2} watts
Base Region Temp Transducer	• Experience minimal effects from radiated and conductedheat • Be protected from solar radiation • Experience minimal effects from terminal propulsion system	1.7 cu in.	0.5 lb	10^{-2} watts

Figure 2.3.1-3

point until Aeroshell separation from the stagnation transducer at which time primary reliance will be placed on measurements from the base region transducer. Although both transducers are powered throughout the entry, useful data is obtained only during selected portions of this period. Stagnation data is useful only from 30 seconds after peak pressure until Aeroshell separation, while dynamic tests on the CB will indicate the useful data range of the base region transducer. The base region temperature sensor continues to provide data for 120 seconds post touchdown.

2.3.1.3 Interfaces - The ESP temperature probe interfaces lie in five categories: mounting, power, data, thermal, and operational.

The stagnation temperature sensor is mounted on the interior of the Aeroshell and protrudes through it to provide atmosphere gas access. The vent tube connected at the aft end of the instrument is routed through the CB and passes near the mass spectrometer on its path to the CB base region. This tube will be configured so that it separates readily with the Aeroshell without restricting flow into the remaining portion of the tube so that mass spectrometer analyses can be continued following Aeroshell separation.

The type of readout circuit employed with these sensors requires a well regulated power source, therefore, the 5.00 ± 0.05 vdc power supply is used for both instruments.

The outputs (0 to 40 mvdc) from the two transducers are hardwired to a low level multiplexer in the ESP Telemetry Subsystem for multiplexing, amplification, and subsequent routing through the telecommunications functions.

The thermal interface is satisfied during the detailed design phase of this program. The transducers must be designed so that tolerable temperatures are not exceeded during the peak heating portion of the entry.

Operational interfaces which must be satisfied to successfully obtain the temperature measurements are:

- The valve in the stagnation temperature transducer vent tube must remain closed until 30 seconds (nominal) after peak pressure is sensed by the stagnation pressure transducer.
- Ablation products must not block flow in the stagnation temperature sensor vent tube.
- The wire bundle running to the stagnation region transducer must be separated prior to Aeroshell separation.
- Aeroshell separation and corresponding vent tube separation must leave the new entrance port for the mass spectrometer inlet unobstructed.

- Terminal propulsion plume effects on the base region transducer must be determined for use in data analysis.

2.3.2 Pressure Transducers - Two pressure transducers are used during entry and terminal descent to measure pressures relatable to free-stream pressures. One is located at the nominal CB stagnation point and the other is located in the base region. Both transducers make pressure measurements over as large a portion of the entry as possible. The upper altitude operating limits are determined by transducer threshold sensitivity, while the lower altitude limits are determined by Aeroshell separation altitude in the case of the stagnation pressure transducer and by termination of ESP surface operations in the case of the base region transducer. Both transducers are combined with their corresponding temperatures to form integrated pressure-temperature assemblies as shown in Figures 2.3.1-1 and 2.3.1-2.

2.3.2.1 Instruments - The pressure transducers are variable capacitance type using a deflecting diaphragm transducer with two capacitance pick-off signals and an output circuit which provides a 0 to 5 vdc output signal proportional to the applied pressure on the diaphragm.

The stagnation transducer is designed to measure the pressure over the entire entry profile where pressures are high enough to be measured and the Aeroshell is present. Special consideration is given to the operating temperature range requirement for this instrument. The transducer design is such as to provide some protection from conducted heat and the installation in the Aeroshell provides any additional conduction barriers required. Tubing length and size are optimized to keep instrument response times to 25 milliseconds or less. The range of this transducer is 0 to 3.0 psia.

The base region transducer is designed to make measurements from the time transducer threshold is reached until the ESP is powered down. The range of this transducer is 0 to 0.5 psia which is sufficient to cover the entire range of pressures encountered during entry (within our entry condition envelope) into any of the 10 VM model atmospheres. Figure 2.3.2-1 presents a requirements and characteristics table for the pressure transducers. A more detailed description of the instruments is presented in Part F, Section 1.3 of this volume.

2.3.2.2 Operation - In order to achieve the mission objectives of these instruments, the operation described below will be followed. Approximately 300 seconds prior to the 800,000 feet entry point, the ESP science instruments are powered and each pressure transducer is sampled at a rate of one sample per second. The transducer outputs are sampled at this rate over the entire ESP operation period even though

ESP PRESSURE TRANSDUCER REQUIREMENTS AND CHARACTERISTICS

INSTRUMENT	REQUIREMENTS AND CONSTRAINTS	VOLUME	WEIGHT	POWER
Stagnation Pressure Transducer	• Tolerate radiated and conducted heat at diaphragm during entry • Mount on stagnation temperature assembly • Have response time short enough to obtain data at a rate of one sample per second. • Have small acceleration sensitivity	6.3 cu. in.	1 lb	1.4 watts
Base Region Pressure Transducer	• Integrated with base region temperature sensor • Provide meaningful data after aeroshell separates. • Have small acceleration sensitivity • Response time must be short enough to permit readings at a rate of one sample per second. • Be mounted so that data is unambiguous while terminal propulsion engines are firing.	6.3 cu. in.	1 lb	1.4 watts

Figure 2.3.2-1

meaningful data is generated over only a portion of the entry. The stagnation transducer senses a slope change in pressure versus time from positive to negative and activates a nominal 30 second delay circuit. After the 30 seconds have elapsed, the stagnation transducer generates a signal to open the valve which has been inhibiting flow through the stagnation temperature transducer vent tube and to break the inlet seal on the mass spectrometer. The Aeroshell is separated between 15,000 and 19,000 feet altitude; so, the power leads to this transducer must be separated before this altitude. At this point the primary reliance for pressure data shifts to the base region transducer. The base region transducer will continue to obtain meaningful data for approximately 120 seconds after touchdown.

2.3.2.3 Interfaces - The ESP pressure transducer interfaces lie in five categories: mounting, power, telemetry, thermal control and operational.

The stagnation pressure transducer is mounted on the stagnation temperature assembly with access to the Martian atmosphere. The access to the atmosphere is achieved by utilizing a concentric cylinder placed around the outermost radiation shield of the total temperature transducer and attaching the transducer port near the base of this cylinder.

The base region transducer is married to its corresponding temperature transducer so that it also is an integrated assembly. The base region assembly is mounted at the aft end of the ESP with access to the environment at the aft end of the CB.

These transducers utilize the 28.5 $\pm$ 5 Vdc power level available from the Power Subsystem and any regulation or level changing functions required are performed by the transducer electronics.

The outputs (0 to 5 vdc) are hardwired to the high level multiplexer in the ESP Telemetry Subsystem for multiplexing and subsequent routing through the other telecommunications functions.

The stagnation pressure transducer must be mounted such that conducted heat is kept to a level compatible with transducer design limits. An additional thermal control subsystem interface requirement may arise due to operating temperature limits on the electronics. Existing transducers of this type can tolerate electronics temperatures from -200°C to 125°C.

Operational interfaces which must be satisfied to successfully obtain the pressure measurements are:

- o CB angle-of-attack excursions must be kept to $\pm$20° so that pressure coefficient knowledge is acceptable.

- Ablation products must not be allowed to block the transducer inlets.
- The wire bundle running to the stagnation pressure transducer must be separated prior to Aeroshell separation

2.3.3 Accelerometer - A tri-axis accelerometer will be used on the ESP to measure accelerations from the beginning of entry at 800,000 feet to touchdown on the surface. A tri-axis unit is used rather than three single-axis units to minimize the weight and volume requirements. The unit is located just ahead of the SL on the CB roll axis and slightly ahead of the c.g. for both the CB entry and CB terminal descent configurations. The usefullness of the accelerometer measurements will decrease at lower altitudes due to: longer integration periods, parachute deployment, Aeroshell separation, and terminal propulsion.

2.3.3.1 Instrument - The accelerometer is a tri-axis, servo force balance type with a digital output circuit. The digital output was selected in order to obtain an accuracy of ±0.1% of full scale, since the analog - to - digital converter in the Telemetry Subsystem has only seven bit capability which yields a quantization error of approximately 0.8%. The instrument is filtered to provide a flat response from dc to 1 cycle per second with an 18 db per octave roll-off. The ranges for the accelerometer (in Earth g's) are:

o Roll Axis	0 to 5g and 0 to 30g
o Pitch and Yaw Axes	± 2g

Figure 2.3.3-1 presents the accelerometer requirements and characteristics.

2.3.3.2 Operation - The purpose of this instrument is to obtain very accurate acceleration measurements over the entire entry for use in reconstructing atmospheric temperature and density profiles.

In order to accomplish this purpose, the operating procedure described below is followed. When a nominal 300 seconds before the entry altitude of 800,000 feet is reached, the accelerometer is powered along with all other ESP science instruments. Data will be taken at a rate of 2 samples per second from this point until touchdown. During this operation period, a knowledge of c.g. motions must be available to permit data corrections. The c.g. location will shift gradually due to ablating and will shift suddenly two times: at parachute deployment, and at Aeroshell separation.

2.3.3.3 Interfaces - The ESP accelerometer interfaces fall in four general categories: mounting, power, telemetry and thermal.

The ESP accelerometer is mounted to the Capsule Bus structure on the roll axis just ahead of the SL. The mounting attachment permits the accelerometer to be

ESP ACCELEROMETER REQUIREMENTS AND CHARACTERISTICS

INSTRUMENT	REQUIREMENTS AND CONSTRAINTS	VOLUME	WEIGHT	POWER
Accelerometer	• Must be aligned with CB entry configuration axes • Must be located outside of SL • Must be located near c.g. for entry and terminal descent configuration. • Temperature during operation must be controlled • Digital Output required	10 cu. in.	2.0 lb	4.0 watts

Figure 2.3.3-1

aligned with the CB entry configuration axes.

This instrument uses the 28.5 $\pm$ 5 Vdc power from the ESP Power Susbystem, with any further regulation or level changes being performed by the accelerometer electronics. The unit has four data outputs which are tabulated in Figure 2.3.3-2. These outputs are hardwired to the ESP Telemetry Subsystem for formatting and subsequent routing through the other telecommunications operations. An additional interface is the data request pulses sent to the accelerometer to activate the readout circuits.

The accelerometer requires thermal control to keep its operating temperature within design limits, typically -40°C to 90°C. In addition, if significant accuracy gains can be achieved by correcting the output per the temperature coefficients of the instrument, the accelerometer temperature will be monitored.

2.3.4 Capsule Bus Equipment - The instruments contained in the ESP require supporting information from the CB instruments in order to determine the atmosphere temperature and density profiles by the method described in Section 2.2. Information generated by the CB altimeter, gyros, accelerometer, and landing radar will be used to supplement the ESP instrument data. The information obtained from each is tabulated in Figure 2.3.4-1.

The altitude from the radar altimeter provides the data for correlating all ESP measured values to altitude over the region from approximately 200,000 feet to 50 feet.

The gyros provide two types of outputs - three-axis body rates and three-axis attitude errors. This information is used to make attitude generated corrections to the data. Examples of this application are pressure transducer coefficient corrections due to angle-of-attack excursions and stagnation temperature transducer recovery coefficient corrections due to the same angle-of-attack excursions. Attitude rate information will be used to modify the accelerometer data.

The accelerometer provides an indication of Z-axis acceleration and serves as a back-up for the ESP accelerometer roll axis acceleration output.

The landing radar also provides slant range data which is used to supplement the ESP data for altitudes from 15,000 feet to 10 feet.

2.4 ATMOSPHERIC COMPOSITION DETERMINATION - The composition measurement experiment needs further definition before a recommendation of a specific mass spectrometer can be made. For purposes of discussion a quadrupole mass spectrometer is assumed, operating in a continuous scan mode with sampling begun at Mach 5.0.

ESP ACCELEROMETER DATA SUMMARY

PARAMETER	RANGE	SIGNAL TYPE	SAMPLE RATE
Roll Acceleration	0 to 5g 0 to 30g	10 bit digital	2 sps
Roll Range	5g or 30g	0 or 28.5 vdc	2 sps
Pitch Acceleration	± 2g	10 bit digital	2 sps
Yaw Acceleration	± 2g	10 bit digital	2 sps

Figure 2.3.3-2

CB SUPPORT INFORMATION TELEMETERED FOR ATMOSPHERIC PROFILE RECONSTRUCTION

INSTRUMENTS	PARAMETER	WORD LENGTH	RATE
Radar Altimeter	Altitude from after black out or 200K feet to 50 ft	14 bits	1 SPS
Gyros	Roll Rate Pitch Rate Yaw Rate Roll Attitude Error Pitch Attitude Error Yaw Attitude Error	6 bits + Sign 6 bits + Sign 6 bits + Sign 6 bits + Sign 6 bits + Sign 6 bits + Sign	1 2 2 1 1 1
Accelerometer	Z Axis Acceleration	7 bits	5
Landing Radar	Slant Range	15 bits	1

Figure 2.3.4-1

2.4.1 Mass Spectrometer Characteristics - A typical quadrupole mass spectrometer instrument weighs 8 lbs., encloses a volume of 200 cu. in. and requires 7 watts of power. The time required to scan the 10 to 60 mass range is 2 seconds. The mass spectrum data output at the end of each scan is 400 bits. The instrument is powered from 300 seconds prior to entry to 120 seconds post touchdown. It receives atmospheric samples however, only from about Mach 5.0 through 120 seconds post touchdown. Laboratory studies are necessary to determine the accuracy for water vapor detection at the levels expected in the Martian atmosphere.

2.4.2 Atmospheric Gas Sampling Equipment - Atmosphere samples are obtained via a molecular leak, from the stagnation temperature outlet tube from Mach 5.0 to touchdown. Prior to Mach 5.0 the stagnation temperature outlet is closed. The valve closing this tube and the seal on the mass spectrometer inlet are opened on signal from the stagnation pressure transducer. The stagnation pressure sensor senses peak pressure and after an appropriate time delay initiates a signal to open the valve. The time delay between Mach 5.0 and peak pressure varies primarily with entry conditions. Entry conditions are predicted based on deorbit commands and the time delay required is selected and transmitted with the deorbit commands prior to capsule separation. Separation of the stagnation temperature outlet tube employs a slip free fitting and is such that aerodynamic flow will continue through the tube following Aeroshell separation. After touchdown, the mass spectrometer inlet is switched to a small capillary sampling tube. This second sampling system is necessary to insure fresh samples as aerodynamic flow through the stagnation temperature outlet tube ceases at touchdown. The switch to the second or surface sampling system is made by a valve operating on signal from the ESP telemetry subsystem programmer initiated at terminal propulsion cut-off.

2.5 INTEGRATION AND SUPPORT ESP SUBSYSTEM REQUIREMENTS - The integration of the ESP science instrument payload into the ESP imposes requirements on almost all ESP subsystems. This section details the requirements imposed on the various ESP subsystems.

2.5.1 Power - In order to support the ESP science instruments the Power Subsystem provides:

- Voltage levels of 28.5 $\pm$ 5 Vdc and 5.00 $\pm$ 0.05 Vdc
- Peak power of 43.8 watts, non-recurring, of 1 sec maximum duration
- A maximum energy of 11.20 watt hours based on a 300 second warm-up period prior to 800,000 feet and a 600 second post-landed mode. A nominal ESP science instrument power profile is presented in Figure 2.5.1-1.

NOMINAL ESP SCIENCE INSTRUMENT POWER PROFILE

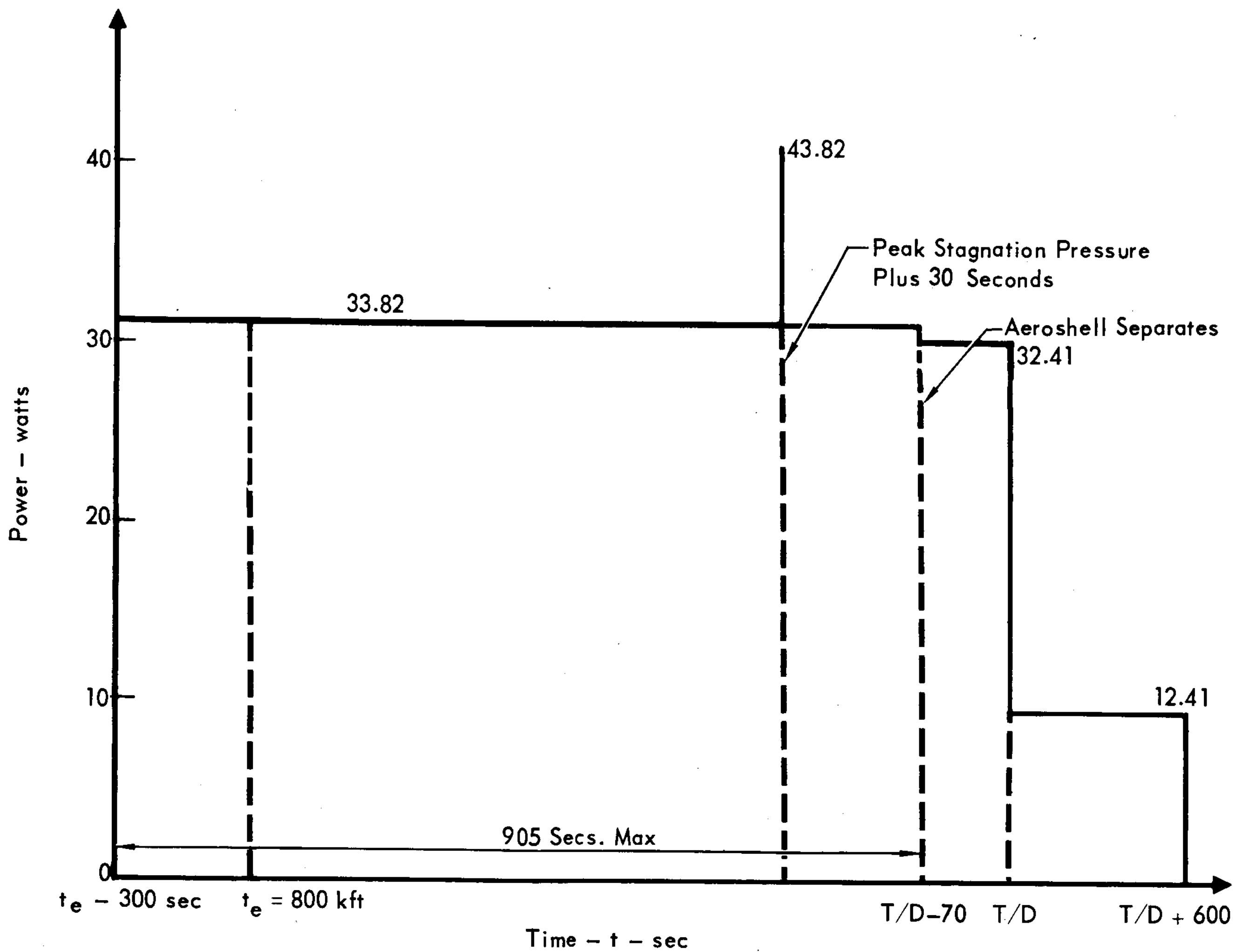


Figure 2.5.1–1

The power levels used in the preparation of this profile are:

Descent TV	20 watts
Accelerometer	4 watts
Pressure Transducer	1.4 watts each
Temperature Probes	10^{-2} watts each
Mass Spectrometer	7 watts
Stagnation Inlet Valve	10 watts

2.5.2 Sequencing - The accommodation of the ESP science instruments imposes a requirement for five sequencing signals to be provided from external sources:

a. Generate a turn-on signal 300 seconds prior to reaching the nominal 800,000 ft. altitude

b. Generate a back-up signal, based on the aerodecelerator deployment sequence at 23,000 ft., for opening the valve on the stagnation temperature outlet tube and breaking the mass spectrometer inlet seal

c. Generate a TV separation signal based on a 90 ft. slant range signal from the Landing Radar

d. Generate a Surface Mode Operation signal based upon the terminal propulsion cut-off command

e. Generate an ESP turn-off command 10 minutes after touchdown

The sequence in Figure 2.5.2-1 illustrates the sequence of events beginning with turn on at 300 seconds prior to the predicted altitude of 800,000 ft. and ending 600 seconds after touchdown.

2.5.3 Data Handling, Storage, and Communication - The ESP science instruments impose numerous requirements on the telecommunications functions. The Telemetry subsystem must accommodate the science instrument data outputs presented in Figure 2.5.3-1, commutate them, perform 7 bit analog-to-digital conversions on them (when required), and prepare them for routing through the remaining telecommunications functions of transmission and storage.

All science data generated by the ESP science instruments must be transmitted from the Capsule Lander to the Spacecraft and received and stored by the Spacecraft for subsequent relay to Earth. The data which must be accommodated includes the high rate (TV), the low rate science, and the engineering support data.

2.5.4 Structure and Mechanisms - The ESP Science instruments are located throughout the CB with as many of the instruments as possible grouped with the ESP science instrument support equipment along side the SL at the aft end of the CB.

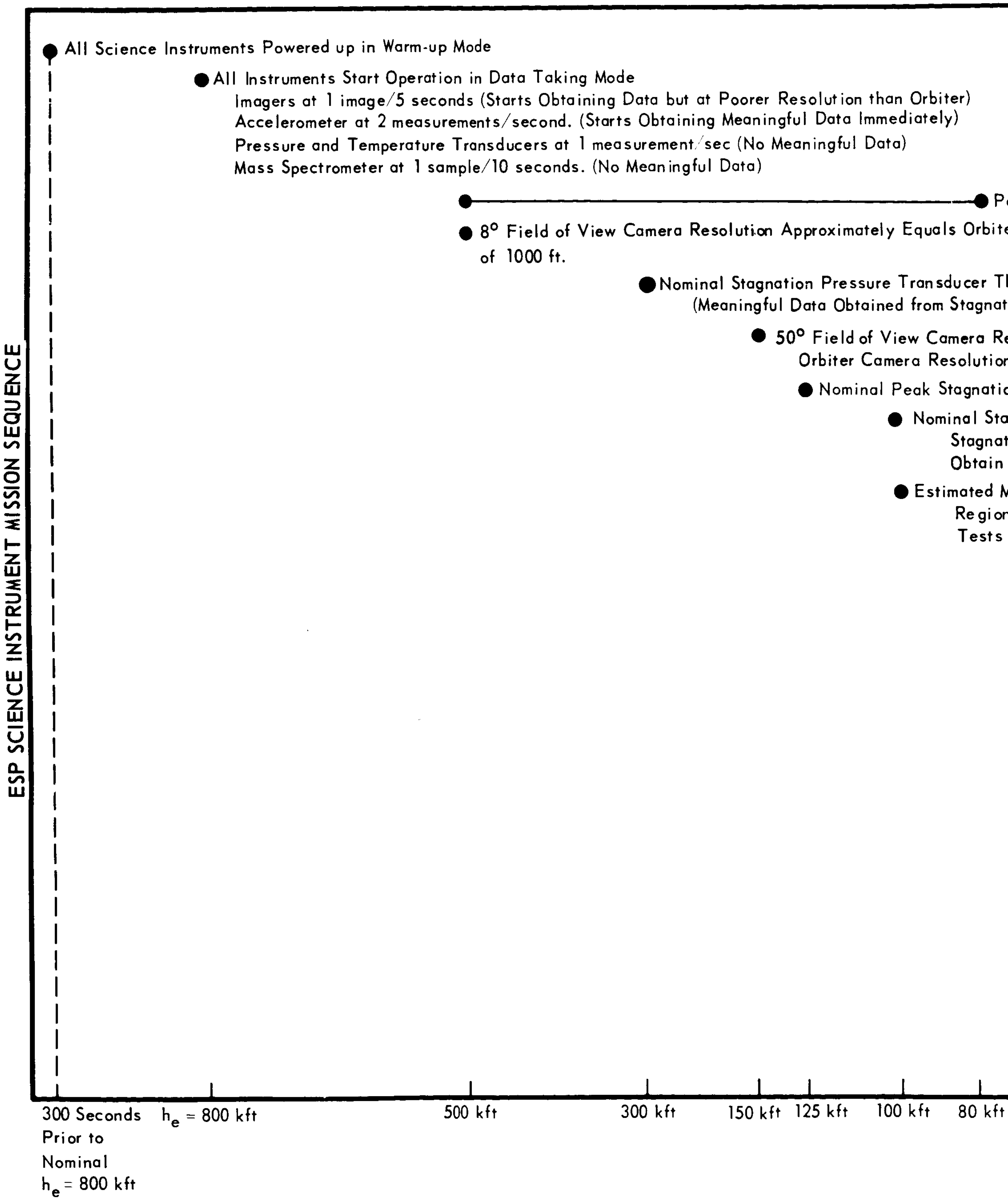

ESP SCIENCE INSTRUMENT MISSION SEQUENCE
All Science Instruments Powered up in Warm-up Mode
All Instruments Start Operation in Data Taking Mode
Imagers at 1 image/5 seconds (Starts Obtaining Data but at Poorer Resolution than Orbiter)
Accelerometer at 2 measurements/second. (Starts Obtaining Meaningful Data Immediately)
Pressure and Temperature Transducers at 1 measurement/sec (No Meaningful Data)
Mass Spectrometer at 1 sample/10 seconds. (No Meaningful Data)
P
8° Field of View Camera Resolution Approximately Equals Orbite
of 1000 ft.
Nominal Stagnation Pressure Transducer Th
(Meaningful Data Obtained from Stagnat
50° Field of View Camera Re
Orbiter Camera Resolution
Nominal Peak Stagnatio
Nominal Sta
Stagnat
Obtain
Estimated M
Region
Tests
300 Seconds Prior to Nominal h_e = 800 kft
h_e = 800 kft
500 kft
300 kft
150 kft
125 kft
100 kft
80 kft

Figure 2.5.2-1

ɔtential Communications Black Out Period (Imager Data Lost)
r Resolution

ıreshold (Range 520 kft to 105 kft)
ion Pressure Transducer)

solution Approximately Equals
of 1000 ft.

n Pressure Point (Range = 220 kft to 55 kft) Start 30 second delay in Stagnation Pressure Transducer

gnation Vent Tube Open (Range 180k to 32 kft)
ion Temperature Sensor and Mass Spectrometer
Meaningful Data.

eaningful Data Threshold for Base
Pressure – Temperature Data (Development
Required for Precise Determination)

- Nominal Parachute Deployment
- Nominal Aeroshell Separation. Stagnation Pressure and Temperature Transducers Separate with Aeroshell and thus Cease Generating Meaningful Data.
- Terminal Propulsion System Ignition Possible Degradation of Mass Spectrometer Data
- Descent Imagers Forcefully Separated (Termination of Imaging)
- Touchdown – Termination of Accelerometer Meaningful Data
- Switch Mass Spectrometer to Surface Operation Mode. During this Period, the Base Region Pressure and Temperature Sensors and the Mass Spectrometer Obtain Meaningful Data
- Entry Science Package Operations are Terminated

23kft | 19 to 15 kft | 5 kft | 90 ft | Terminal Propulsion Termination Command | T/D + 600 seconds

2-28-2

TYPICAL ESP SCIENCE INSTRUMENT PARAMETER LIST

INSTRUMENT	PARAMETER	FORM	RATE
Descent Imagers	*Video – Imager #1	Digital 6 bits/el.	50,000 bps
	Video – Imager #2	Digital 6 bits/el.	50,000 bps
	Frame Number	Digital 6 bits	72 bpm
	Iris Setting	Digital 6 bits	72 bpm
	Filter Position	Digital 2 bits	24 bpm
	Camera Number	Bilevel	12 sps
	Mode	Bilevel	12 sps
	Vidicon Temperature(2)	0 to 40 mvdc	.1 sps
	Lens Assy. Temperature(2)	0 to 40 mvdc	.1 sps
	Electronics Temperature	0 to 40 mvdc	.1 sps
	Target Voltage (2)	0 to 5 vdc	.1 sps
	Erase Lamp Status (2)	Bilevel	1 sps
	Light Level	0 to 5 vdc	.2 sps
Mass Spectrometer	Spectrum	Digital 8 bits/el	40 bps
	B^+ Voltage	0 to 5 vdc	.1 sps
	RF Voltage	0 to 5 vdc	.1 sps
	Diode Temperature	0 to 40 mvdc	.1 sps
	Pump Pressure	0 to 5 vdc	.1 sps
	Ion Chamber Temperature	0 to 40 mvdc	.1 sps
	Filament Number	Bilevel – 2 bits	.1 sps
Accelerometer	Acceleration (3)	Digital 10 bits/el	20 bps
	Range	Bilevel	2 sps
Stagnation Pressure	Pressure	0 to 5 vdc	1 sps
Stagnation Temp.	Temperature	0 to 40 mvdc	1 sps
Base Pressure	Pressure	0 to 5 vdc	1 sps
Base Temp.	Temperature	0 to 40 mvdc	1 sps

Notes: bps = bits per second
bpm = bits per minute
sps = samples per second

* Video #1 & 2 Alternate During Operation Periods

Figure 2.5.3-1

The Structural/Mechanical Subsystem is required to provide mounting attachments compatible with science instrument configuration, mission induced environments, and instrument operating requirements. Remotely mounted hardware and their location requirements are listed in Figure 2.5.4-1.

FIGURE 2.5.4-1

REMOTELY MOUNTED ESP SCIENCE INSTRUMENT REQUIREMENTS

Instrument	Location Requirements
Accelerometer	Ahead of and as near as possible to the CB entry configuration center of gravity
Descent TV	Forward looking parallel to CB roll axis compatible with viewing window configuration
Stagnation Assembly	On Aeroshell at nominal stagnation point

The mass spectrometer and the base region pressure temperature assembly are mounted in the basic ESP area at the aft end of the CB. The instrument mounting must also be such that parachute deployment and Aeroshell separation will not destroy any but the stagnation region instruments. The most significant requirements imposed on the separation sequence are:

a. The descent imagers must continue to function.

b. The mass spectrometer sample tube separation must not inhibit subsequent mass spectrometer analyses by closing off the remaining portion of the tube.

2.5.5 Actuation - During the operational lifetime of the ESP science instruments, several actuation signals and functions must be supplied by the ESP supporting subsystems. They include:

a. Separation of the stagnation region wire bundle prior to Aeroshell separation.

b. Back-up actuation signal for breaking the mass spectrometer inlet seal and opening the stagnation temperature outlet tube based on an aerodecelerator deployment mark.

c. Separation of the stagnation temperature transducer vent tube at Aeroshell separation.

d. Pyrotechnic severance of the descent TV based on a 90 ft height signal from the landing radar.

e. Switching the mass spectrometer inlet based on a terminal propulsion termination signal.

2.5.6 Cabling - The location of the ESP science instruments throughout the CB requires routing of power, timing, and data signals from the basic ESP locations to the accelerometer, stagnation pressure transducer, stagnation temperature sensor, descent imagers, and electrically, mechanically, and pyrotechnically actuated devices used to support the ESP science instruments.

2.5.7 Thermal Control - The ESP science instruments require thermal control heater and insulation support.

The mass spectrometer must operate at an elevated temperature and insulation is required to protect surrounding equipment and keep temperature maintenance costs to a minimum.

The accelerometer and descent TV are temperature sensitive and require thermal control during their operation.

The pressure transducer located in the stagnation region requires mounting insulation to keep diaphram area temperatures within pressure transducer design operating temperature limits.

SECTION 3

SUPPORTING ESP SUBSYSTEMS

The ESP support equipment includes the electrical power, telemetry, radio, antenna, structural mechanical and thermal control subsystems and a portion of the spacecraft mounted capsule support equipment. Figure 3.0-1 shows a functional diagram of the ESP supporting subsystems and their interfaces with each other and with other PV subsystems.

In addition to distributing operational ESP power, the electrical power subsystem supplies regulated spacecraft power or ESP battery power to the cruise commutator and heaters during interplanetary cruise. Spacecraft power is provided to all subsystems in response to in-flight checkout commands. SL battery power is available to the ESP as a backup mode.

The telemetry subsystem controls the science instruments and conditions science and engineering data prior to its transmission by the radio subsystem. The CB and ESP telemetry subsystems interleave ESP engineering and low rate science data with CB engineering data, thus providing a functionally redundant path for the low rate data. Spacecraft-generated synchronizing signals are routed to the ESP by the CB cruise commutator during interplanetary cruise. After FC activation, synchronizing signals are generated by the CB cruise commutator. In addition, in-flight checkout data and a synchronizing signal are routed through the CB to the SC.

The limited ESP sequencing and timing requirements are met by the telemetry subsystem programmer. A signal from the CB sequencer and timer activates the ESP. The CB turn-off command is used as a back-up for the ESP self turn-off.

The antenna subsystem has both an operational RF interface (ESP/SC relay link) and a test RF interface with the ESP portion of the spacecraft-mounted Capsule support equipment. The thermal control subsystem accepts power from the electrical power subsystem for operation of electrical heaters.

3.1 ELECTRICAL POWER - The electrical power subsystem provides power for inflight monitoring during cruise periods when spacecraft power is not available to the Flight Capsule, and power for equipment operation from preseparation to ESP shutdown after landing.

3.1.1 Requirements and Constraints - The requirements and constraints influencing the design of the electrical power subsystem are as follows:

ENTRY SCIENCE PACKAGE SUBSYSTEM INTERFACES

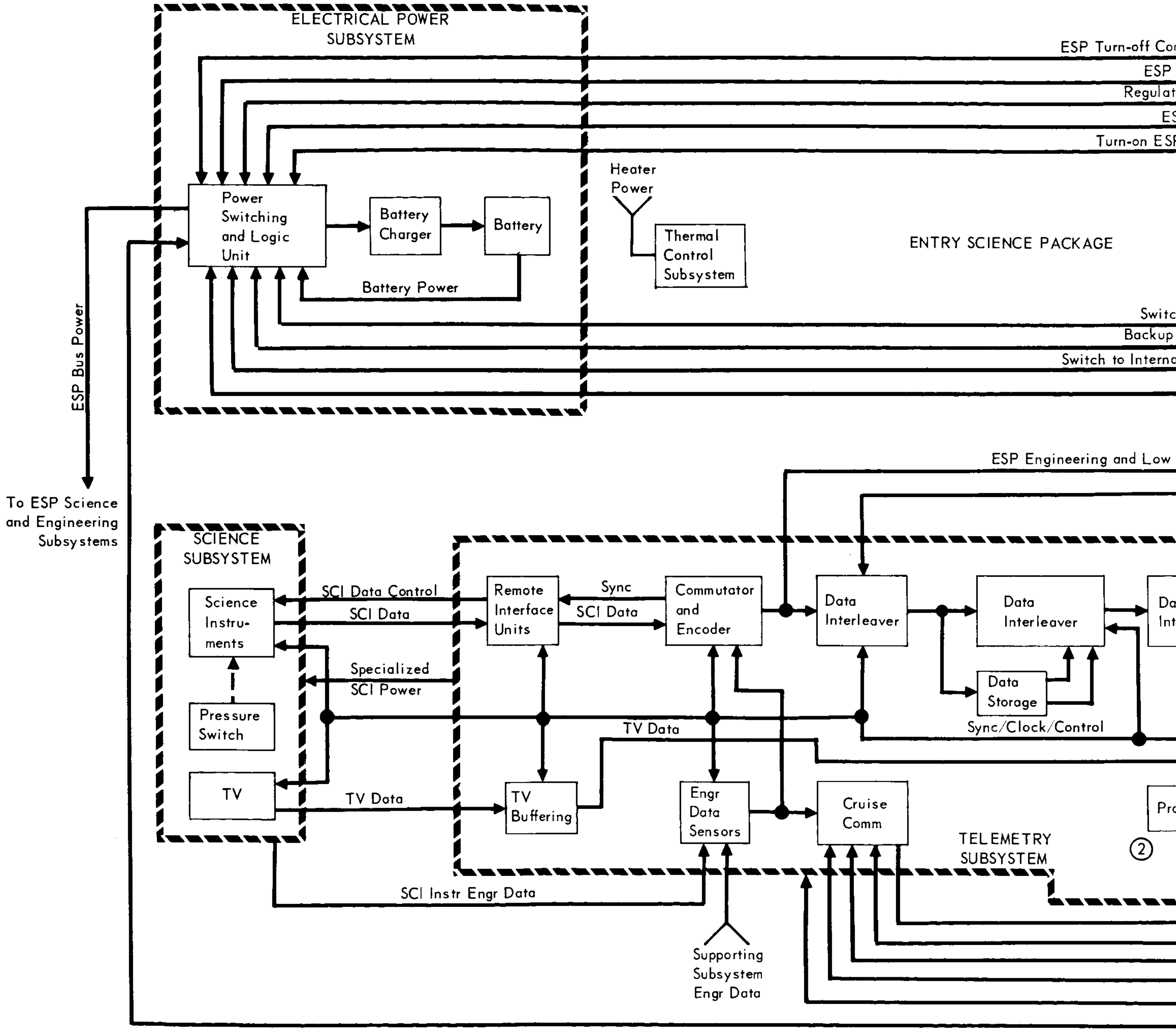
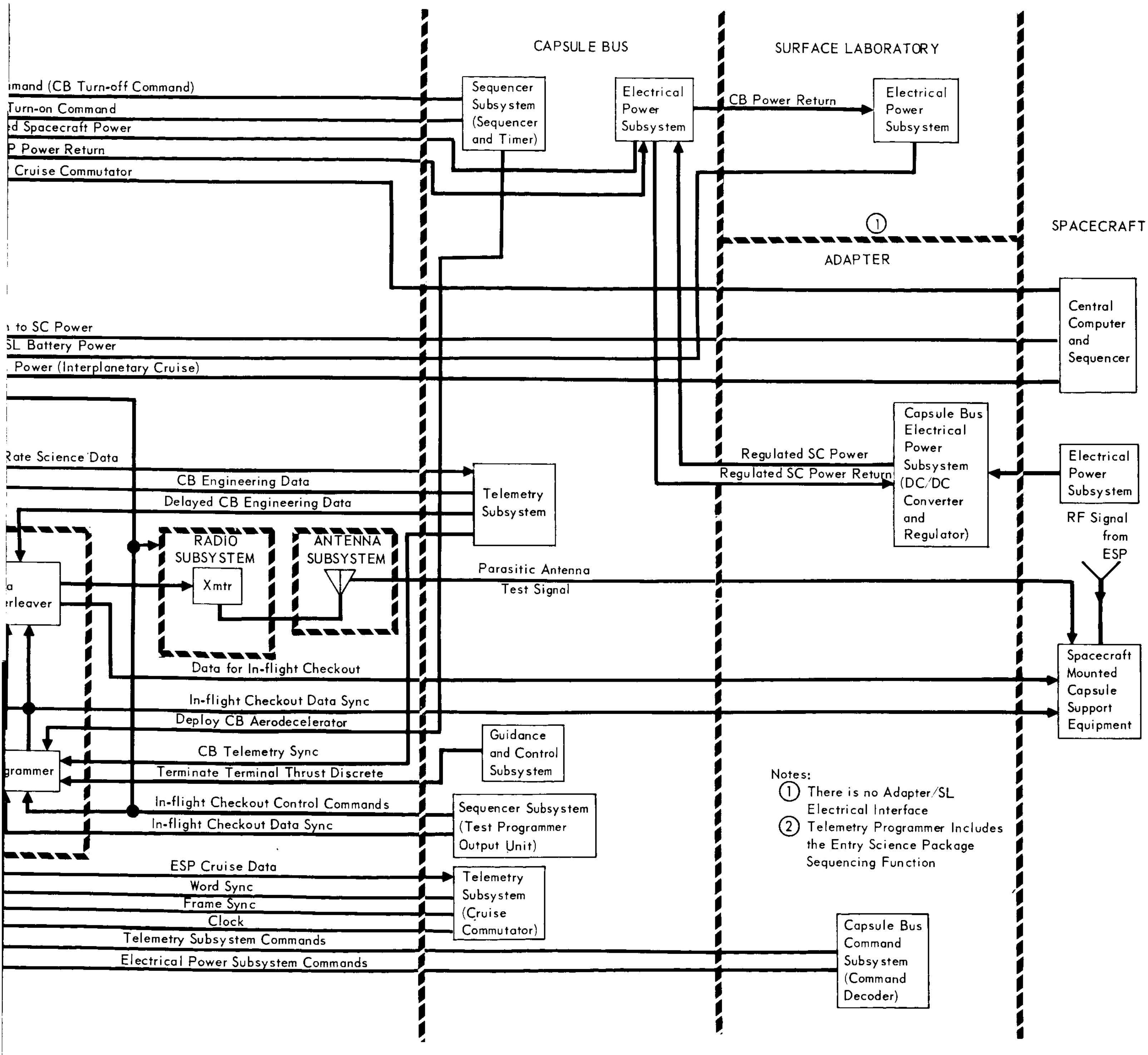


Figure 3.0-1


CAPSULE BUS
SURFACE LABORATORY
SPACECRAFT
ADAPTER
1
Sequencer Subsystem (Sequencer and Timer)
Electrical Power Subsystem
CB Power Return
Electrical Power Subsystem
imand (CB Turn-off Command)
Turn-on Command
d Spacecraft Power
P Power Return
Cruise Commutator
Central Computer and Sequencer
to SC Power
SL Battery Power
Power (Interplanetary Cruise)
Capsule Bus Electrical Power Subsystem (DC/DC Converter and Regulator)
Regulated SC Power
Regulated SC Power Return
Electrical Power Subsystem
Rate Science Data
CB Engineering Data
Delayed CB Engineering Data
Telemetry Subsystem
RADIO SUBSYSTEM
Xmtr
ANTENNA SUBSYSTEM
a
rleaver
Parasitic Antenna Test Signal
RF Signal from ESP
Spacecraft Mounted Capsule Support Equipment
Data for In-flight Checkout
In-flight Checkout Data Sync
Deploy CB Aerodecelerator
CB Telemetry Sync
Terminate Terminal Thrust Discrete
grammer
Guidance and Control Subsystem
In-flight Checkout Control Commands
In-flight Checkout Data Sync
Sequencer Subsystem (Test Programmer Output Unit)
ESP Cruise Data
Word Sync
Frame Sync
Clock
Telemetry Subsystem Commands
Electrical Power Subsystem Commands
Telemetry Subsystem (Cruise Commutator)
Capsule Bus Command Subsystem (Command Decoder)
Notes:
1 There is no Adapter/SL Electrical Interface
2 Telemetry Programmer Includes the Entry Science Package Sequencing Function

a. The subsystem must provide approximately 345 watts of power and 230 watt hours of electrical energy during the Flight Capsules' descent to the surface of Mars.

b. The subsystem must provide approximately 10 watts of electrical power for operation of ESP operating equipment during cruise when Flight Spacecraft power to the ESP is turned off.

c. The subsystem must distribute power from other sources to the ESP operating equipment.

d. The subsystem must be independent of the other Flight Capsule electrical power subsystems to the maximum extent practical.

3.1.2 Equipment Description - A block diagram of the electrical power subsystem is shown in Figure 3.1-1. The subsystem consists of a sealed, silver zinc battery, a battery charger, and a power switching and logic unit. The battery is an 8.0 amp-hour, sealed, manually activated (at fabrication) silver zinc battery, designed for a high discharge rate. The battery charger is a two-step float charger. It charges the battery at the 50 hour rate (C/50) after periods of battery usage during cruise, and then maintains the battery in a float charge mode, applying essentially open circuit battery potential to its terminals.

The power switching and logic unit contains the power distribution bus, and a power transfer switch (latching relay) for selecting either external power from the CBS or internal power from the ESP battery. The power switching and logic unit also contains voltage sensors for (1) detecting loss of external power from the spacecraft and transferring the power distribution bus to internal battery power, or detecting return of the power and removing internal power from the distribution bus, and (2) detecting loss of internal power, and transferring the ESP to Surface Laboratory power. The unit also contains on-off switches (latching relays) to control the Entry Science Package equipment and power source fault protection devices.

3.1.3 System Operation - During the launch phase and until Flight Spacecraft solar cell panel deployment, the Entry Science Package operates on internal battery power. This power is required for cruise commutator (telemetry), instrumentation, and electrical heater operation.

When solar power becomes available, the Flight Spacecraft central computer and sequencer transfers the ESP to Flight Spacecraft power and the battery in the Entry Science Package is put on charge. A voltage sensor in the power switching

ENTRY SCIENCE PACKAGE ELECTRICAL POWER SUBSYSTEM BLOCK DIAGRAM

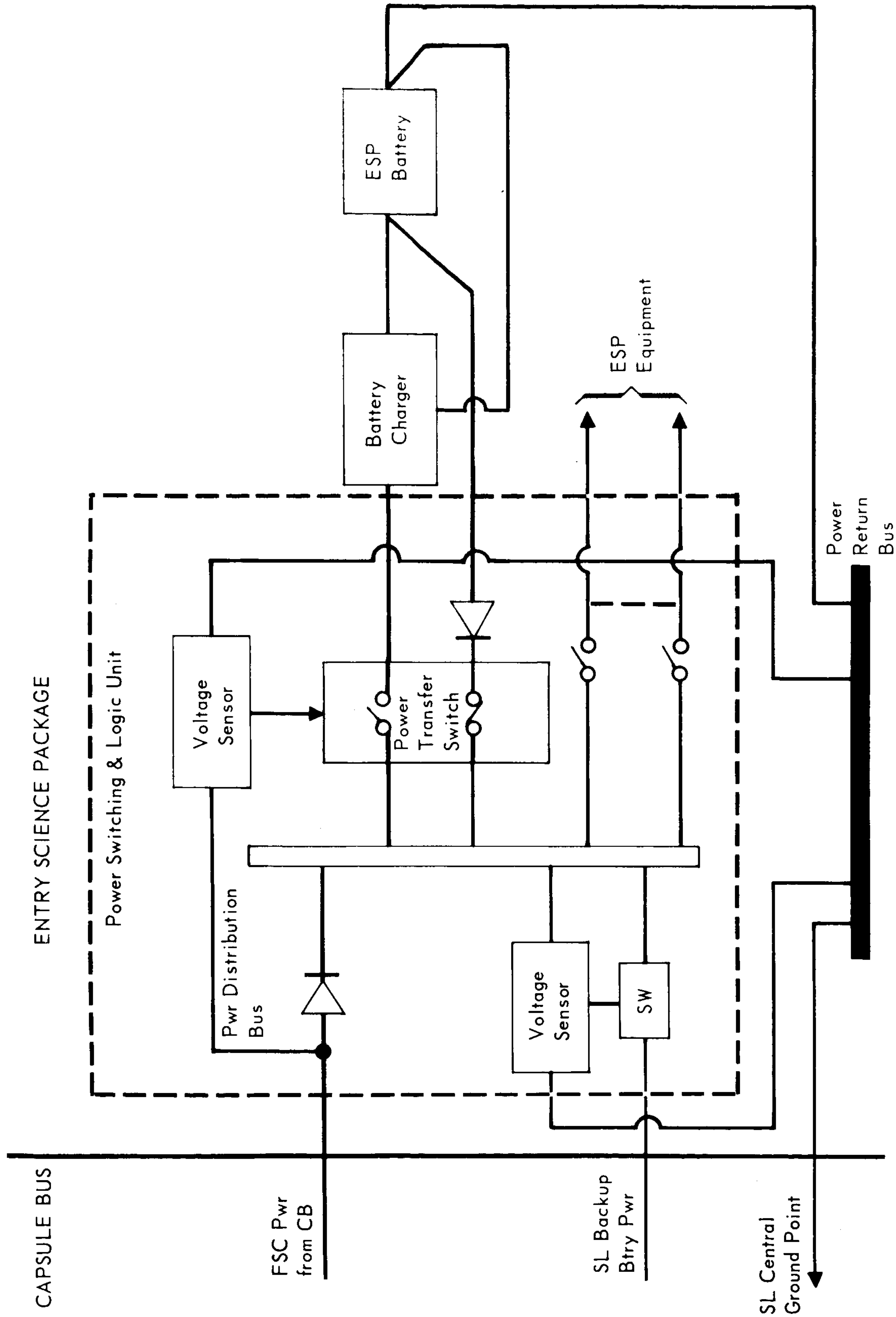


Figure 3.1-1

3-4

and logic unit senses the turning on of Flight Spacecraft Power and provides a redundant means of switching.

During spacecraft maneuvers, when Flight Spacecraft power to the ESP is not available, the Flight Spacecraft central computer and sequencer transfers the ESP to internal battery power, to continue operation of the cruise commutator, instrumentation, and electrical heaters. The voltage sensor in the power switching and logic unit provides an alternate means of accomplishing this switching. When Flight Spacecraft power is again turned on, the battery is put on charge, and power for equipment operation is again derived from the Flight Spacecraft.

Preseparation checkout of the ESP equipment is performed, using spacecraft power for those tests requiring less than 175 watts. Transfer to internal power is made for checkout of the high-rate radio subsystem and is controlled by the test programmer in the CB. At termination of preseparation checkout the test programmer switches the battery back to the charging mode.

Prior to separation, the ESP is put on internal power by the Capsule Bus sequencer and timer. A voltage sensor in the power switching and logic unit is enabled by the CB S&T also, and if low bus voltage is detected after this time the voltage sensor closes a switch in the unit, making power from the surface laboratory available to the power distribution bus.

3.1.4 Major Trade Summaries - Major trade studies in the electrical power subsystem area include (1) selection of an ESP power source, and (2) selection of the method of providing redundant power to the ESP. A summary trade study table is shown in Figure 3.1-2.

3.2 TELECOMMUNICATIONS - The requirement for the ESP telecommunications subsystems is to return accurate data during all phases of the mission. In particular, these subsystems must monitor critical engineering parameters from launch until ESP activation; transmit science and engineering data from activation at 800,000 feet until landing, and support checkout of the ESP before launch and during preseparation checkout in orbit.

The data transmitted during the active phase of the ESP mission are dominated by digital television data which are transmitted at a rate of one frame every five seconds or approximately 50,000 bits per second. The minimum total data requirement is 5×10^6 bits during the ESP mission. The probability of error must not exceed 10^{-3} in transmission to the spacecraft to insure a probability or error of 5×10^{-3} at the output of the MDE on Earth.

ESP ELECTRICAL POWER SUBSYSTEM TRADE STUDY SUMMARY

STUDY	ALTERNATIVES	MERITS OF SELECTED APPROACH
Power Source Selection	$\boxed{\text{Sealed Ag Zn Battery}}$ Automatically Activated Ag Zn Battery Silver Cadmium Battery Nickel Cadmium Battery Hydrazine Powered Turbine Alternator Provide Power From CB	Minimum Weight No Surface Contamination Independence of Systems
Redundant Power for ESP	Redundant Battery in ESP $\boxed{\text{Redundant Power From SL}}$	Minimum Weight

Selected Alternative $\boxed{}$

Figure 3.1-2

The telecommunications subsystems must provide a means of recovering the data (with the exception of television) collected during blackout and minimize the effects of multipath interference. This interference can occur due to reflection of the RF signal from the surface of Mars during the period immediately after blackout when the capsule attitude is such that significant RF energy is radiated toward the planet.

3.2.1 Subsystem Descriptions - A functional diagram of the ESP subsystems is shown in Figure 3.2-1 and includes the telemetry, data storage, radio and antenna subsystems.

Telemetry Subsystems - The telemetry subsystem contains a cruise commutator which monitors critical engineering parameters and transfers its output to a similar unit in the Capsule Bus. The remainder of the ESP telemetry subsystem consists of a clock, programmer, commutator and data interleavers.

As shown in Figure 3.2-1, data from the Capsule Bus and low rate data from the ESP are interleaved and transmitted over both links so that all data except ESP television have an alternate transmission path.

Since the ESP television dominates the data rate, the resultant ESP data rate is virtually unaffected by the addition of data from the CBS. The ESP telemetry subsystem clock is slaved to that in the Capsule Bus to simplify this interleaving process. However, in case of failure in the CBS, the ESP clock will operate independently and asynchronously with the CBS.

The commutator and programmer use an interlaced tube format. The formats are controlled by instructions stored in the memory of the programmer. The programmer also generates or forwards to the other ESP subsystems five signals required for ESP sequencing during its active phase. This commutator has 85 channels. The cruise commutator has 22. The data rates at various points are shown on Figure 3.2-1.

Data Storage Subsystem - The data storage subsystem consists of two core memories and associated electronics. All data, with the exception of television, is transfered through these memories which provide output delays of 50 and 150 seconds. The operation is equivalent to a 150-second delay line with a tap at 50 seconds. The values of the delays have been chosen to insure complete retransmission of all data at least once between the end of blackout and impact on Mars in the case of a terminal deceleration system failure.

The data storage equipment on the spacecraft includes a 30 x 10^6 bit tape recorder/reproducer and auxillary electronics. The tape recorder interfaces with

ESP TELECOMMUNICATIONS FUNCTIONAL BLOCK DIAGRAM

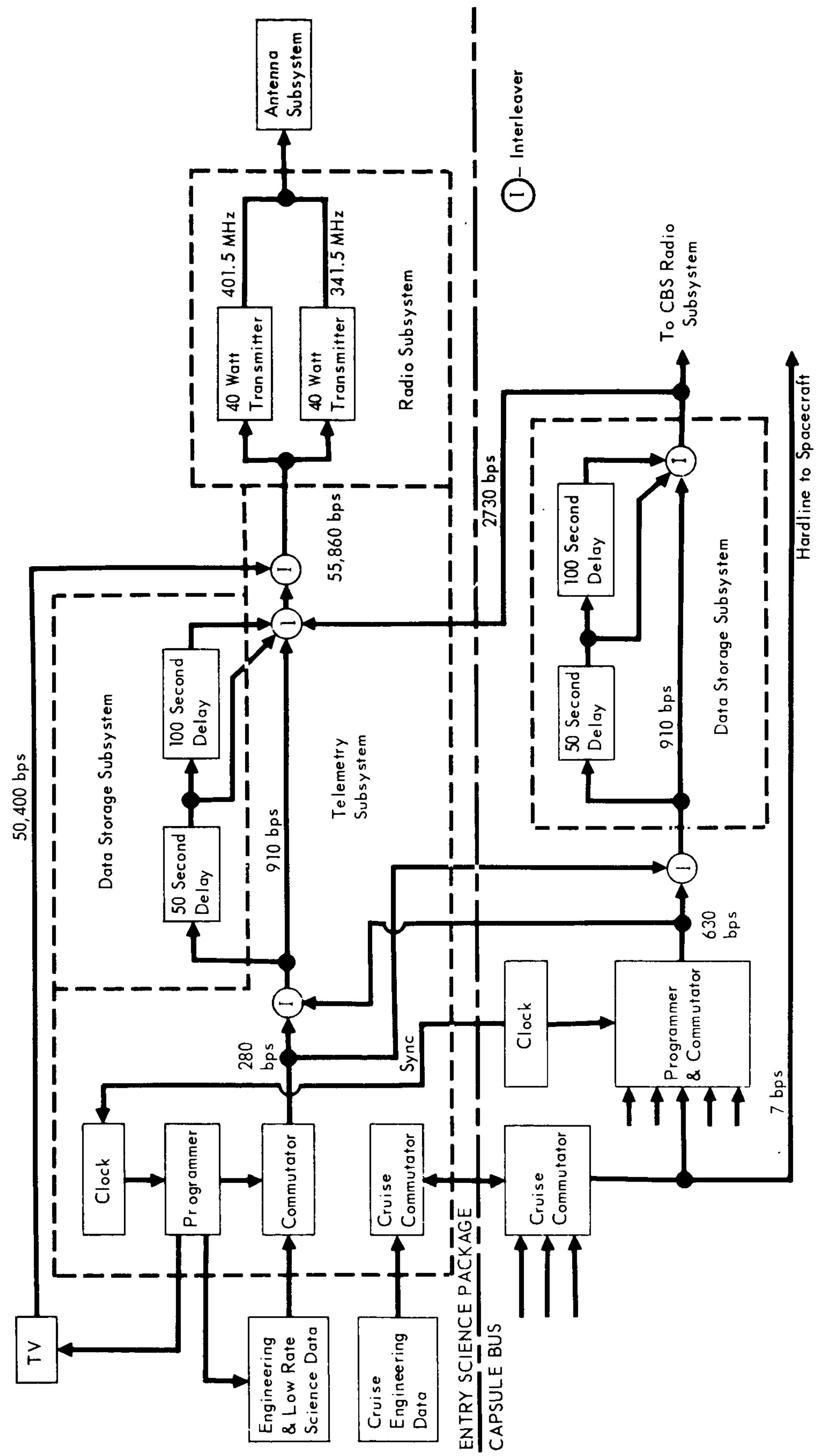


Figure 3.2-1

the spacecraft mounted Data Distribution Unit, which relays record, off, playback instructions and playback bit sync from the spacecraft, and routes the data. Tape speed is controlled by an internal precision oscillator during the record mode, and by the playback electronics in the dump mode.

Radio Subsystem - The radio subsystem consists of two 40 watt frequency shift keyed (FSK) transmitters mounted on the ESP, and two non-coherent receivers, a diversity combiner, and a bit synchronizer mounted on the spacecraft. Two transmission links provide frequency diversity which has been incorporated to overcome multipath interference.

Antenna Subsystem - The ESP mounted antenna is a cavity-backed Archimedian spiral which provides a 95-degree beamwidth and a nearly Gaussian pattern which is symmetrical about the roll (-Z) axis of the capsule. This antenna has circular polarization and a maximum gain of 5.1 dB.

The spacecraft mounted receiving antennas are axial mode helices mounted perpendicular to a mast extending from the spacecraft and normal to the plane of the capsule descent orbit. The CB receiving antenna is mounted on the same mast which is positioned before capsule spacecraft separation to provide coverage for the particular descent and entry trajectory selected. The ESP receiving antenna has a 55-degree beamwidth and a peak gain of 9.9 dB.

The telemetry subsystem also supports checkout of all other ESP subsystems both before launch and during preseparation checkout. In these tests, an end-to-end check of the telemetry subsystem is performed first with provisions for more detailed checks should anomalies be detected. The telemetry subsystem then gathers the data from checks of other subsystems.

The telemetry subsystem has two modes for checkout. The low rate checkout mode (2730 bps) monitors ESP subsystems other than television and the high rate checkout mode (55,860 bps) allows checkout of the television subsystem. The checkout sequence is controlled by the Capsule Bus Test Programmer.

3.2.2 Operational Description - The telecommunications subsystems operate in one mode at a data rate of 55,860 bps throughout the entry and descent phases of the ESP mission. In addition the cruise commutator monitors critical engineering parameters from before launch until the ESP is turned off shortly after landing. These data are fed to the Capsule Bus cruise commutator for relay to the spacecraft.

The ESP continues to operate for several minutes after landing. Communication coverage during this period will be limited by the spacecraft mounted receiving antenna pattern. However, all ESP data (except television) is relayed via the Capsule Bus relay link for a period of 120 seconds after landing, which will insure recovery of all essential data.

Major trade studies in the ESP telecommunications area included selection of frequencies, modulation, synchronization, and transmitter type. A summary of the primary alternatives considered in these and other areas is presented in Figure 3.2-2.

3.3 STRUCTURAL/MECHANICAL - The structural/mechanical subsystem provides the physical elements in the Flight Capsule for the support, alignment, protection and separation of the components of the ESP Subsystem from the time of the initial installation of these components until completion of the ESP mission.

3.3.1 Requirements - All components of the ESP cannot be conveniently located together in a single package. The functional requirements of certain components dictate their location at widely separated points in the Flight Capsule as summarized in Figure 3.3-1. An environmentally controlled compartment is required for most equipment, but remote location and alignment is required for some science instruments.

3.3.2 Subsystem Description - The ESP structure consists of the six elements, as defined in Figure 3.3-1. The Capsule Bus structure serves as the interconnecting element for the ESP structural components. Figure 3.3-2 shows the ESP structural arrangement in the Capsule Bus.

Accelerometer Mounting - The entry center of gravity for the capsule is located on the roll axis at capsule station 152.6 which falls within the parachute catapult envelope. The Capsule Lander base platform hub fitting supports the catapult and affords the nearest available location to the c.g. for the accelerometer package. Mounting holes are provided in the hub fitting to attach the accelerometer at a position on the roll axis 3.0 inches below the entry c.g.

Mounting for the Entry Television Cameras - The entry television cameras and associated electronics are packaged into a single container, which is insulated with externally located fiberglass blankets. The assembly is supported below the lower surface of the Capsule Lander footpad by the pyrotechnic thruster that is required to eject the package prior to impact. Three adjustable compression struts are attached to the container, fitting into sockets provided on the lower surface of the footpad. A window is incorporated in the Aeroshell to provide the unobstructed field of view required for the cameras during entry.

TRADE STUDY CONCLUSIONS

DESIGN AREA	ALTERNATIVES AND SELECTED APPROACH	MERITS OF SELECTED APPROACH
Transmitter Frequency	100 – 330 MHz * **330 – 405 MHz** 405 – 500 MHz	Transmitter frequencies of 341.5 MHz and 401.5 MHz allow both transmitters to be in FCC Space Allocated bands and provide convenient separation
Modulation	PSK/PM **FSK** MFSK PCM/FM DPSK	Predictable performance in multipath. Easy combination of diversity channels.
Transmitting Antenna	**Broad Beam Conical** Fan Beam Switched Array	Consistant with view angle requirements. Does not require capsule roll stabilization.
Transmitter Power Amplifier	**Transistor** Tube	Simpler and more reliable power supply.
Synchronization	NRZ Data **Split Phase Data** Separate Sync Signal	Strong clock component in data stream. No extra power required.
Commutation Programming Technique	Ring Counter Matrix **Interlaced Tubes** Table Look-Up	Efficiency and adaptability to stored programs.
Commutation Programming Memory	Hardwired **Stored Program**	Flexibility to experiment changes and adaptability to multi-mode operation.
Memory Devices	Magnetic Tape **Magnetic Cores** Plated Wire	Proven space hardware. Operation in an entry environment.

*Selected Approach

Figure 3.2-2

REQUIREMENTS IMPOSED ON ESP STRUCTURAL/MECHANICAL SUBSYSTEM

ESP SUBSYSTEM ELEMENT	REQUIREMENT
Accelerometer	• Locate as near entry C.G. as possible • Package size – 2.5 x 2.5 x 2.5 in. • Package weight – 2.0 lb
Entry Television Cameras	• Provide unobstructed forward field of view of 8° for narrow field camera and 50° for wide field camera parallel to roll axis within 0.1° • Minimize heat loss from camera electronics • Package size – 8 x 8 x 12 in. • Package weight – 14.0 lb.
Stagnation Point Port	• Provide port at 0° angle of attack stagnation point • Provide for installation of temperature sensor in this port. Size – 0.5 dia. x 2.0 in. Weight – 0.5 lb. • Provide for installation of pressure transducer near this port. Size – 2.0 dia. x 2.0 in. Weight – 1.0 lb. • Provide for obtaining uncontaminated gas samples for mass spectrometer. • Provide thermal protection for compartment behind ML closure
ESP Equipment	• Provide for installation in a single package. Provide access into package. • Minimize heat loss. • Provide for mass spectrometer vent • Provide for electrical connectors • Package size – 12.5 x 15.0 x 19.5 in. (includes structure and insulation) • Package weight – 14 lb (structure) 180 lb (total incl equipment, structure, & insulation.)
UHF Antenna	• Provide unobstructed 120° aft field of view parallel to roll axis within 1.0° • Package size – 15.0 dia. x 7.0 in. • Package weight – 6.0 lb
Base Region Port	• Provide for installation of temperature sensor in the base region (same size and weight as above) • Provide for installation of pressure transducer in the base region (same size and weight as above)

Figure 3.3-1

ESP STRUCTURAL ARRANGEMENT IN CAPSULE BUS

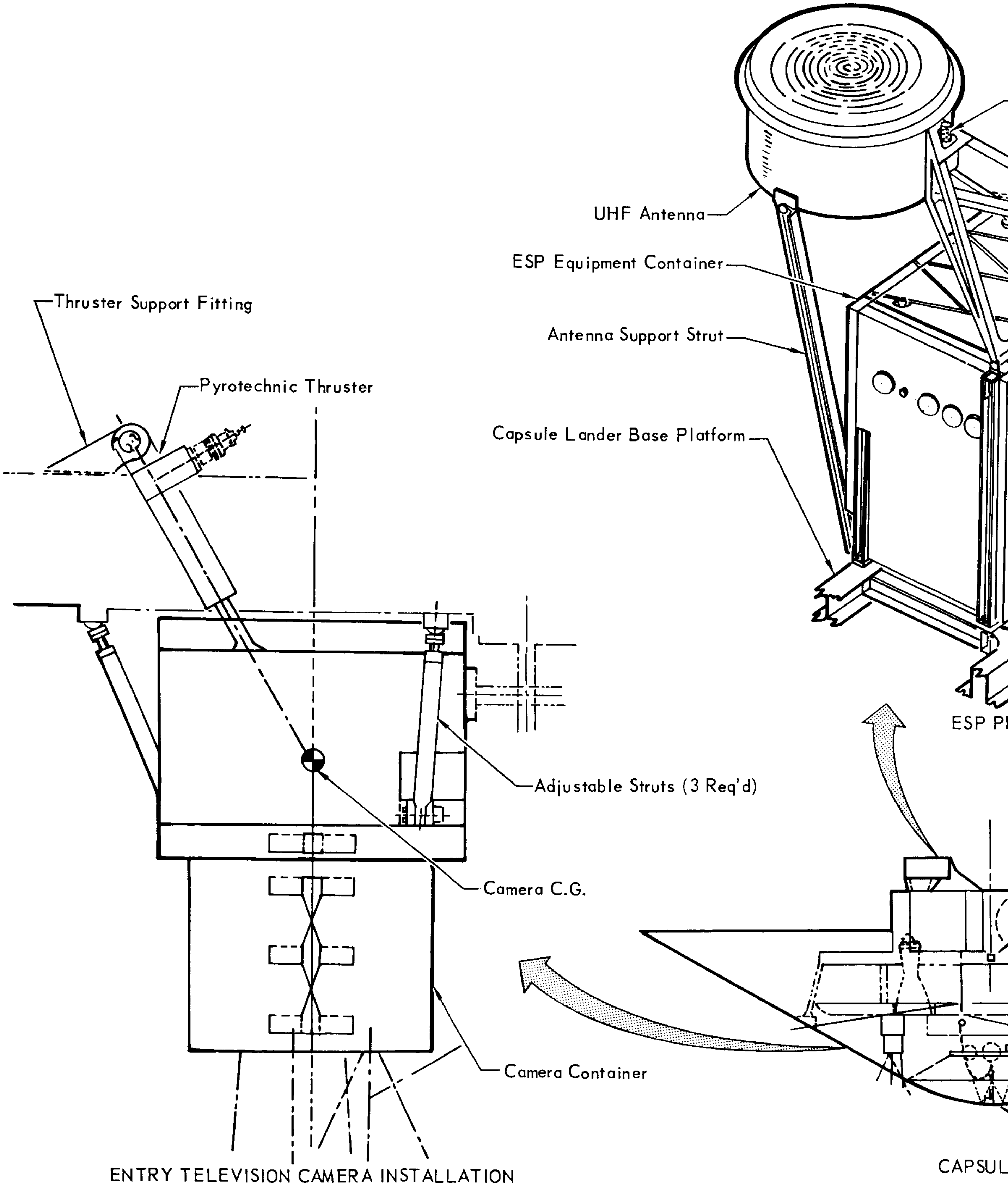
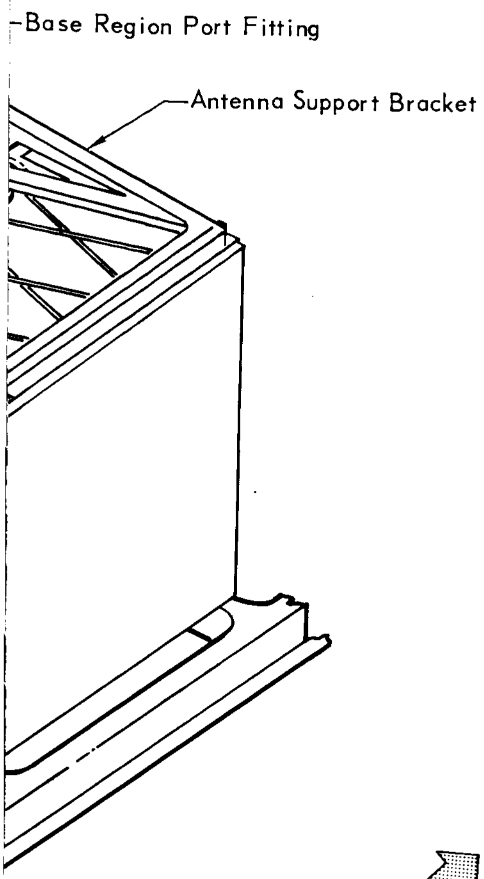
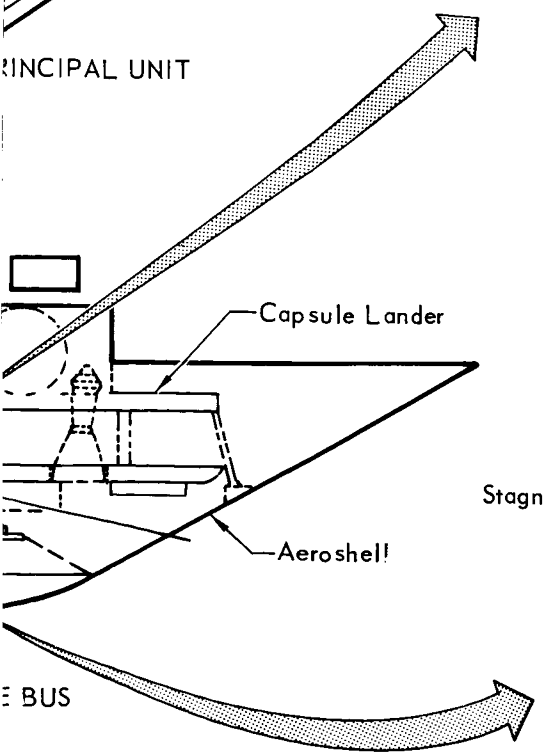


Figure 3.3-2

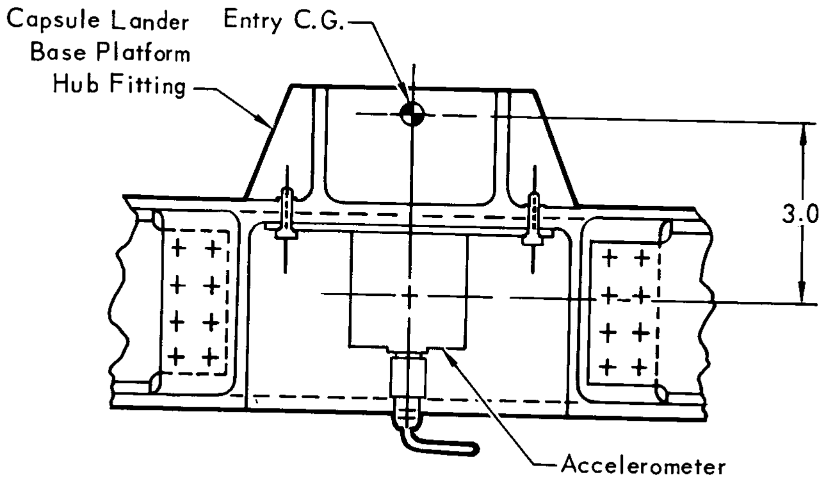
3-13 -1



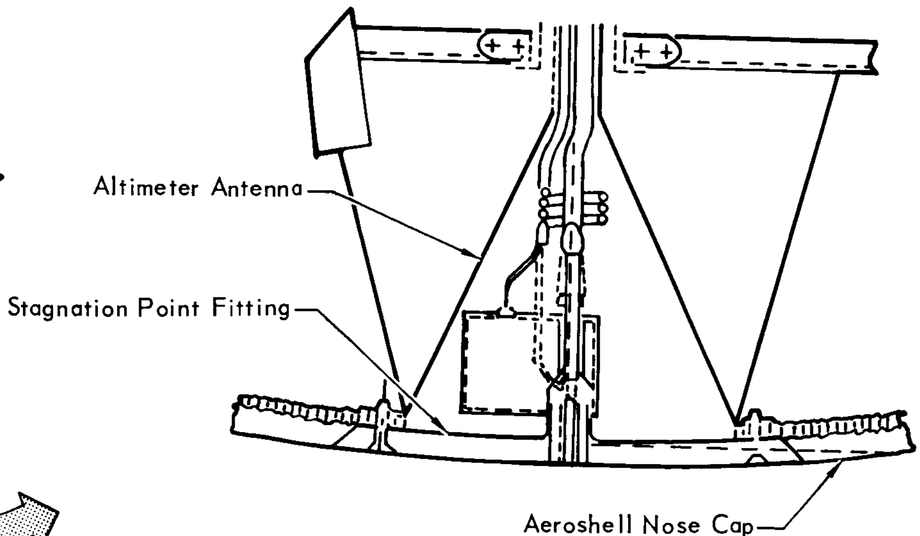
ʀINCIPAL UNIT



E BUS



ACCELEROMETER INSTALLATION



STAGNATION POINT FITTING

Stagnation Point Ports - A beryllium fitting which incorporates two concentric ports is located at the apex of the Aeroshell nose cap. The tube which makes up the inner port contains the temperature sensor and is routed to permit a bleed to the capsule base region. The mass spectrometers sample is brought through a restriction from this bleed line. The outer port is connected to the stagnation pressure transducer. The beryllium fitting serves as a heat sink as well as providing structural support.

ESP Equipment Mounting - A structural container with two aluminum honeycomb shelves incorporating inserts for attaching the individual components is provided for the equipment in the ESP principal unit. The lower shelf, which supports the heavier ESP components, forms the base of the container. The ends, top, and base are permanently joined together, while the front and back panels are removable to provide access. Fittings are provided at the four corners for mounting to the Capsule Lander base platform structure. Fittings are provided for attachment of the UHF antenna supports. Thermal control is provided by fiberglass blankets attached to the outboard surface of the container and by structural insulating blocks located between the mounting fittings and the container structure and between the mounting fittings and the platform structure.

UHF Antenna Support - The antenna is supported by a bracket and two struts attached to fittings provided on the ESP equipment container. Four single-bolted joints are used to minimize thermal transfer paths.

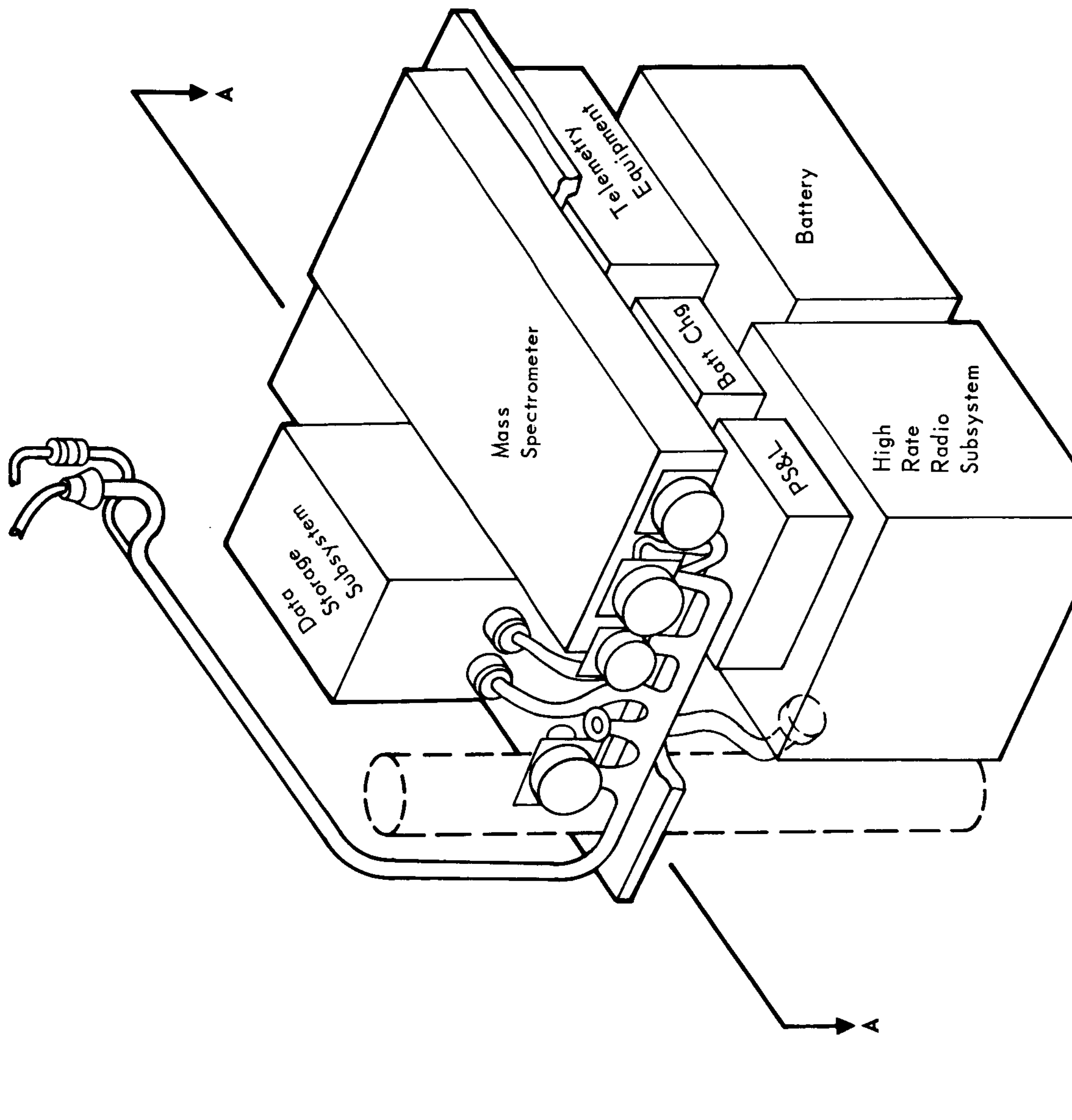
Base Region Port - A fitting incorporating a perforated tube is mounted on the antenna support bracket to support the temperature sensor. This fitting also provides the mounting for the pressure transducer, which utilizes the perforated tube for its input source.

3.4 PACKAGING AND CABLING - The ESP subsystems are packaged, installed, and interconnected within supporting structure in such a way as to insure compatible operations during all phases of the program, from initial assembly and installation to completion of the flight mission. The most important considerations include:

- o Achieving a compact geometrical configuration which can be readily integrated with the Capsule Bus and Surface Laboratory Systems and readily removed with minimum effect on the other systems.
- o Minimizing weight and volume
- o Providing thermal control during all mission phases.

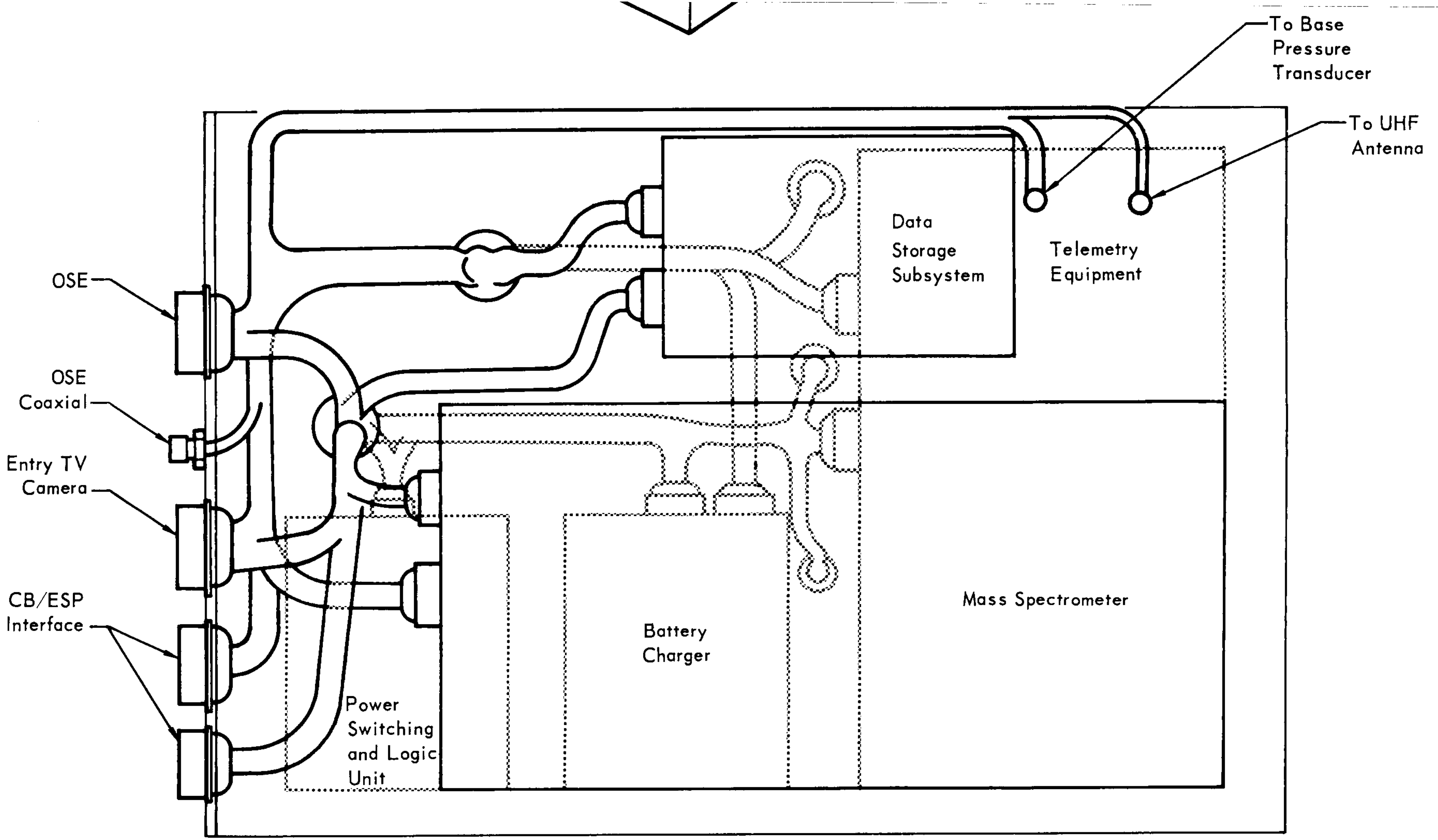
3.4.1 Equipment and Cabling Installation - The ESP principal unit and the cabling subsystem are installed as indicated in Figure 3.4-1. Some of the science equip-

ESP PRINCIPAL UNIT EQUIPMENT INSTALLATION



Fig

3-15-1



A–A

ure 3.4–1

3-15-2

ment must, in order to properly perform its function, be located in a specific area of the Capsule Bus. For example, the entry camera is installed on the impact footpad for unobstructed field of view, the accelerometer is located near the center of gravity, and other sensors are located in an instrument head on the aeroshell nose cap. The subassemblies that are not location-constrained are assembled in the ESP Principle Unit. This assembly includes both science and support subsystems in an integrated unit. The high-rate UHF antenna is supported by truss members. The subassemblies are installed in black box form with an integrated cable harness within a thermally insulated enclosure. Assembly of this equipment into an ESP Principle Unit permits integrated tests in both the bench and installed configurations.

3.4.2 Cabling - The cabling subsystem for the ESP includes the primary harnesses within the ESP Principal Unit and the cabling necessary to interface with the location-constrained equipment. To perform these functions, qualified standard hardware and established fabrication procedures have been selected. The preferred wire for all cabling is MIL-W-81381/1 "Kapton" wire. This wire was selected because of its high temperature and ETO compatibility, light weight, high resistance to abrasion, and high strength. All connectors are per MIL-C-38999, utilizing crimp contacts except where distinct advantages are offered by soldered contacts in special applications (e.g. hermetically sealed connectors).

The interconnecting cabling is sealed in its entirety by potting all wire terminating devices. Potting provides wire bundle support at the wire terminations, with minimum weight and volume, as well as providing environmental protection. The cabling subsystem is designed to allow flexibility of circuit design and subassembly relocation during development phases and for future missions.

3.4.3 Packaging - The preferred packaging approch for electronics equipment is controlled-geometry black boxes. This technique utilizes functional modularized elements, combined to form one package. The elements are designed as discrete devices which have volumes consistent with functional requirements and dimensions compatible with the geometry of the ESP Principal Unit. This approach was selected because (1) the relatively large and inflexible dimensional requirements of the Mass Spectrometer tend to dictate overall assembly dimensions that are incompatible with standardized assembly packaging, (2) recommended standardization for future missions is limited to the TV and telemetry, and (3) this approach will result in minimum cost.

The black box approach is consistent with all qualified internal packaging techniques. The maximum use of potted and encapsulated modules with welded or wire-wrapped (rather than soldered or friction type) interconnects is preferred for internal packaging. The critical elements of internal packaging include circuitry layout, thermal control (including differential expansion), and mechanical integrity. As the equipment is developed, these items are controlled by design approval and rigorous adherence to fabrication procedures.

3.4.4 Trade Study Summary - Trade studies were conducted on equipment form factors, internal packaging, wire, general purpose connectors, and wiring design and fabrication techniques. Figure 3.4-2 summarizes these studies. Further details are presented in Part E, Section 4.4.

3.5 THERMAL CONTROL - The ESP thermal control subsystem maintains the ESP Equipment Container and ESP Capsule Bus-mounted equipment within allowable temperature ranges throughout all mission phases prior to landing. The preferred design concept uses a combination of thermal control coatings, insulation, and electrical heaters.

3.5.1 Requirements - From launch until lander separation from the Aeroshell, the subsystem operates within the overall temperature environment provided by the Capsule Bus thermal control subsystem. The Capsule Bus temperature averages -140°F prior to entry and has local areas within the Aeroshell up to 800°F at the end of entry heating. The equipment and instruments inside the ESP principal unit must be maintained within a range of 0° to 60°F during the inoperative period. During the operative period, the limits are 50° to 100°F. These requirements are constrained by the ESP batteries.

The temperature requirements for the descent TV camera are -4° to 140°F during the inoperative period, and -40° to 104°F during the operative period. For the stagnation pressure transducer, the limits are -324° to 250°F throughout the mission, thus no heater is necessary.

3.5.2 Thermal Control Subsystem Description and Operation - The thermal control subsystem consists of electrical heaters, insulation, and thermal control surfaces. Small electrical heaters provide 2, and 3 watts of power to the ESP Equipment Container, and descent TV camera, respectively. The heaters are located within the insulated region of the packages and operate continuously when power is available from the spacecraft. The insulation (1) minimizes the heater power required, (2) provides contingency for later equipment design changes, (3) prevents excessive cooldowns of equipment during power interruptions at spacecraft midcourse corrections, and (4) prevents overheating during the brief Martian entry period.

INTERCONNECTION EQUIPMENT AND TRADE STUDY SUMMARY

COMPONENT TECHNIQUE	ALTERNATES	MERITS OF SELECTED APPROACH
Equipment Form	**Standardized Subassembly** Random Geometry Black Box Drawer Hinged Bay Structurally Integrated	Flexibility, Accessibility, Test Capability, Simplicity
	Black Box Standardized Subassembly Structurally Integrated	Compatibility with integrated assembly leading to ESP module, minimum weight and volume cost.
	Modularized Assemblies	Accessibility, flexibility test capability
	Black Box	Guidance Equipment near C.G.
Internal Packaging	Circuit Board Modules Embedded (Cordwood) Modules Radio Frequency Packaging	Selection Based on Function and Application and Components Utilized.
Wire	**MIL-W-81381/1 Kapton (7 mil)** MIL-W-81381 Kapton (5 mil) MIL-W-16878 Teflon MIL-W-81044/3 Kynar Raychem Thermorad	Compatible with ETO and heat sterilization, strong and tough, weight and volume savings.
General Purpose Connectors	MIL-C-38999 MIL-C-26482 MIL-C-81511 NAS-1599	High temperature operation, environmental seal, low weight and volume, crimp contacts
Interconnecting Wiring – Seal	**Potting Seal** Grommet Seal Unsealed	Environmental sealing and mechanical support
– Contact Termination	**Crimp** Solder	Reliable wire/contact transition, removable
– Wire Covering	**None** Jacket Sleeving	Light weight, flexibility for modifications
– Multi terminations	**Terminal Junction Modules** Stud Terminal Strips	Low weight and volume, flexibility of terminations.

Preferred Concept

Figure 3.4-2

The insulation material consists of low density fiberglass fibers, bonded with silicone. The insulation thickness required is 0.75 inches (3.5 lb) on the ESP Equipment Container, 1.0 inch (1.0 lb) on the descent TV package, and 1.0 inch (0.4 lb) on the stagnation pressure transducer. Low emittance gold coating is used on the outer surfaces of the insulation to assist in heat retention during cruise and reduction of heat input during entry. The insulation, combined with the low emittance coatings, provides a cool-down rate of 2°F/hour during power interruptions when spacecraft trajectory corrections are made.

SECTION 4

CAPSULE BUS/ENTRY SCIENCE INTERFACE

Definition of the atmospheric properties of Mars and acquisition of descent television images are major science goals of the Entry Science Package. Potential errors and uncertainties in quality and accuracy of entry science data due to mission and operational characteristics of the Capsule Bus must be determined and evaluated.

Errors in execution of the deorbit functions contribute to uncertainties in the predicted entry conditions. Inaccurate knowledge of the Martian atmosphere, and its effect on the Capsule's entry trajectory, adds to these entry condition uncertainties to produce the final landing site uncertainties. Telemetry of sensor data provides for post flight reduction of uncertainties in all phases of the mission.

The Capsule Bus System is designed to insure a safe landing of the VOYAGER Lander on the surface of Mars. Design accommodations in the Capsule Bus System to insure a safe landing on Mars are not necessarily optimum for the Entry Science Package. The Capsule Bus design was, however, modified to improve entry television and acquisition of atmospheric property data by using a non-ablative nose cap, but only after it had been determined that this design concept did not compromise the safe landing on Mars. Lighting constraints for high quality ESP television images require either a local morning or evening landing. The Capsule Bus mission profile in this case has been constrained by this requirement. The morning landing has been selected as the preferred mission profile. Attitude control, Aeroshell heat protection, Aeroshell separation and the altimeter were all designed to insure a safe landing and to obtain maximum entry science data. Flow field studies of the Aeroshell aid in minimizing the uncertainties of atmospheric measurements by positioning sensors in optimum locations. Rate damping of the attitude control system improves the quality of both the television images and atmospheric data measurements.

The following paragraphs discuss the interactions of the Capsule Bus and Entry Science Package and the resulting effect on the entry science data.

4.1 FLIGHT VARIATIONS FOR ESP OPERATIONS - Atmospheric entry of the Capsule Bus, containing the Entry Science Package, has a wide spread of flight conditions. The atmosphere of Mars is broadly defined. The spread of model atmospheres, VM-1 through VM-10, represents the expected range of ambient conditions, (See Volume IV, Part C Section 3), which will be measured by the ESP. Four of the model atmospheres (VM-7, VM-8, VM-9, and VM-10) are extreme models in terms of temperature, density, and pressure and produce extremes in vehicle airloads, stagnation heating rate, descent time, and range (described in Volume II, Part B, Section 2.3.6).

The trajectory of the Capsule Bus within these model atmospheres will depend on the orbit size, location of the de-orbit maneuver and the applied de-orbit velocity vector. An entry corridor, Figure 4.1-1, was used to bound the design of the Capsule Bus. Entry velocities from 13,000 ft/sec to 15,000 ft/sec and entry flight path angles from 20° to vacuum graze have been considered as the design limit.

For the four extreme atmospheres and the four extreme entry conditions, the altitude of the Capsule Bus (without added retardation) decreases rapidly for the fast, steep entry, Figure 4.1-2. The flight path angle, Figure 4.1-3, becomes more shallow initially and then rapidly becomes steep. The most shallow atmosphere (VM-8) has the shortest flight time because the Capsule does not decelerate (Figure 4.1-4) until it reaches low altitudes. The peak acceleration (Figure 4.1-5) of 21g occurs with VM-8.

Figure 4.1-6 presents the Mach Number time history variation for the critical atmospheres and entry conditions.

Calculated stagnation pressure and stagnation temperature are presented in Figures 4.1-7 and 4.1-8 respectively. A value of peak stagnation pressure on the order of 400 lb/ft^2 is predicted. Peak stagnation temperature at Mach Number 5 is not expected to exceed 1300°K.

The time of day for the entry trajectory does not affect the flight conditions but is of significance to the descent television. To obtain shadows of the general terrain from a descending Capsule, the solar angle must be low to cast long shadows for photographic interpretation. A constraint for landing in the range of 15° to 30° from the terminator was used for Capsule mission profile analyses to provide the desired lighting for the ESP. Either a morning or evening landing will fulfill this requirement. However, as discussed in the analysis of the Capsule mission (Volume II, Part B 2.4), the morning landing is preferred for the Capsule Bus and the Surface Laboratory.

4.2 DE-ORBIT - Accurate determination of the Martian atmosphere demands accurate reconstruction of the Capsule Bus atmospheric trajectory. To this end, it is desirable to make the uncertainties in entry conditions as small as possible. This section describes those features of the preferred design which relate to the reduction of uncertainty in the de-orbit conditions, and thus reduction in entry uncertainties. Alternative de-orbit mechanizations are discussed in Part E, Section 5.2.

ENTRY INITIAL CONDITION ENVELOPE

h_e = 800,000 ft
Graze Angle Defined as That Angle Which
Produces $\gamma = 0$ at h = 0

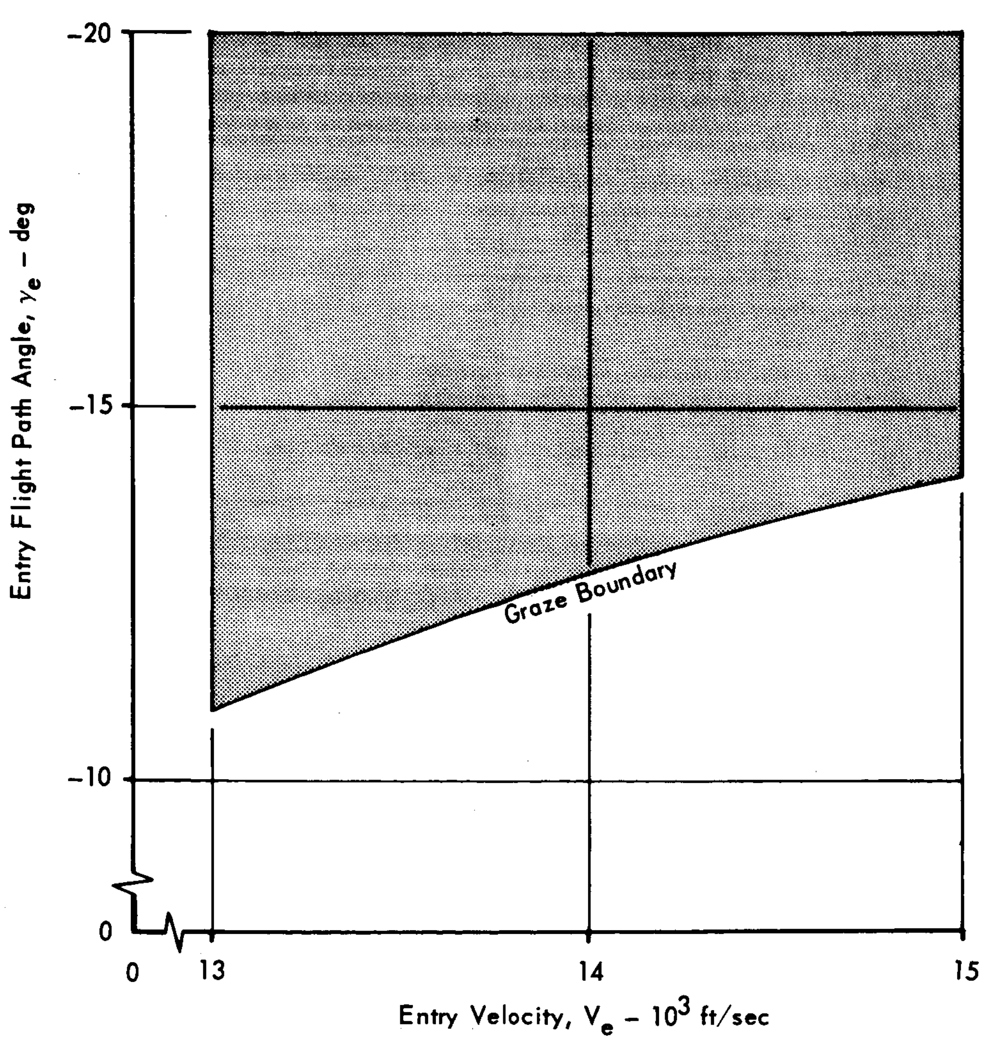


Figure 4.1-1

ALTITUDE TIME HISTORY

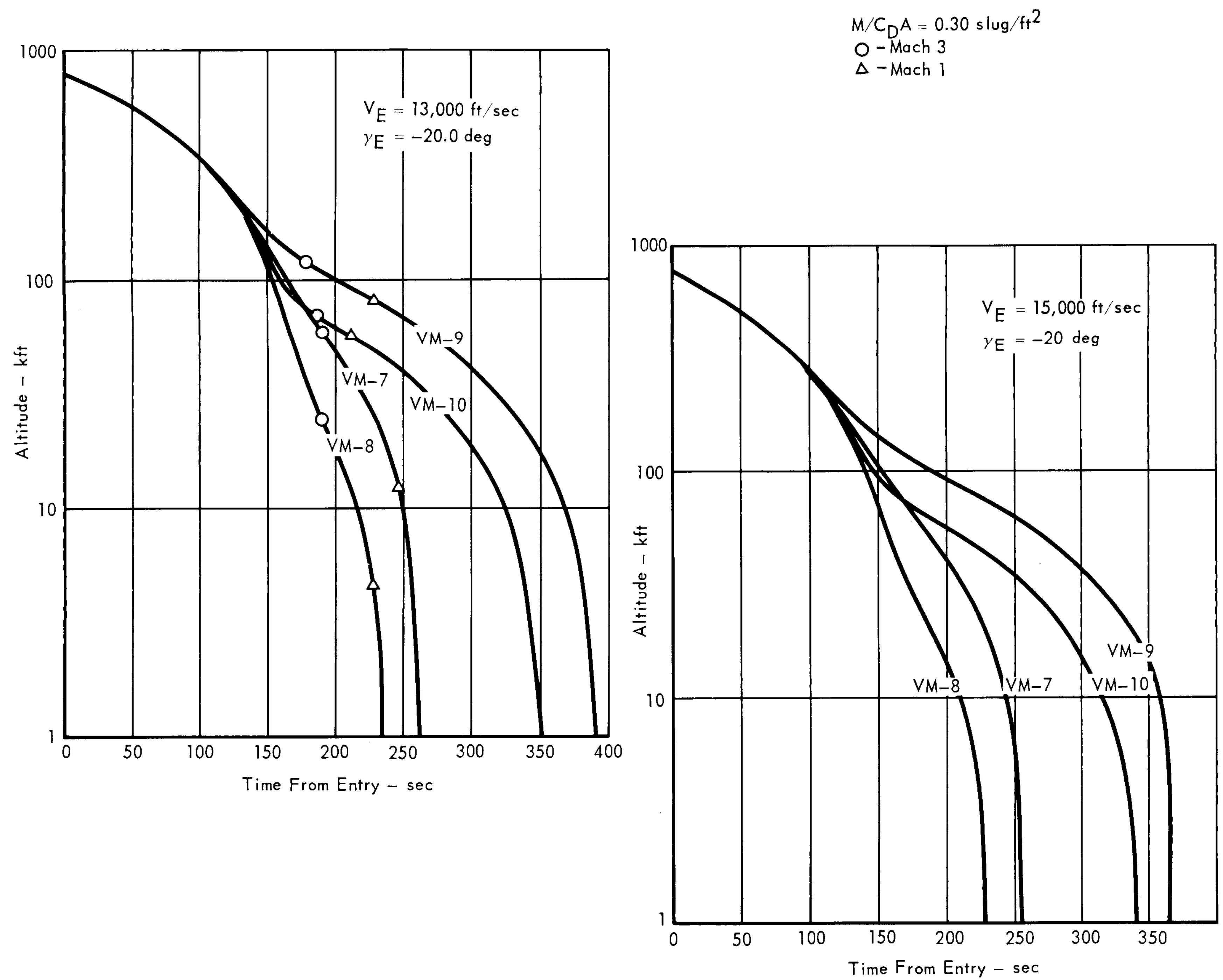
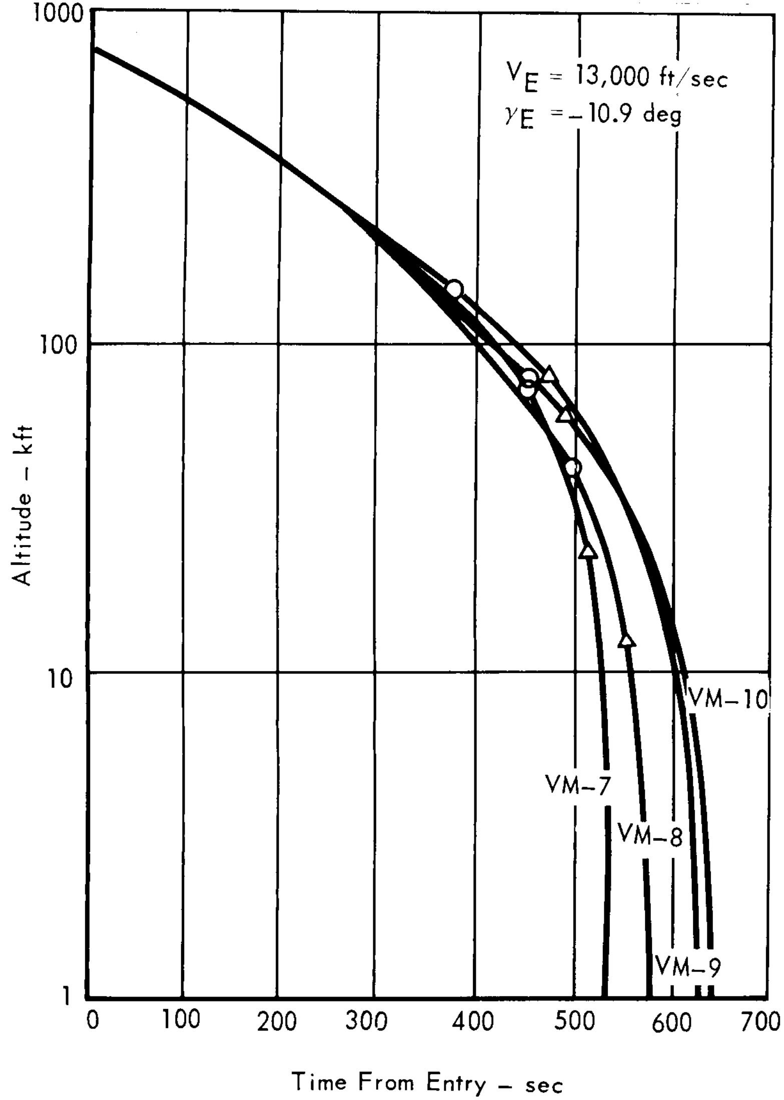
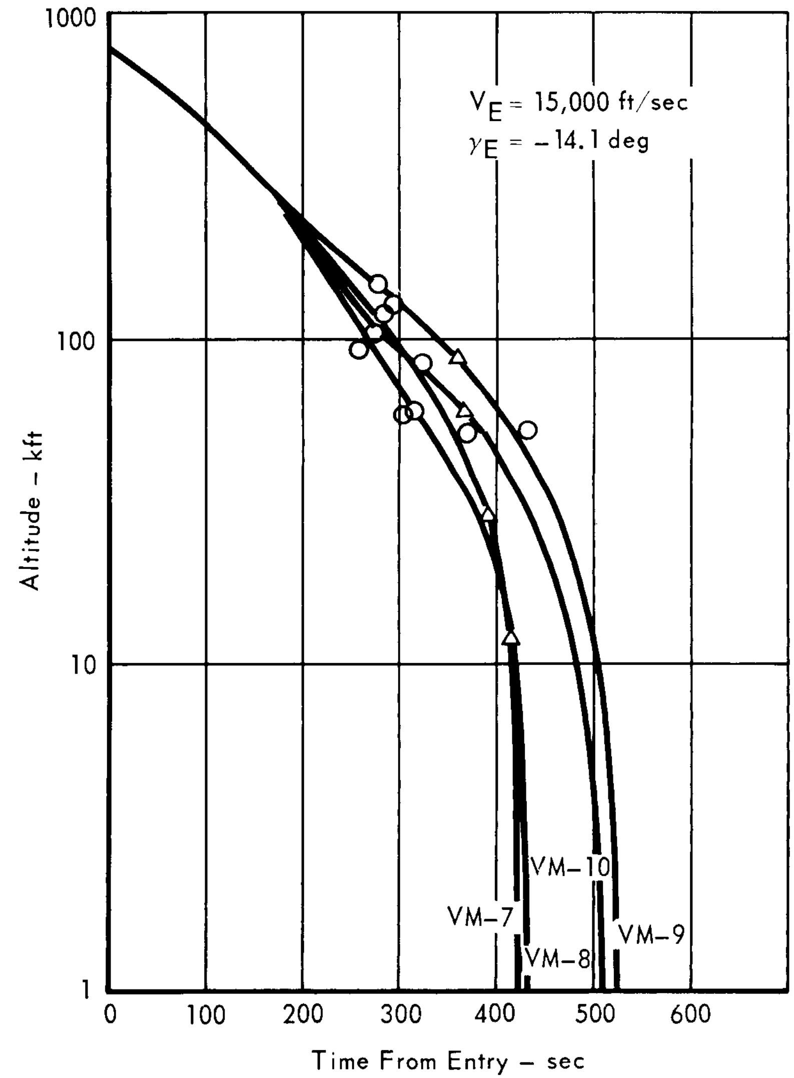


Figure 4.1–2



V_E = 13,000 ft/sec
γ_E = -10.9 deg
VM-7
VM-8
VM-9
VM-10
Altitude - kft
1000
100
10
1
0
100
200
300
400
500
600
700
Time From Entry - sec



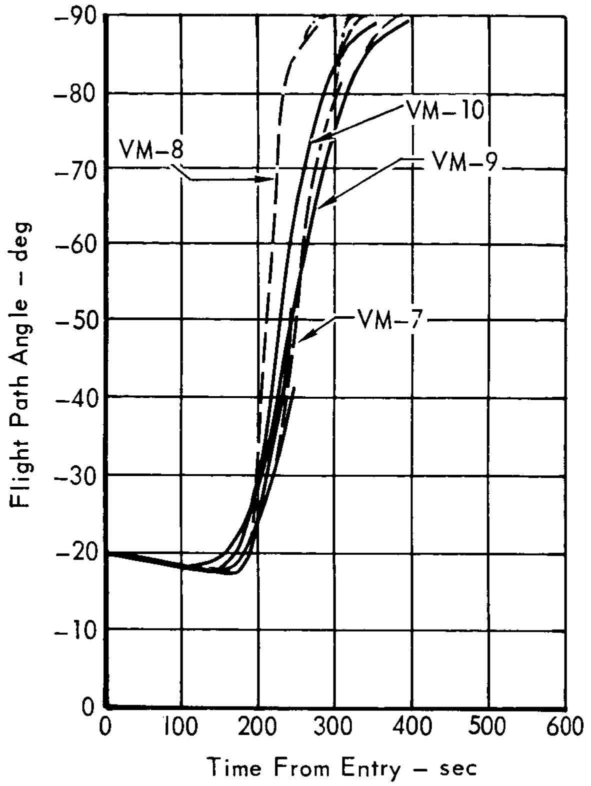
V_E = 15,000 ft/sec
γ_E = -14.1 deg
VM-7
VM-8
VM-9
VM-10
Altitude - kft
1000
100
10
1
0
100
200
300
400
500
600
Time From Entry - sec

2-4-4

FLIGHT PATH ANGLE TIM

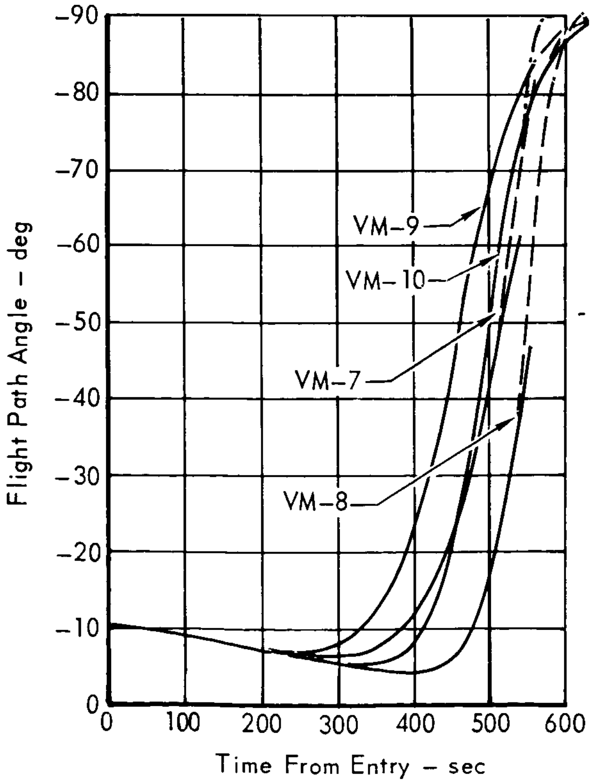
$m/C_DA = 0.3$ slug/

$h_e = 800,000$ ft



$\gamma_e = -20°$

$V_e = 13,000$ ft/sec

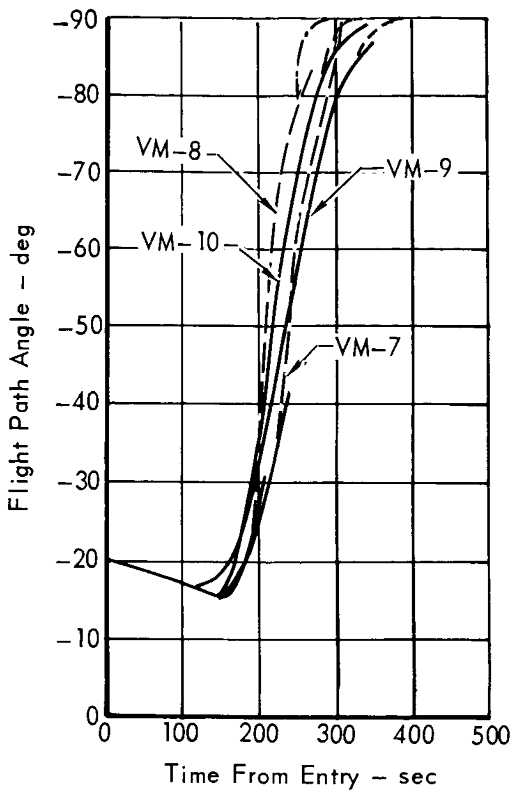


$\gamma_e = -10.9°$

$V_e = 13,000$ ft/sec

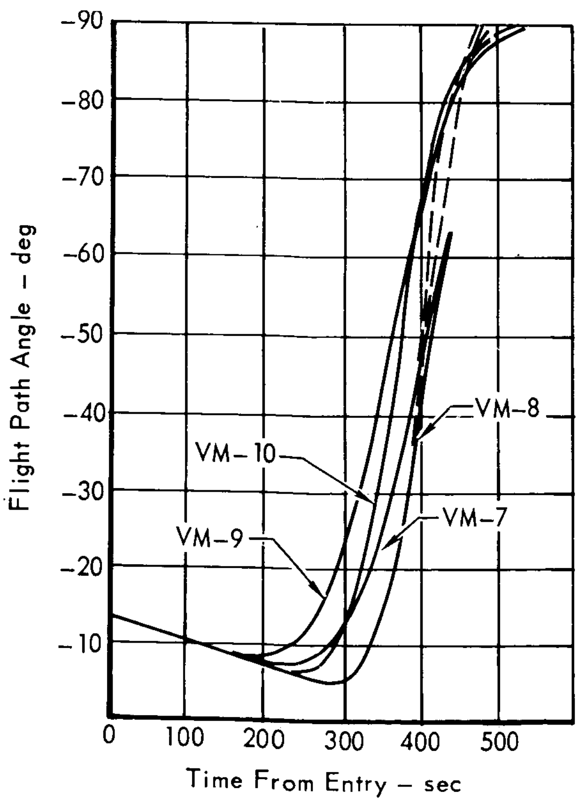
Figure 4.1-3

E HISTORIES
ft²



$\gamma_e = -20°$

$V_e = 15{,}000$ ft/sec



$\gamma_e = -14.1°$

$V_e = 15{,}000$ ft/sec

– – – – Flight Path Angle on Parachute

——— Flight Path Angle During Terminal Descent

4-5-2

VELOCITY TIME HISTORY

CAPSULE BUS
NO ADDITIONAL DECELERATION

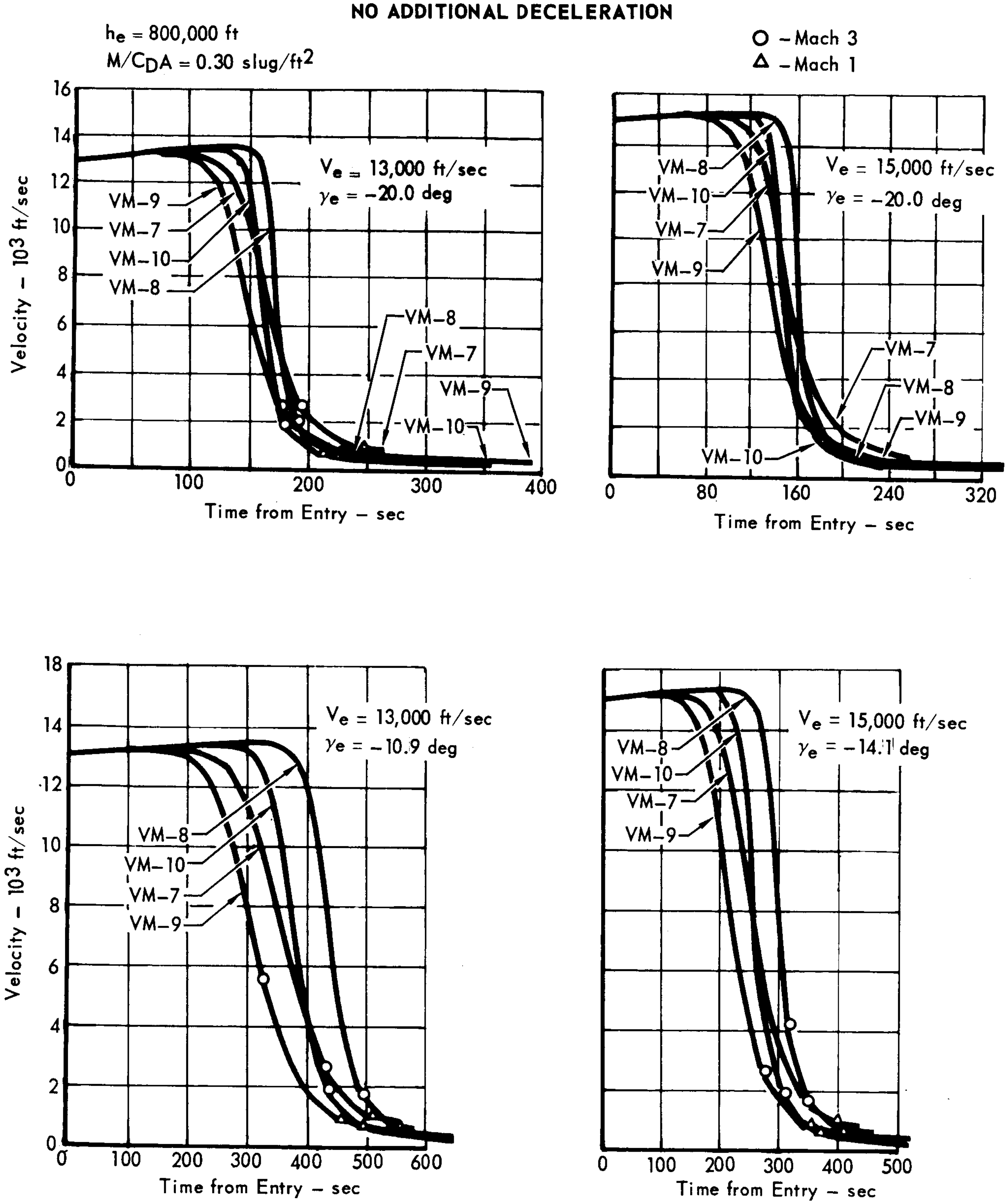
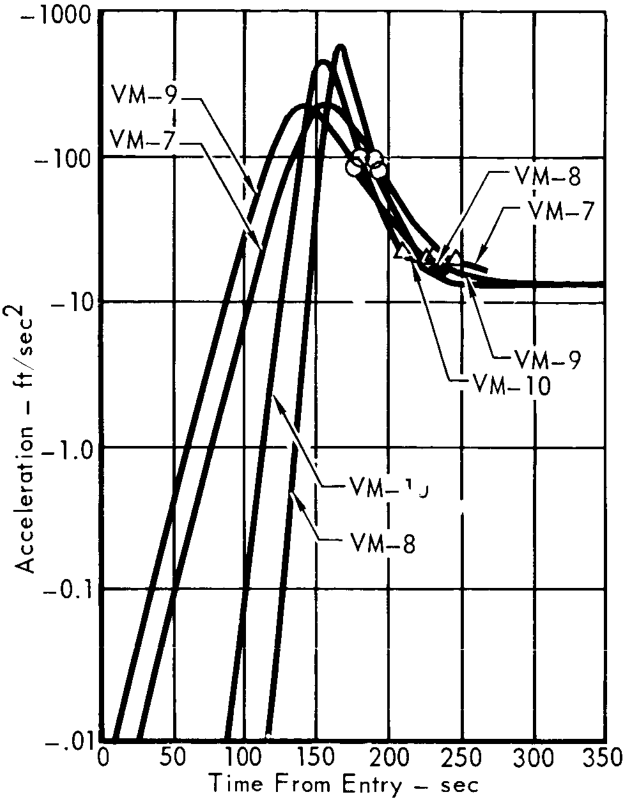


Figure 4.1–4

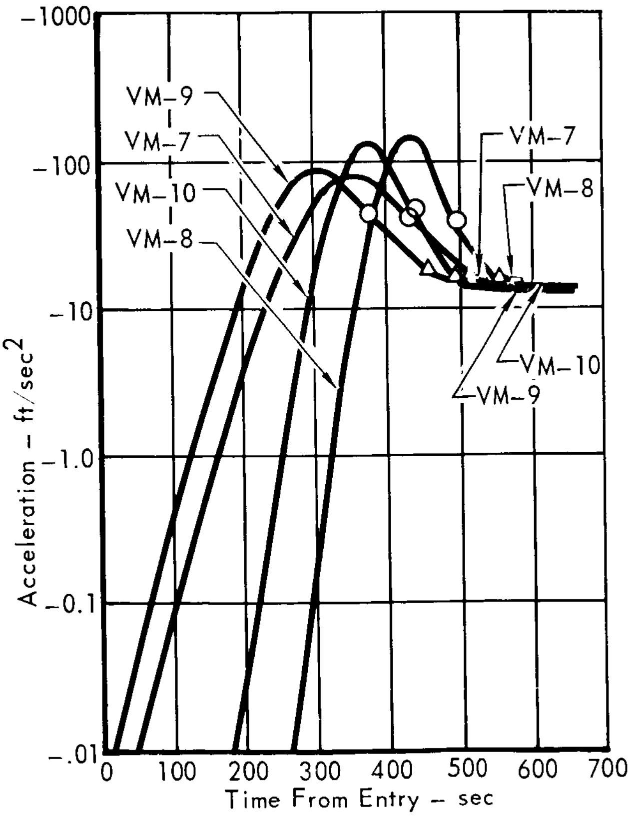
ACCELERATION TIME H

$h_e = 800,000$ ft

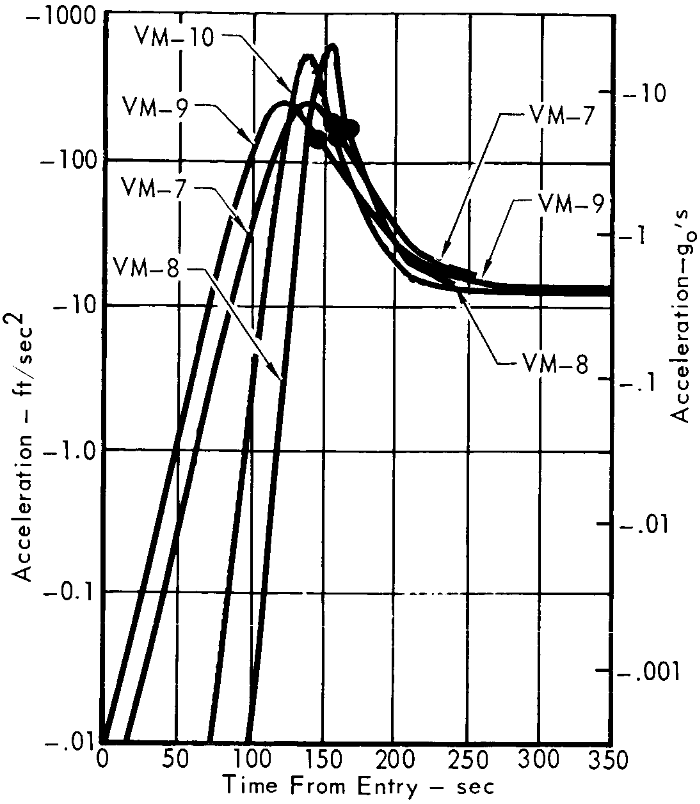
$M/C_DA = 0.3$ slug/ft^2



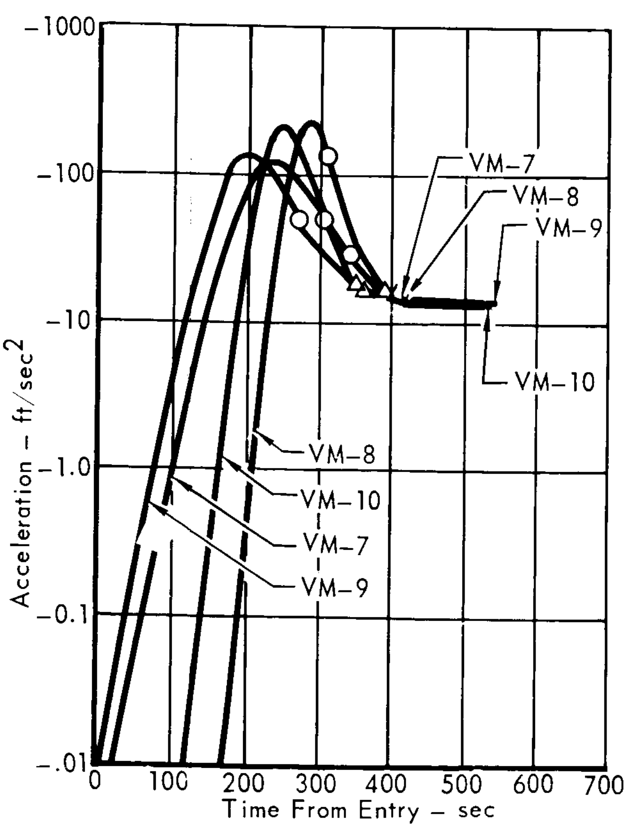
$V_e = 13,000$ ft/sec
$\gamma_e = -20.0$ deg



$V_e = 13,000$ ft/sec
$\gamma_e = -10.9$ deg



V_e 15,000 ft/sec
γ_e -20 deg



V_e 15,000 ft/sec
γ_e = -14.1 deg

● Mach 5
○ Mach 3
△ Mach 1

Figure 4.1-5

MACH NUMBER TIME HISTORY

CAPSULE BUS

NO ADDITIONAL DECELERATION

$h_e = 800{,}000$ ft

$M/C_DA = 0.30$ slug/ft^2

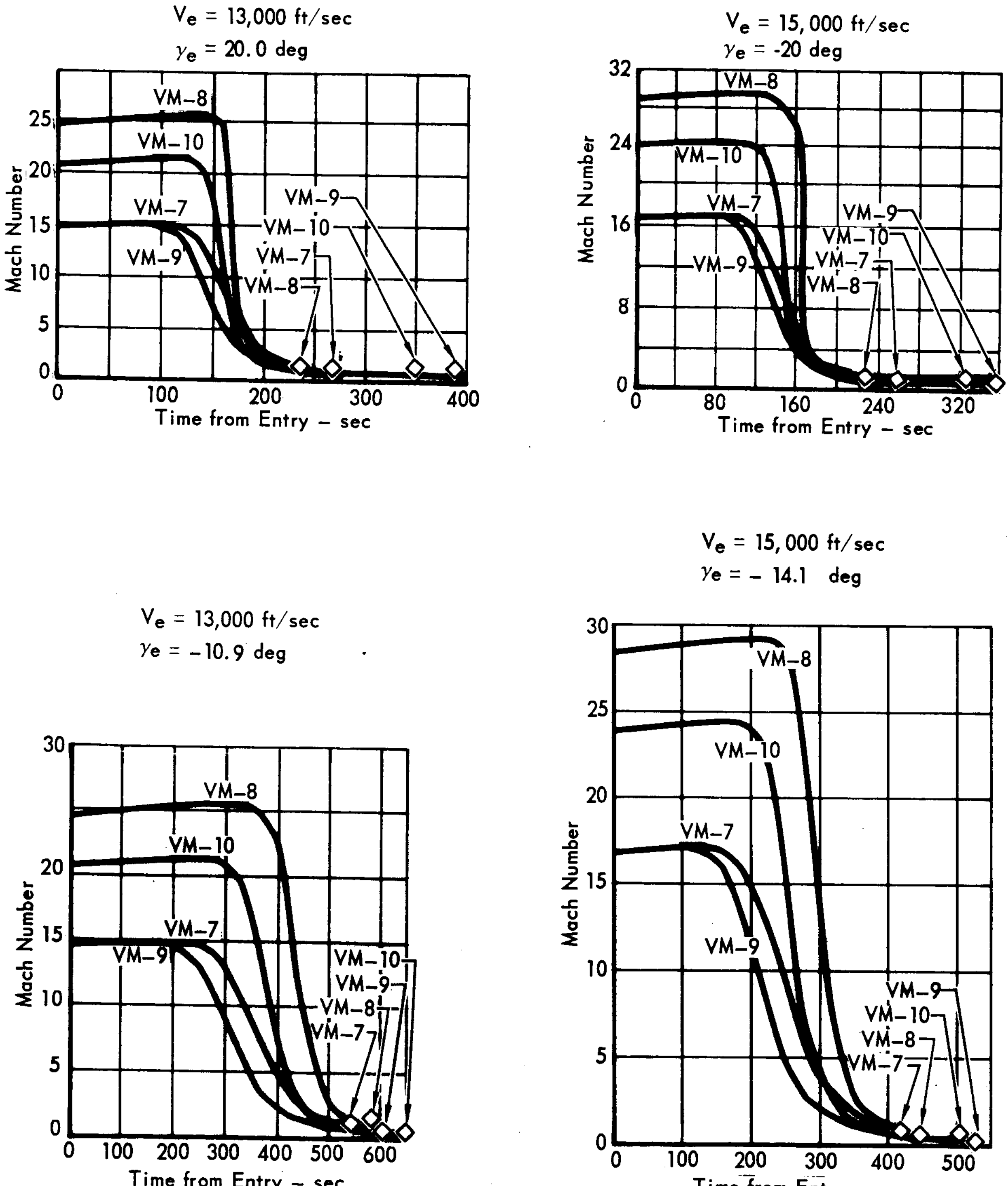
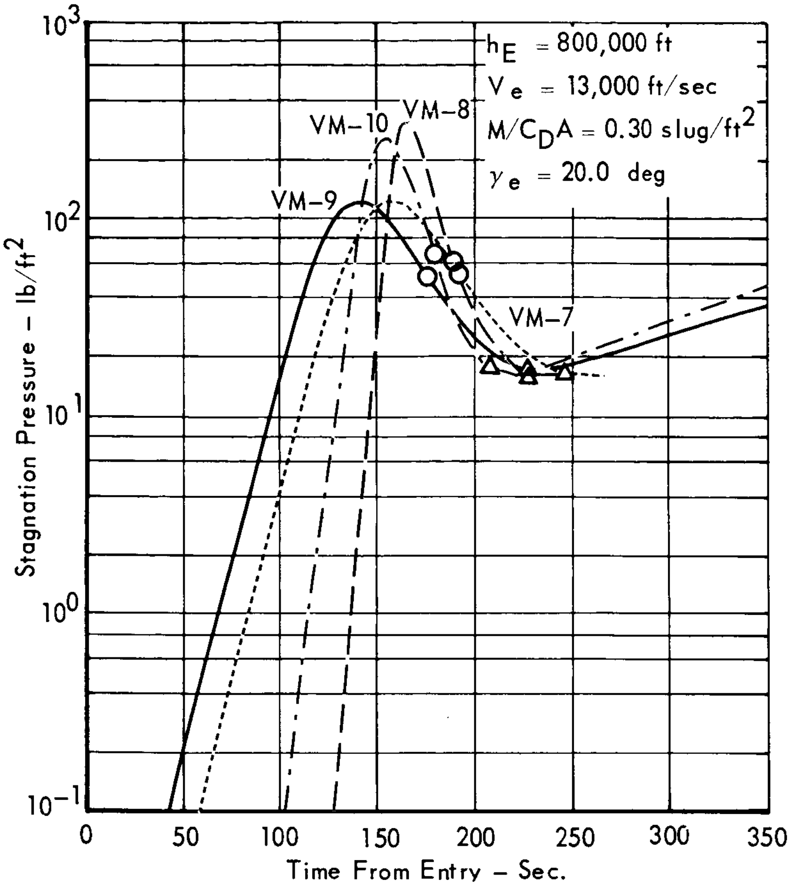


Figure 4.1–6

CALCULATED STAGNATION PRESSURE TIME HISTORY



● Mach 5
○ – Mach 3
△ – Mach 1

$$\text{– – –}\quad \frac{P_{STAG}}{P} = \left[\frac{\left(\frac{\gamma+1}{2}M^2\right)^{\gamma}}{\frac{2\gamma}{\gamma+1}M^2 - \frac{\gamma-1}{\gamma+1}}\right]^{\frac{1}{\gamma-1}}$$

$$\cdots\cdots\quad \frac{P_{STAG}}{P} = \left[1 + \frac{\gamma-1}{2}M^2\right]^{\frac{\gamma}{\gamma-1}}$$

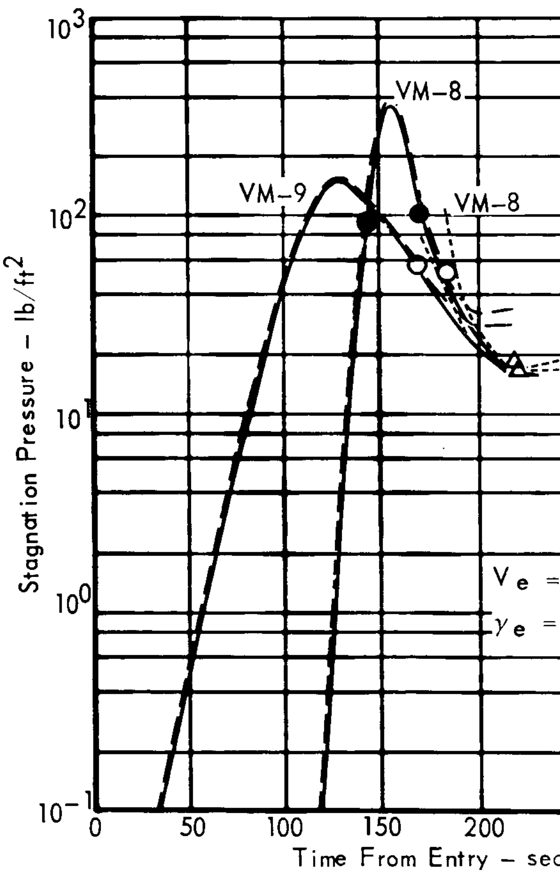
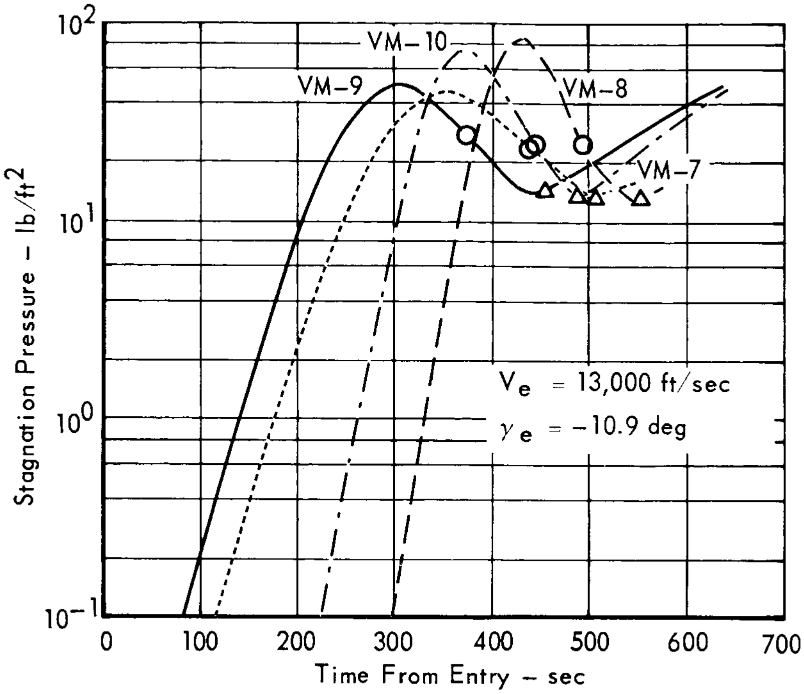
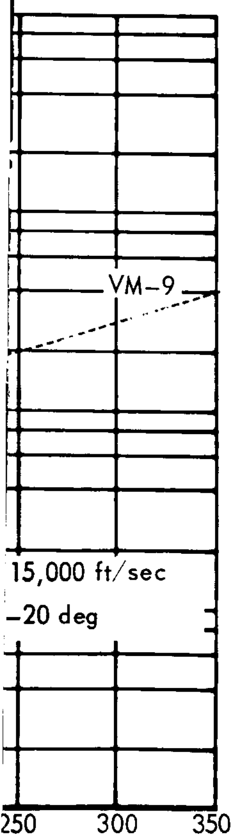
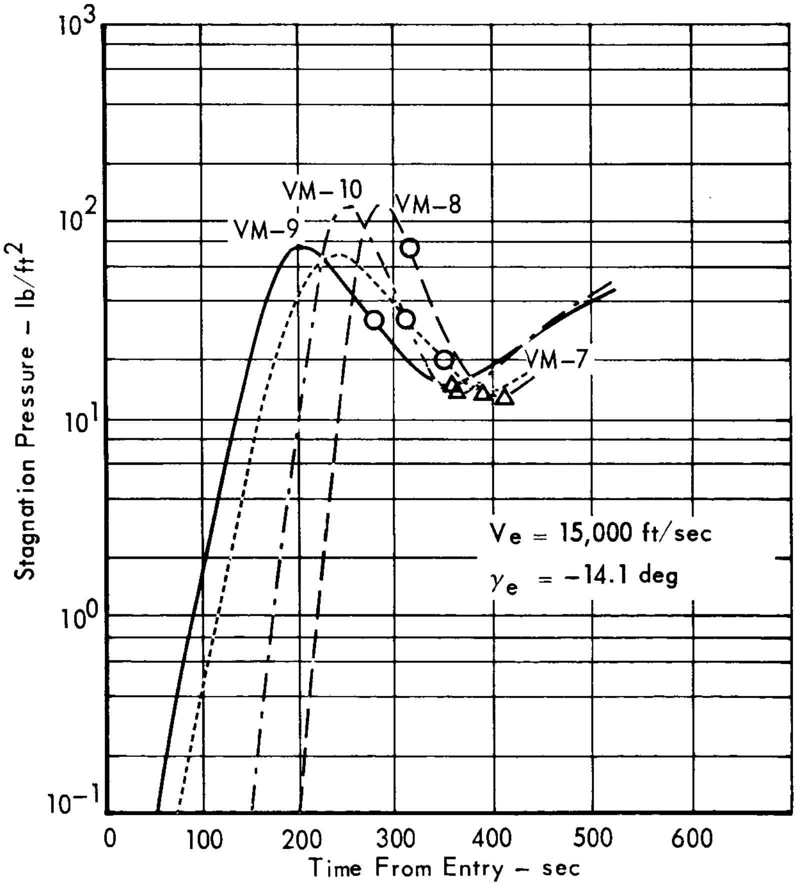


Figure 4.1–7


VM-10
VM-9
VM-8
VM-7
V_e = 13,000 ft/sec
γ_e = −10.9 deg
Stagnation Pressure − lb/ft^2
10^2
10^1
10^0
10^{-1}
0 100 200 300 400 500 600 700
Time From Entry − sec


VM-9
15,000 ft/sec
−20 deg
250 300 350

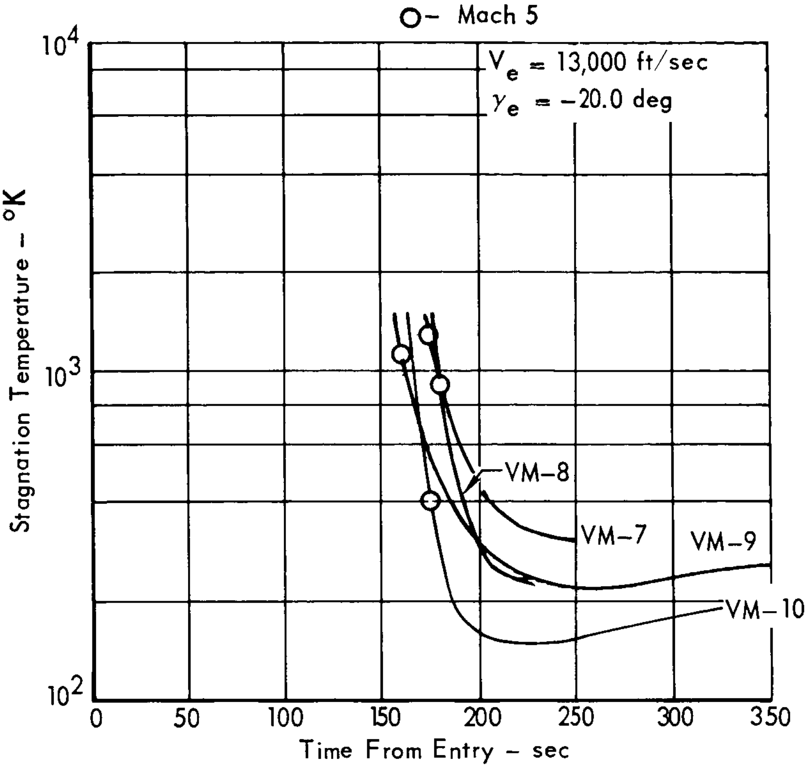

VM-10
VM-8
VM-9
VM-7
V_e = 15,000 ft/sec
γ_e = −14.1 deg
Stagnation Pressure − lb/ft^2
10^3
10^2
10^1
10^0
10^{-1}
0 100 200 300 400 500 600
Time From Entry − sec

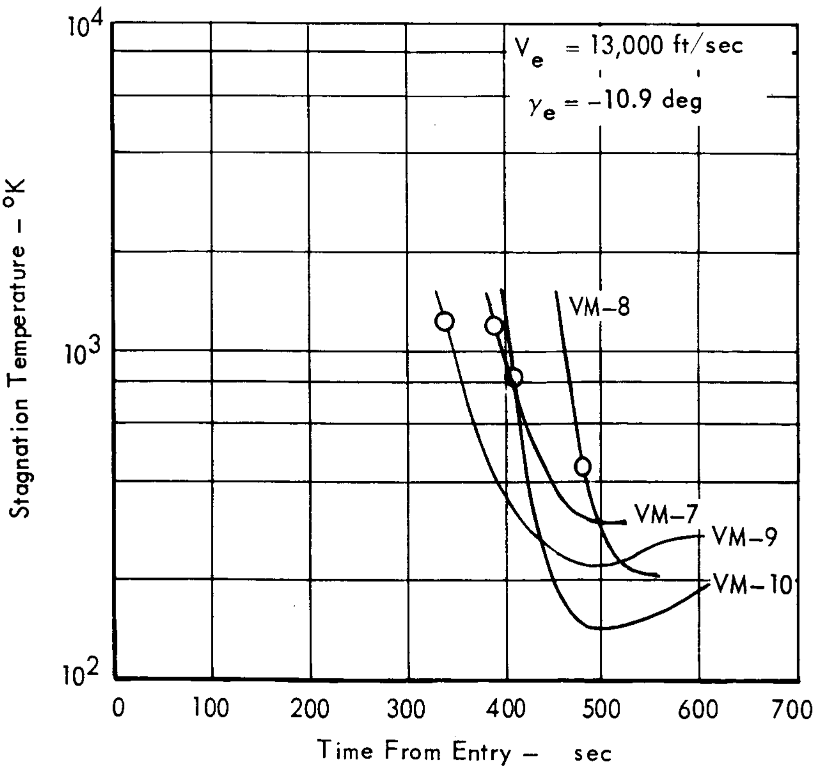
4-9-2

CALCULATED STAGNATION TEMPERATURE TIME HISTORY

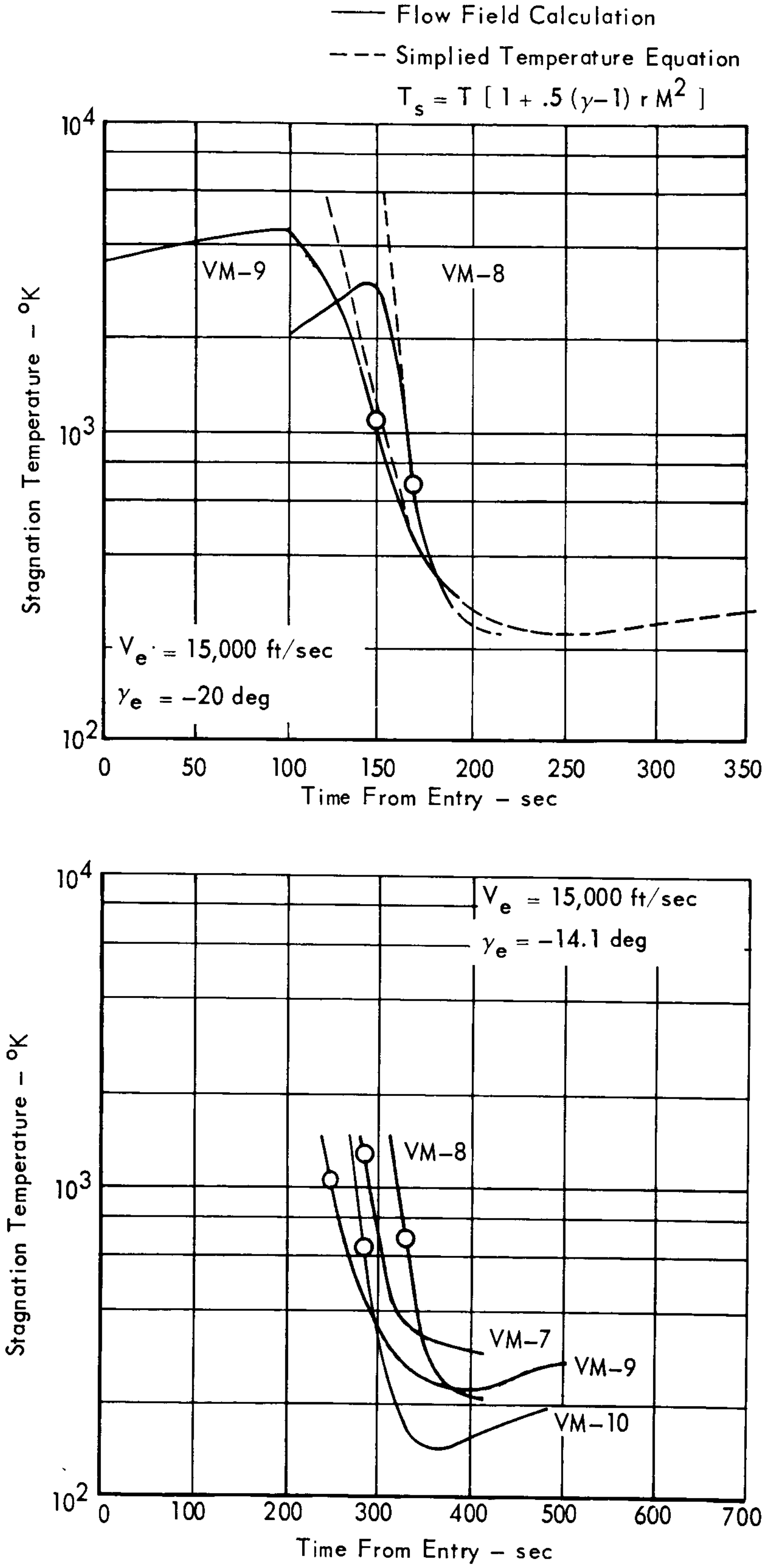
$h_e = 800,000$ ft
$M/C_DA = 0.30$ SLUG/FT^2

O– Mach 5





4-10-1



Flow Field Calculation
Simplied Temperature Equation
$T_s = T\ [1 + .5\,(\gamma - 1)\ r\,M^2\ ]$
10^4
10^3
10^2
Stagnation Temperature – °K
VM-9
VM-8
V_e = 15,000 ft/sec
γ_e = –20 deg
0
50
100
150
200
250
300
350
Time From Entry – sec
10^4
10^3
10^2
V_e = 15,000 ft/sec
γ_e = –14.1 deg
Stagnation Temperature – °K
VM-8
VM-7
VM-9
VM-10
0
100
200
300
400
500
600
700
Time From Entry – sec

Figure 4.1-8

4.2.1 Preferred Design and Mechanization - The Capsule Bus attitude reference is provided by three orthogonal, strapdown, pulse-rebalanced gyros, and coordinate conversion by the Guidance and Control Computer (GCC). These gyros are uncaged shortly before separation of the Capsule Bus from the Flight Spacecraft. During the 20 minute coast from separation to de-orbit, the reaction control system (RCS) orients the Capsule Bus to the desired de-orbit attitude, and maintains this attitude within a 2° deadband.

The de-orbit velocity decrement, ΔV, is provided by a 6000-lb solid rocket motor. Motor cutoff is commanded when the integrated axial acceleration reaches the desired value, and is accomplished by nozzle blowoff. Capsule Bus attitude is controlled by the RCS to within an 0.25° deadband during de-orbit motor burn.

The resultant system uncertainties in pointing angle and velocity magnitude are shown in the first column of Figure 4.2-1. A means for reduction of these uncertainties is "de-orbit monitoring," i.e., telemetry of engineering data which will allow post-flight reconstruction of the de-orbit kimematics. The telemetry allocation for de-orbit monitoring is 20 bits five times per second, over the period from CB/FS separation until computer shutdown (de-orbit + 11 minutes). An optimal "mix" of gyro, accelerometer and RCS outputs, plus ignition and cutoff event markers and status,data is to be commutated into this gross allocation.

Error minimization - Before separation of the Capsule Bus, the Planetary Vehicle maintains attitude by its own RCS, with an 0.5° deadband. At the instant the Capsule Bus gyros are uncaged, they register zero angles, but their actual displacement from the angular reference of the Flight Spacecraft is in general non-zero. somewhat less than the 0.5° which is used as the worst-case initial angular reference error without monitoring. Monitoring the Flight Spacecraft gyros at this instant essentially removes this error. Reference drift during pre-de-orbit coast (gyro drifts and computer roundoff) cannot be improved by monitoring.

During de-orbit thrusting the disturbance torque, due to motor misalignment and c.g. offset, is essentially constant. The resultant RCS response gives very small (.0003°), high-frequency (15-40 cps) oscillations about one side of the deadband. Without monitoring, the worst-case angular uncertainty is equal to the deadband, 0.25° per axis in pitch and yaw. Gyro monitoring (possibly backed up by RCS firing signals) will determine the algebraic sign of the pitch/yaw offsets, thereby removing this error.

The engine cutoff level is biased somewhat below the desired ΔV, to account for blowoff execution lags and thrust tailoff (a few hundred milliseconds). Monitoring of the actual ΔV will remove the lag and tailoff uncertainties, leaving only the errors due to the accelerometer and the integration routine.

Effect on Entry Uncertainties - The uncertainties in the de-orbit conditions, with and without monitoring, may now be mapped into uncertainties in the entry conditions, using computed sensitivity coefficients for the nominal coast period. These results appear in Figure 4.2-2 and indicate that de-orbit monitoring provides improvements of about 30% in the entry parameters of interest. Note that these values are for propagation tc nominal entry time, not propagation to the defined entry altitude of 800,000 ft. The uncertainties in entry velocity and altitude are very highly correlated. This correlation must be taken into account in the priori statistics for atmospheric reconstruction.

4.3 AEROSHELL PROPERTIES - The Aeroshell aerodynamic characteristics and their uncertainties affect the ESP results by the local flow properties around sensors and the reconstruction of atmospheric profiles from the determination of the entry trajectory parameters. The preferred design utilizas a non-ablative heat protection system in the area of the atmospheric composition sensor and the camera windows. By the local elimination of the products of ablation, the quality of the science data is improved. The following section discuss the Aeroshell preferred design and its influence on ESP science measurements.

4.3.1 Aerodynamics - The aerodynamic characteristics of the Capsule Bus influences the ESP measurements in two respects:

a. Aeroshell aerodynamic uncertainties affect the atmospheric reconstruction technique.

b. Sensor data must be corrected for local flow properties.

4.3.1.1 Aeroshell Aerodynamic Characteristics - Aerodynamic characteristics of the 120° sphere cone with .5 nose to base radius and a flat, closed base are presented in Figure 4.3-1. These estimates were obtained by theoretical prediction and analysis of experimental data obtained at McDonnell and NASA centers on potential VOYAGER configurations.

Uncertainties in these coefficients arise in four areas:

a. Atmosphere Composition

b. Reynolds Number

c. Afterbody Configuration

d. Dynamic Characteristic

UNCERTAINTIES AT DE-ORBIT (3σ)

ERROR SOURCE	RESULTING UNCERTAINTY	
	NO MONITORING	BEST MONITORING
Pointing Errors, degrees		
Drift of Angular Reference	0.73	.532
Oscillations During Retrofire	.0003	.0003
Deadband	.25	--
RSS Pointing Error, degrees	.772	.532
Velocity Errors, ft/sec		
Lags and Tailoff	.4	--
Accelerometer/Integrator Error	.25	.25
RSS Velocity Error, ft/sec	.472	.25

Figure 4.2-1

RESULTANT UNCERTAINTIES AT ENTRY (3σ)
(DE-ORBIT ERRORS PROPAGATED TO NOMINAL ENTRY TIME)

PARAMETER	UNCERTAINTY	
	NO MONITORING	BEST MONITORING
Velocity, ft/sec	65.3	44.8
Flight Path Angle, deg	1.4	0.96
Altitude, kft	89.0	61.6
Central Angle, deg	0.61	0.42

Figure 4.2-2

RESULTS OF MCDONNELL POLYSONIC WIND TUNNEL TEST PROGRAM #195

ADDITIONAL AERODYNAMIC CHARACTERISTICS

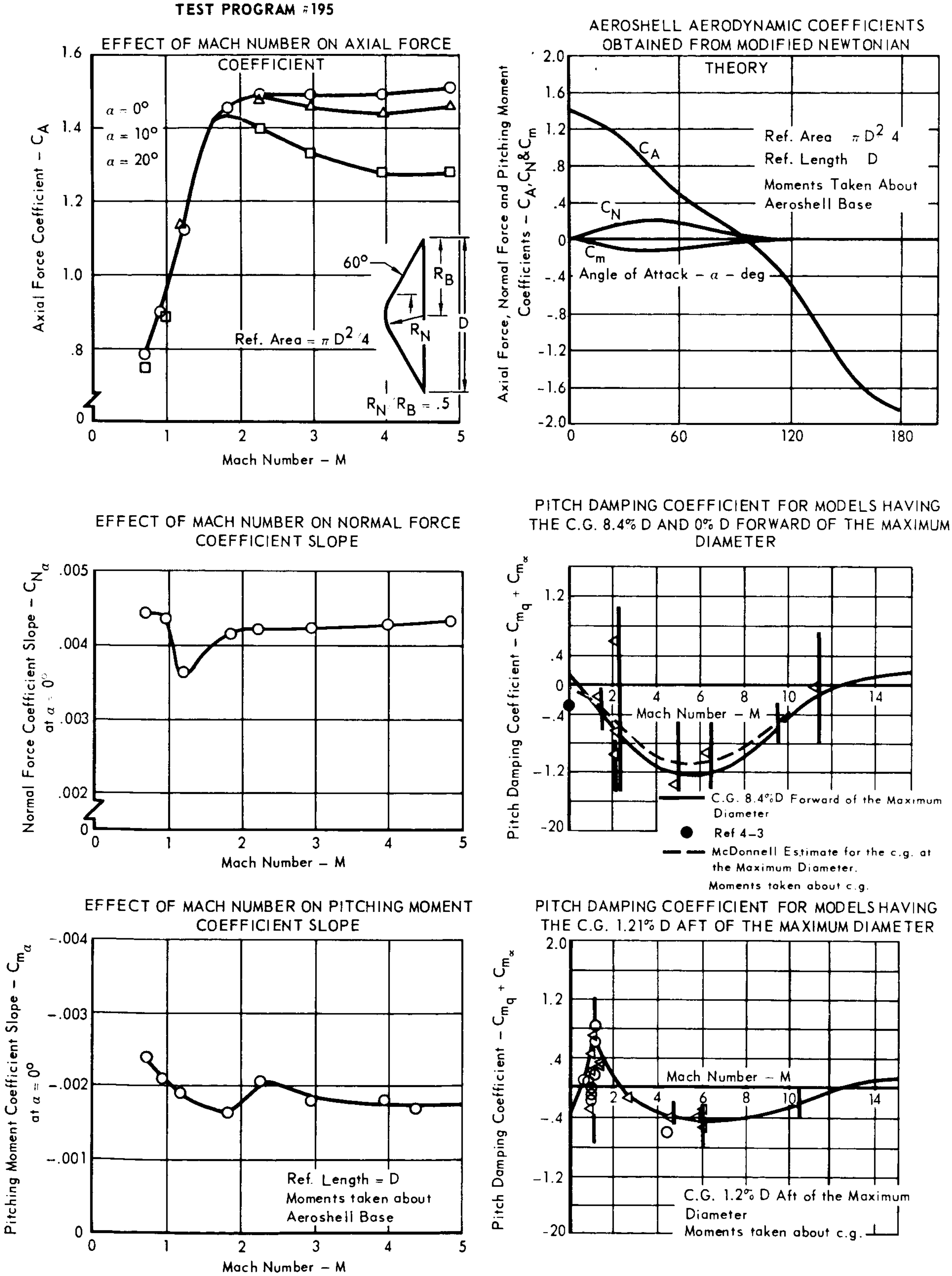


Figure 4.3-1

Atmosphere Composition - The uncertainties due to atmosphere gas composition are primarily caused by real gas effects occurring in the high temperature, low density shock layer at high Mach numbers. These effects manifest themselves by their influence on the magnitude and shape of the surface pressure distribution and on the corresponding aerodynamic characteristics. Tests on blunt shapes in air, and mixtures of air and carbon dioxide, Reference (4-1), indicate that these effects are minimal when flow separation is absent in the shock layer as it is for our configuration. The aerodynamic characteristics can be estimated from approximate theories such as modified Newtonian and tangent cone with more confidence when flow separation is absent. To estimate the effects of gas composition on aerodynamic characteristics, the terms in the approximate theories due to gas composition were examined. Newtonian theory states that the static aerodynamic characteristics for a given configuration are functions of the stagnation pressure coefficient, $C_{P_{STAG}}$. The difference in axial force coefficient, C_A, normal force coefficient, C_N, and pitching moment, C_m, from gas A to gas B is

$$\frac{(C_{P_{STAG}})_A}{(C_{P_{STAG}})_B} = \frac{(C_A)_A}{(C_A)_B} = \frac{(C_N)_A}{(C_N)_B} = \frac{(C_m)_A}{(C_m)_B} \qquad (1)$$

This ratio varies with Mach number, altitude and atmospheric gas composition. Using the VM atmospheric models, our studies show that $C_{P_{STAG}}$ may vary from 1.85 to 1.96 between Mach 8 and 30. This means that Aeroshell static aerodynamic characteristics, determined from tests on Earth in air, will have an uncertainty no greater than six percent in the Martian environment.

Reynolds Number - Reynolds number induces uncertainty in axial force coefficients and base pressure coefficients at subsonic Mach numbers due to transition of the boundary layer from laminar to turbulent flow. Preliminary estimates indicate that the Aeroshell boundary layer should be laminar at subsonic flight conditions. At this time experimental data available on Aeroshell base pressure coefficient is insufficient to account for Reynolds number effects. Consequently, Aeroshell base pressure estimates are based on correlations of data from wind tunnel tests with similar shaped bodies.

Afterbody - Our studies show that symmetrical afterbodies do not significantly affect the aerodynamic characteristics of the Aeroshell if they lie close to the base and are shielded in the base flow. This is demonstrated, within the limits of data scatter, by References (4-2) and (4-3).

Dynamic Effects - The largest uncertainties occur in the dynamic damping coefficient ($C_{m_9} + C_{m_2}$). Data from Reference 4-2, obtained in the Naval Ordance Laboratory Ballistic Range, indicate a trend to dynamic instability as the Mach number is decreased from supersonic to subsonic values. This conflicts with data obtained in the Langley Spin Tunnel, Reference 4-3, which indicates dynamic stability at low subsonic Mach numbers. The anomaly must be resolved during the Aeroshell development program. However, the range of expected results will not seriously affect the entry measurements since our design incorporates a rate damping system. This system will maintain a very small angle of attack oscillation envelope which should not influence the accuracy of entry science measurements (see Section 4.5).

4.3.1.2 Local Flow Analysis - Local flow analysis is required to describe the fluid state about the sensors and to provide the equations and techniques needed to interpret freestream conditions from the measured values. The fluid state depends on Mach number, Reynolds number, angle of attack, and gas composition.

Composition Sensor - The composition sensor at the stagnation point, as well as alternative systems described in Part E, Section 5.4.1, requires a knowledge of the shock standoff distance. Figure 4.3-2 presents shock standoff distance, as a function of Mach number, based on correlations and McDonnell test data on the preferred configuration from Mach 1.8 to 4.87. The extrapolation to higher Mach numbers is based on expected shock layer densities and correlations of standoff distance for spheres.

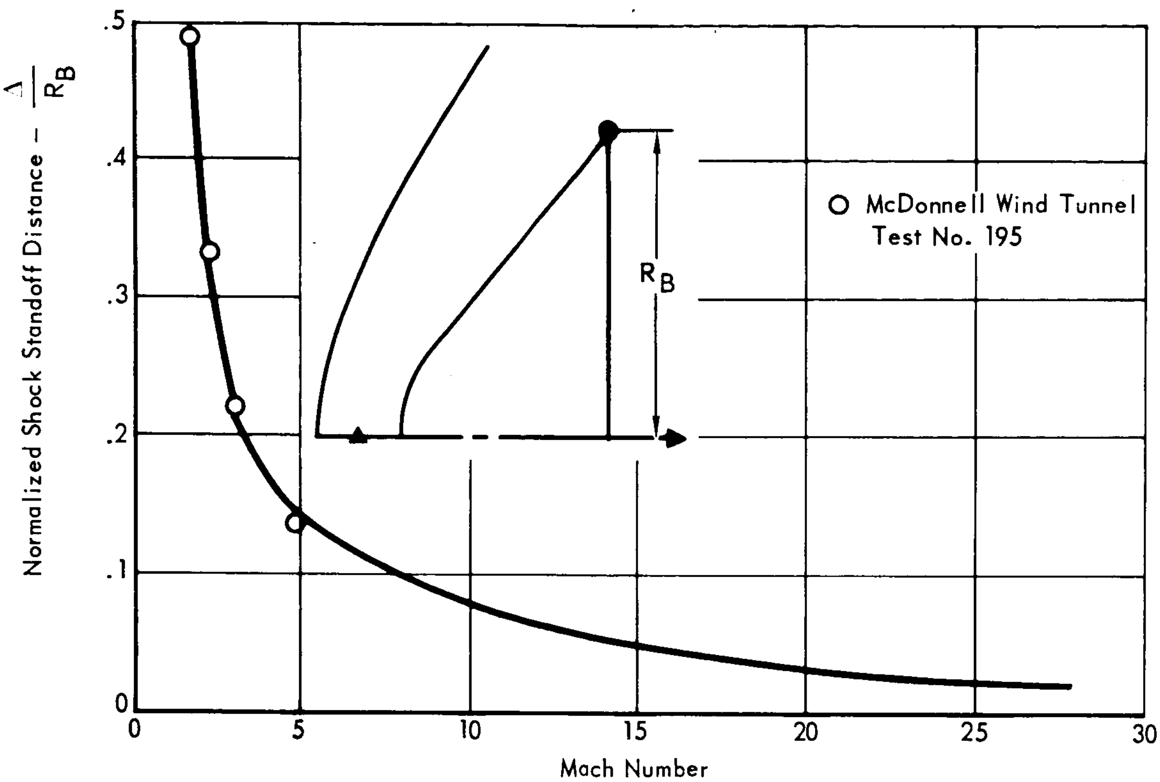
Below Mach 5 the stagnation point temperature is less than about 1300°K so there will be no significant ionization or dissociation in the shock layer to affect composition measurements.

Temperature Sensor - The stagnation temperature measurements can be related to freestream temperature by the equation

$$\frac{T_{STAG}}{T} = 1 + \frac{\gamma - 1}{2} rM^2 \qquad (2)$$

where T_{STAG} is the stagnation temperature, T is the freestream temperature, γ is the freestream specific heat ratio, M is Mach number, and r is the sensor recovery factor. Gas composition will affect the data interpretation by the value of γ chosen.

SHOCK STANDOFF DISTANCE FOR A 120° SPHERE-CONE



COMPARISON OF THEORETICAL AND EXPERIMENTAL SHOCK WAVE SHAPE

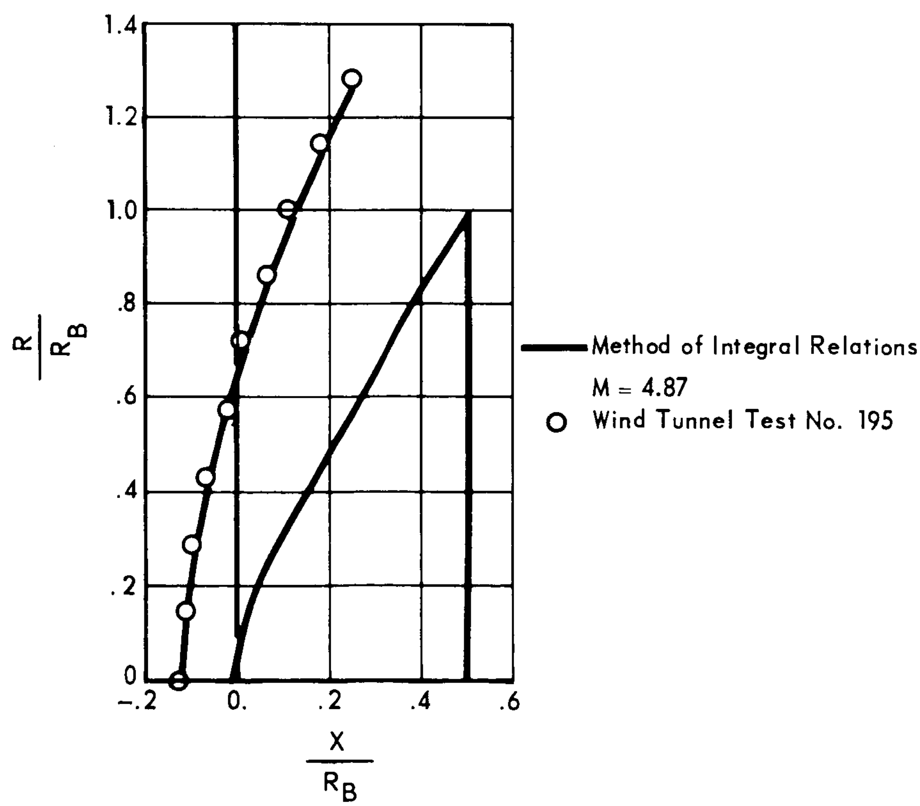


Figure 4.3-2

Pressure Sensors - The perfect gas equation for the stagnation pressure

$$\frac{P_{STAG}}{P} = \left[\frac{\frac{\gamma + 1}{2} M^2}{\frac{2\gamma}{\gamma + 1} M^2 - \frac{\gamma - 1}{\gamma + 1}}\right]^{\frac{1}{\gamma - 1}} \tag{3}$$

is valid from sensible atmospheric entry to Mach 1.0. It accounts for a streamline which passes through the normal shock and is isentropically decelerated to zero velocity at the stagnation point. Below Mach 1.0 the isentropic relation

$$\frac{P_{STAG}}{P} = \left[1 + \frac{\gamma - 1}{2} M^2\right]^{\frac{\gamma}{\gamma - 1}} \tag{4}$$

relates stagnation point and freestream pressures.

Figure 4.3-3 compares the stagnation pressures calculated for two trajectories using Equations 3 and 4 with flow field calculations accounting for the equilibrium chemistry of the gas. The values of the specific heat ration, γ, used in Equations 3 and 4 were 1.37 and 1.38 for the VM-8 and VM-9 atmospheres respectively. The flow field calculations were computed using a computer program which calculates normal shock and stagnation point properties for an arbitrary mixture of nitrogen, carbon dioxide and argon in chemical and thermal equilibrium. Freestream conditions as a function of time were obtained from the trajectories described in D-4.1. The agreement between the two calculations is good and justifies use of the perfect gas Equations 3 and 4 in future analyses.

Angle of Attack - If the Aeroshell does not maintain a zero angle of attack, the measured nominal stagnation point pressure and temperature must be corrected to the true value. Experimental pressure distributions are helpful for this purpose. A wind tunnel test was conducted at McDonnell to obtain these data in the Mach number range of .7 to 4.87. Sample results for the windward and leeward planes at a Mach number of 4.87 are presented in Figure 4.3-4 for angles of attack of 0, 10 and 20 degrees. At zero angle of attack a comparison is made with modified Newtonian theory and calculations obtained with a one strip method of integral relations technique described in Reference 4-4. At angle of attack the data is compared with predictions from modified Newtonian theory. The figure shows that the method of integral relations provides a better definition of the surface pressure distribution at zero degree angle of attack than that provided by the

CALCULATED STAGNATION PRESSURE

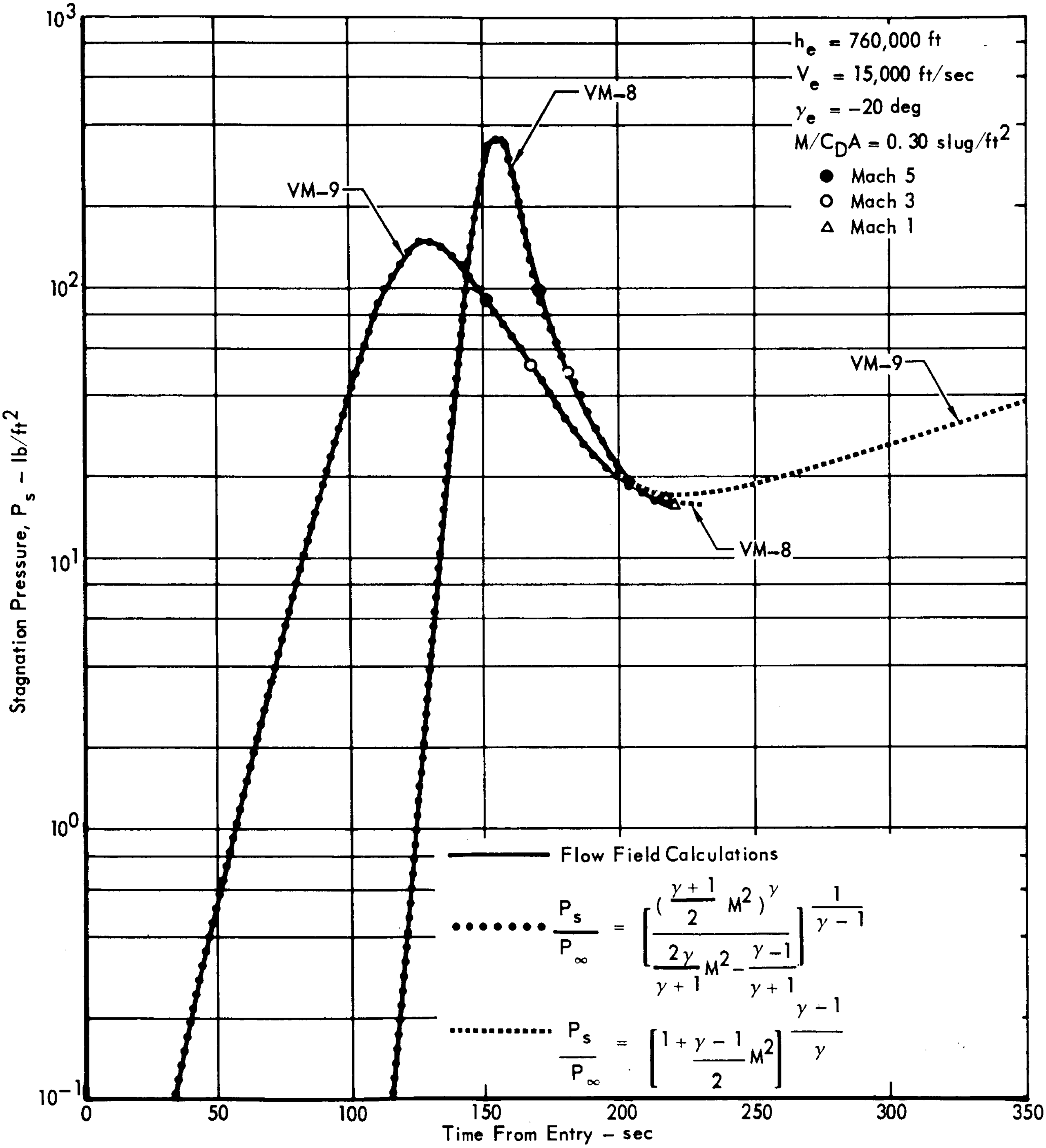
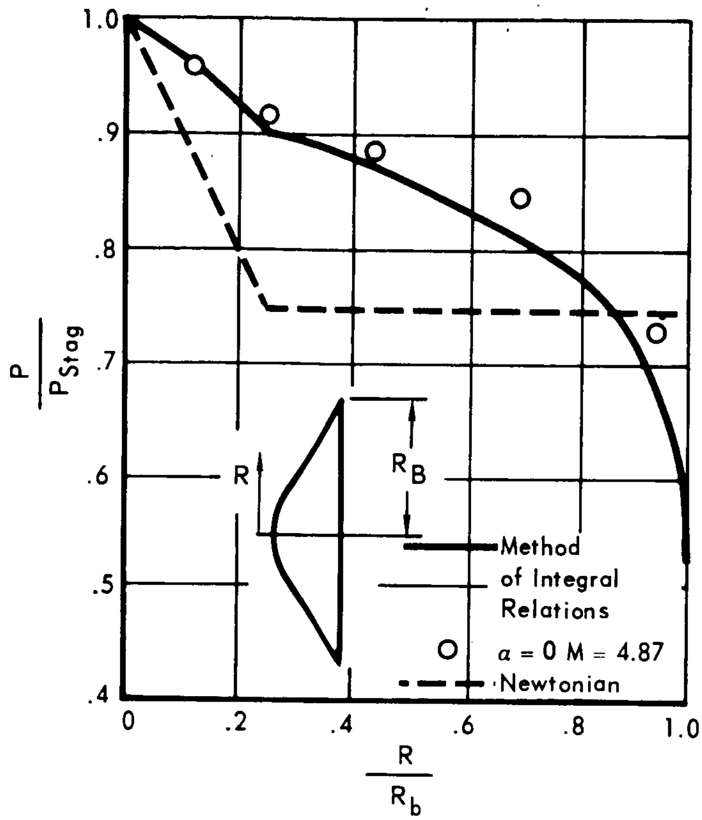


Figure 4.3–3

COMPARISON OF PRESSURE DATA FROM MCDONNELL WIND TUNNEL TEST #195 AND THEORY AT A MACH NUMBER OF 4.87

PRESSURE COEFFICIENT AT ZERO DEGREE ANGLE OF ATTACK



EFFECT OF ANGLE OF ATTACK ON THE PRESSURE COEFFICIENT

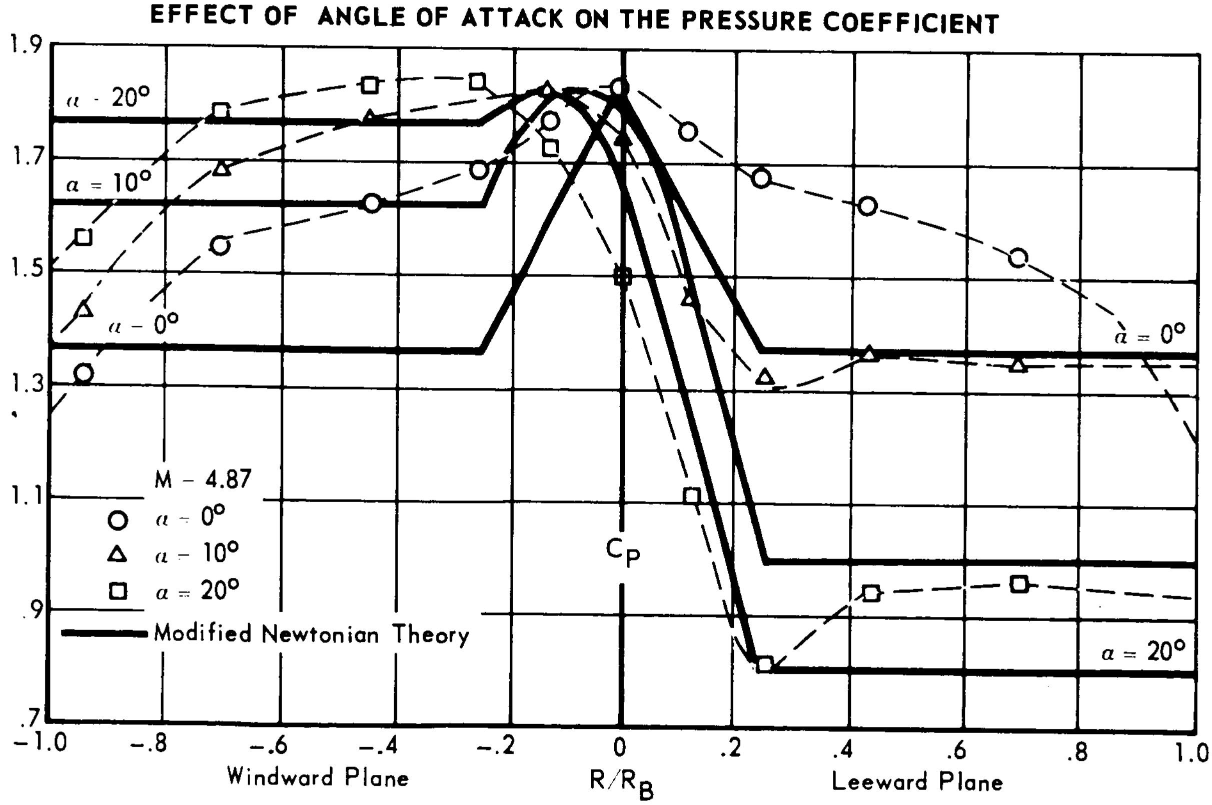


Figure 4.3–4

Newtonian theory. The falling pressure as the corner is approached for the method of integral relations is due to the requirement that the Mach number at the corner be equal to one. This requirement is valid as long as the asymptotic pointed cone Mach number is less than one. Agreement between Newtonian theory and experimental data at angle of attack becomes better as the angle of attack is increased. These two theoretical techniques are used together to estimate the pressure at other points on the Aeroshell and to locate the stagnation point as a function of angle of attack for Mach numbers greater than about 1.5. At subsonic and transonic speeds experimental data is being used to correlate the surface pressure distribution.

4.3.1.3 Development - To provide measurements in the final flight phase requires a definition of the Lander aerodynamic characteristics and the flow field about the Lander, with and without the landing thrusters operative. This definition must be delayed until the lander design is finalized. Base pressure and temperature correlations for the Aeroshell should also be defined in future programs.

4.3.2 Heat Protection System - The basic Aeroshell heat protection system consists of non-ablative, low density, hardened Fiberfrax ceramic material for the spherical nose section, low density General Electric ESM 1004X silicone elastomeric ablator on the conical section, and a fiberglass cloth aft curtain.

Two ESP subsystems interface with the basic heat protection system and necessitate a non-ablating nose cap. The major requirement for these subsystems is that there be no out-gassing from the heat protection system which will contaminate the sampled boundary layer gas and degrade the descent TV window optics. The atmospheric measurement probe is located at the apex of the nose cap, and the TV window is located at the sphere-cone tangency line.

4.3.2.1 Non-Ablative Nose Cap - Selection of a non-ablative nose cap material was made because the Gemini flight experience and test results by McDonnell have indicated that the ablation products from an ablative nose cap would interfere with the ESP atmospheric measurements and degrade the TV camera window optics. The ablation products consist mainly of low molecular hydrocarbons, metallic oxides, and some species in elemental and ionized form. The range of the composition is quite wide and the composition is sensitive to the heating environment and flow field characteristics. This makes the isolation and identification of the ablation products from Martian gas very difficult. It is also known that the ablation products, oxides in particular, deposit on the glass type optical window and obstruct the optical viewing. Therefore, covering the entire nose cap with a non-ablating material will preclude the window deposition of the ablation products as well as the contamination of sampled gas.

Among the several non-ablative materials evaluated, low density (25 lb/ft^3), hardened Fiberfrax showed the best integral thermal structural properties. These include low density and thermal conductivity, good thermal shock resistance, no degradation upon exposure to cold soak (-150°F), high temperature (2500°F), and hard vacuum. It also has good RF transparency, and essentially no out-gassing is expected because this material is completely inorganic.

The ceramic nose cap is bonded to the backup structure, consisting of phenolic fiberglass honeycomb and faceplates, with high temperature film adhesive HT435.

The ceramic nose shield thickness is 0.32 inch and will limit the bondline to 640°F. The maximum surface temperature expected during entry heating is 2100°F, which is well below the material limit temperature.

4.3.2.2 Atmospheric Measurement Probe - The atmospheric measurement probe provides the inlet for the boundary layer gas for the measurements of total temperature, stagnation pressure, and gas composition. The probe is flush mounted on a 9.5 inch diameter and 0.5 inch thick beryllium plug. The beryllium plug provides a structural support and acts as a heat sink to limit the probe temperature to below 400°F. No thermal-structural problem of the plug is expected at this temperature level.

4.3.2.3 Conical Section Heat Shield - For the conical section of the Aeroshell, General Electric ESM 1004X ablator is used. This material is a foamed, low density (16.6 lb/ft^3) silicone elastomeric material, and was found to be the most efficient heat shield material for the conical section of the Aeroshell. The ablator is bonded to the titanium substructure with RTV 560 adhesive. The range of ablator thickness is 0.43 to 0.33 inch, and it is designed to limit the bondline temperature rise to 640°F.

4.3.2.4 TV Camera Window - The TV camera window provides a thermal cover and optical view path for the TV optics. For the window material, a single layer, 0.375 inch thick, of Corning fused silica 7940 is used because it has high service temperature for undistorted viewing and structural integrity. The maximum surface temperature of the window during entry heating is 1050°F, and no thermal-structural problem is anticipated at this temperature level.

To minimize the heating from the window backface to the TV optics, a heat control filter consisting of zinc sulfide and silicon oxide is coated on the window backface. This filter transmits visible light (95%+) and is heat reflecting (95%+). Thus it reflects a large portion of the infrared component of the radiation incident on the window backface back to the source. The maximum service temperature of the filter is 2000°F; and the temperature rise of the TV optics from this heating is expected to be about 50°F, which is well within its tolerance limit.

4.3.3 Structure and Vibration - An estimate of the Entry Science Package vibration level was made for use in determining science instrument accuracy. During entry, fluctuating pressure acting on the Aeroshell causes vibration in the Entry Science Package. The magnitude of the vibration depends on the type of aerodynamic flow, the elastic properties of the Aeroshell, the amount of damping provided by the ablator, and the type of structural attachment of the Capsule Lander to the Aeroshell.

If turbulent flow exists over the Aeroshell, vibration in the Entry Science Package will result. The vibration magnitude as a function of time will be proportional to dynamic pressure. An analysis that considered the entire range of VM atmospheres showed that turbulent flow over the Aeroshell exists only during steep, high-speed entry into the "even numbered" VM-series atmospheres and that the turbulence exists for about 50 seconds. The maximum loading occurs during such an entry into a VM-8 atmosphere, when the dynamic pressure is 210 pounds per square foot. From Gemini, Mercury, and earth re-entry vehicle flight data, an estimate can be made of the overall acoustic energy in the turbulent boundary layer. The overall fluctuating pressure during entry was obtained using the equation

$$\Delta p = 0.006\ q,$$

where q is the free stream dynamic pressure. Using this relationship, the maximum overall sound pressure level was found to be 130 db. This external noise level usually presents no problems to structure, for example, even under prolonged exposure.

Tests were conducted to determine the typical Aeroshell panel response to an acoustic environment. The tests were made on panels both with and without ablator. (See Volume II B 5.3.1.5 for a description of the tests and test results.) Using the resulting Aeroshell (with ablator) acceleration-pressure transfer function and the mass loading law, the overall vibration level for the Entry Science Package was calculated to be 0.95 g_{RMS}.

The fluctuating pressure acting on the Aeroshell is essentially white noise and has little of its energy in the low frequency range. Preliminary analysis indicates that the lowest natural frequency of the Capsule Bus is above 20 cps. Thus,putting the output of the Entry Science accelerometer through a low-pass filter yields essentially the rigid body accelerations of the Capsule Bus. Having these rigid body accelerations reduces the uncertainty in the estimated Capsule Bus trajectory found by the computational technique described in Section D 2.2 of this volume.

4.4 AEROSHELL SEPARATION - The method used to separate the Aeroshell from the Lander has a pronounced effect on the entry science data, particularly on the quantity of subsonic data acquired. These effects are discussed below.

4.4.1 Technique and Conditions - The preferred concept for Aeroshell Separation was selected as a result of a major interdisciplinary trade study which is detailed in Section 4.6, Part B, Volume II of this report. The selected concept consists of a differential drag technique using a parachute which acts as an auxiliary aerodynamic decelerator. Upon release of the Aeroshell, the Lander, because of its greater drag, decelerates more rapidly causing the Aeroshell to fall rapidly away. Deployment of the parachute takes place at 23,000 ft. Twelve seconds later the Aeroshell is released. The Lander/parachute combination descends until an altitude of 5000 ft is reached at which time terminal propulsion ignition is initiated. As soon as successful ignition is detected the parachute is released from the Lander and the Lander descends to the Martian surface along a pre-programmed control path.

4.4.2 Variations in Amount of Subsonic Data Due to Trajectory Spread - Figure 4.1-1 depicts the envelope of possible entry conditions. An examination of time history curves for Mach number and altitude vs time from entry indicates that the Capsule Bus is subsonic for the longest period, namely 120 seconds, for a VM-9 atmosphere with entry at 13,000 ft/sec at an entry angle of -10.4°. The other extreme is encountered in a VM-8 atmosphere with entry at 15,000 ft/sec at an entry angle of -20.0°. In this case the Capsule Bus is still supersonic at parachute deployment and becomes subsonic approximately 1-1/2 seconds prior to Aeroshell release. This gives a total possible range of subsonic operation from 1-1/2 to 128 seconds.

This will yield from 1 to 128 subsonic temperature/pressure readings, from zero to 13 mass spectrometer readings and from three to 256 accelerometer readings. In addition, from zero to 24 entry TV pictures will be generated while subsonic prior to separation.

The condition of minimum subsonic data acquisition will take place only when entry is into a VM-8 atmosphere, at 15,000 ft/sec at an entry angle of -20.0° with parachute deployment at Mach 2.0. Any set of entry or separation conditions representing a less severe aerodynamic regime will yield more data.

4.4.3 Interfaces Affected by Separation - Sensors for the measurement of stagnation temperature and stagnation pressure are located in the nose cap of the Aeroshell. These measurements will therefore cease at Aeroshell separation. All other entry science instruments continue to function after separation, although the flow field relationships applicable to the base region pressure and temperature sensors are of course significantly altered.

4.5 ATTITUDE CONTROL - This section describes the preferred attitude-control mechanizations for the various phases of the VOYAGER mission, and the impact of attitude control performance upon the quality of the ESP experiment data. The discussion is broken into several mission phases, covering the time span from CB/FS separation until landing. Alternative attitude control mechanizations are discussed in Part E, Section 5.6.

4.5.1 Deorbit Phase - This phase begins with separation of the Capsule Bus from the Flight Spacecraft, and ends after shutdown of the deorbit motor. Attitude control performance during this phase affects primarily the trajectory reconstruction task of the Atmospheric Properties Determination Experiment, in that pointing angle uncertainties during deorbit thrusting generate uncertainties in entry conditions. See Section 4.2 for further discussion of this point, and for detail derivation of angular error values.

Separation of the Capsule Bus is accomplished by salvo firing of the four aft-facing (pitch and yaw) RCS thrusters. These thrusters fire for about two seconds to impart a separation Δ V of about 1.4 ft/sec. Salvo firing may be overridden by off-logic as necessary for attitude control; i.e., all thrusters fire with zero attitude errors, individual thrusters are shut off for control.

Attitude reference signals originate from three orthogonal, strapdown, pulse-rebalanced gyros, initially aligned to the Flight Spacecraft attitude reference. These signals are processed by the Guidance and Control Computer (GCC) to generate inertially referenced Euler angles. Euler angle computations are used instead of simple rate integration, to reduce attitude reference drift due to "coning".

During the coasting interval of about 20 minutes until de-orbit, the Capsule Bus is placed in the proper attitude for de-orbit, and this attitude is maintained to within a 2° deadband.

The attitude control deadband is reduced to 0.25° during de-orbit thrusting. Pointing angle uncertainty during de-orbit thrusting is 0.53° for the preferred design.

4.5.2 Pre-Entry Phase - After de-orbit-motor cutoff, the CB is oriented to the desired pitch and yaw entry attitude, and placed in a slow roll mode (3-4 rev/hr) for thermal control. Inertial pitch and yaw attitude is maintained within a 2° deadband. To conserve electric power, the GCC is turned off, and remains off until shortly before entry. Euler angles computations are no longer computed; therefore, coning effects degrade the attitude reference during the six-hour pre-entry coast. For a 2° deadband, the attitude uncertainty at entry (800 K ft altitude) will be 15° (3 σ) in pitch and yaw. Part E, Section 5.6 discusses the effect of this attitude uncertainty upon entry attitude dynamics, fuel expenditure and TV imaging during the inertial-hold portion of the entry phase, and considers various compensation strategies.

Seven minutes before the nominal entry time,the GCC is turned on again, the slow roll command is removed and the ESP telemetry begins transmitting gyro and accelerometer data. As sufficient atmosphere is encountered to raise the aerodynamic accelerations above the instrument noise level, it becomes possible, in post-flight analysis, to refine the inertial pitch and yaw uncertainties by comparison of the axial acceleration with the two normal accelerations. By comparison of the relative magnitudes of the two normal accelerations, it is also possible to form a rough estimate of the inertial roll attitude. Analysis of TV images will also aid the post-flight reconstruction of Capsule attitude during the inertial-hold portion of the entry phase (assuming that Mars comes within the TV field of view).

4.5.3 Entry Phase - When the (axial) guidance and control accelerometer senses an acceleration of 0.05 g_e, the attitude control mode switches from inertial hold to three-axis rate damping, using a 3 deg/sec deadband. The altitude at which this occurs may be from 200-500 Kft, depending upon the entry conditions and atmosphere.

Attitude dynamics of the entry phase are described in detail in Volume 2, Part B, Section 2.3.4. The following brief discussion should suffice in the present context: The capsule is statically stable, so that any angular offset of the aeroshell axis from the relative wind generates a restoring aerodynamic torque. The initial offsets in pitch and yaw therefore store a certain amount of potential energy in the attitude dynamics. This energy manifests itself in angular oscillations, continuing through the entire atmospheric phase, whose magnitude and frequency change with the changes in dynamic pressure over the flight regime. Wind-induced disturbances inject additional energy into the oscillation dynamics. It is the task of the attitude control system to sense these oscillatory rates and fire opposing RCS jets, thus removing energy and bringing the angle and rate envelopes within acceptable levels.

Attitude control performance affects the ESP data in the following ways:

- o Angular rates over 4 deg/sec produce image "smearing," degrading the resolution of the TV pictures.
- o The centripetal accelerations due to angular rates appear as error on the accelerometer outputs, particularly that of the axial accelerometer, which is located about 3 inches forward of the capsule c.g. This acceleration error can be partially corrected, but a residue affects the trajectory reconstruction accuracy.
- o Angle-of-attack excursions also affect the trajectory reconstruction task, by changing the effective axial force. For example, a 1.4° peak oscillation (1.0° rms) appears as an 0.3% uncertainty in the drag coefficient.

To mitigate the effects of the latter two error sources, the post-flight data-reduction process is designed to incorporate reconstruction of capsule attitude dynamics or kinematics, as well as trajectory dynamics. For this purpose, the pitch and yaw gyro rates and the triaxial accelerometer are sampled two or more times per second. With only two samples/second, the accuracy of the attitude reconstruction falls off rapidly as oscillation frequencies rise above one cycle per second. Near maximum dynamic pressure, frequencies of 1.7 to 2.5 cps may be expected; fortunately, the period when the frequency is highest is also the period when the angle-of-attack envelope is smallest.

Steady Winds - Figure 4.5-1 lists attitude envelopes and impulse expenditure in the presence of mean and shear wind profiles of the form specified in reference 4-5. The same entry conditions (V_e = 15 Kft/sec, γ_e = -20, γ_e = 5° at 800 Kft) were used for all runs. Runs were made in VM-8 and VM-10 model atmospheres, using the winds appropriate to each. The three aerodynamic damping curves shown in Figure 4.5-2 were used: "N" denotes $C_{m_q} + C_m = 0$ for all mach numbers, "B" denotes

SAMPLE ENVELOPE AND IMPULSE EXPENDITURE DATA
(ENTRY CONDITIONS: V = 15k ft/sec, $\gamma = -20°$, $\alpha = 5°$ AT 800k ft)

RUN	ATMO VM–	WIND		C_{mq} CURVE	OSC ENVELOPE			IMPULSE lb-sec
		TYPE	AZIMUTH		100k ft	50k ft	25k ft	
12f	8	None	–	N	.12	.06	.14	147.3
20		Mean	Tail	N	.12	.06	.15	152.6
21		Mean	Head	N	.12	.07	.17	157.6
22		Mean	Cross	N	.12	.08	.19	193.2
24a		Shear	Tail	N	.21	.09	.21	167.7
23		Shear	Head	N	.12	.08	.19	218.3
103a		Shear	45°	A	.50	.20	.12	158.4
97a		Shear	Cross	B	.50	.48	.22	191.1
27		Shear	Cross	N	.12	.12	.25	289.1
12g	10	None	–	N	.08	.15	.28	192.1
99		Mean	Head	B	.21	.04	.03	138.0
93		Mean	Head	N	.19	1.0	1.4	158.0
104		Mean	45°	A	.26	1.4	1.6	281.0
92		Mean	Cross	N	.19	1.6	1.8	226.2
98		Shear	Head	B	.21	.95	.85	181.2
103		Shear	45°	A	.26	1.6	2.2	322.3
97		Shear	Cross	B	.21	1.7	.63	216.8
106		Shear	Cross	N	.20	1.8	1.3	323.0
102		Shear	Cross	A	.26	1.5	1.9	337.6

C_{mq} Curves: B, c.g. at base of aeroshell
A, c.g. 1.6-ft aft of base of aeroshell
N, neutral

Figure 4.5–1

VOYAGER PITCH DAMPING COEFFICIENT VS. MACH NUMBER

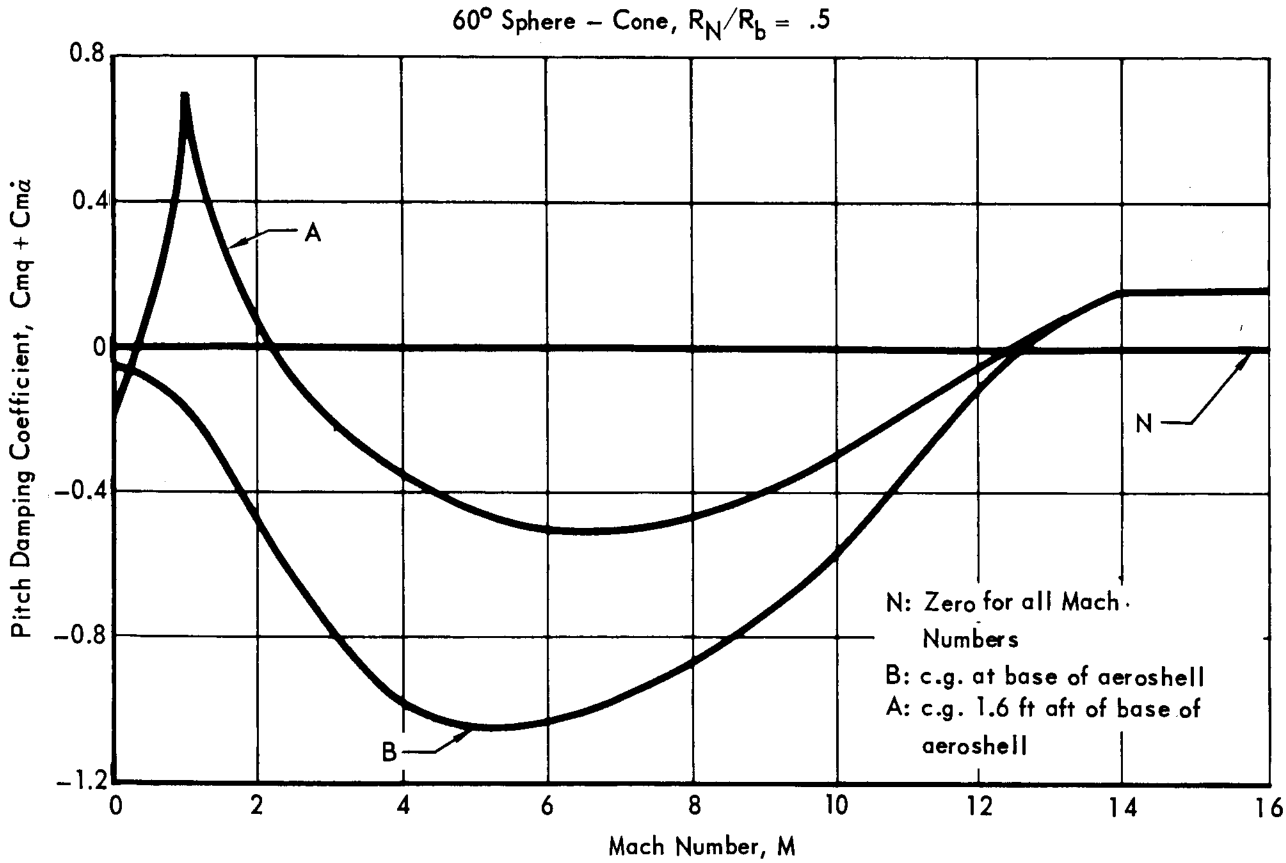


Figure 4.5–2

our current best estimate of the damping for the c.g. at the base of the aeroshell, and "A" denotes the estimate for the c.g. 1.6 ft aft of the base. The "B" curve is most relevant to the present entry configuration, for which the c.g. is 4.6 in. forward of the base of the aeroshell.

Figure 4.5-1 indicates that the system maintains adequate control of the attitude disturbances in the presence of design winds of a steady nature, even with aerodynamic damping more pessimistic than the current design. That is, the attitude envelopes are small enough that the effective change in drag coefficient, even without correction, is well below the uncertainty in the coefficient itself.

It should also be noted that in all runs using these wind profiles, the RCS held the rates well below 4 deg/sec. With the axial accelerometer 3 in.forward of the c.g., this means that the centripetal acceleration due to attitude rate is less than 0.001 ft/sec^2, and that TV images will be smear-free.

Transient Winds - Winds of a transient nature - step and gust winds - are much different from the steady wind case, as shown by Figure 4.5-3. Here we define a "step" as a wind which is zero above a certain altitude, and constant below it; we define a "gust" as a wind which is non-zero only for a fixed altitude range or time period. All gusts used in this study were "resonant" gusts , of duration exactly half the Capsule's natural oscillation period.

Zero dynamic damping coefficients were used for most of the gust-response simulations (corresponding to the use of the "N" curve of Figure 4.5-2). Positive (unstable) damping-coefficient values were used for some of the simulation runs, to illustrate the significance of uncertainty in the available damping data. No negative (stable) values were used; improved inherent stability would decrease the time-to-control and impulse-to-control, but would have little effect on the peak angles and rates. To obtain the relevant damping-coefficient values from the "B" and "A" curves, note that with the specified entry conditions, the mach number in the VM-8 atmosphere is 11.7, 3.4 and 1.9 at altitudes of 50, 30 and 20 Kft respectively; the mach number for the VM-10 atmosphere is 0.5 at both 30 and 20 Kft altitude.

The sample results of Figure 4.5-3 show that design wind transients at low altitudes overpower the attitude control system for short periods of time. The magnitude of the transient angular and rate oscillations generally increases with decreasing wind-onset altitude, due to lower capsule velocity and steeper flight path angle. The combined result of these two effects is that a horizontal wind of a given velocity produces a larger angular deflection of the velocity vector.

RESPONSE TO LOW-ALTITUDE WIND TRANSIENTS: SHARP-EDGED STEPS AND SHARP-EDGED RESONANT GUSTS

(ENTRY CONDITIONS: V= 15kft/sec, $\gamma = -20°$ AT 800 kft; 16.5-lb RCS)

RUN	ATMO-SPHERE VM–	WIND CONDITIONS				AERO DAMPING	SYSTEM RESPONSE			
		h_w kft	V_w ft/sec	TYPE	AZIMUTH		θ_{max} deg	$\dot{\theta}_{max}$ deg/sec	T_c sec	I_c lb-sec
20	8	50	320	S	Head	0	1.05	8.0	4.0	38.0
22		50	320	S	Cross	0	2.72	34.0	8.4	137.3
81		30	200	G	Head	0	2.64	18.3	3.5	52.5
82		30	200	G	Cross	0	9.47	66.4	13.4	215.7
47		20	200	S	Head	0	3.25	13.0	2.4	37.2
78		20	200	S	Cross	0	8.98	38.0	7.3	117.3
80a		20	200	G	Head	0	6.14	25.4	5.4	85.0
80b		20	200	G	Head	0.5	6.40	27.9	8.8	140.0
79a		20	200	G	Cross	0	16.84	73.8	14.5	234.3
79b		20	200	G	Cross	0.3	29.89	83.7	–	–
79c		20	200	G	Cross	0.5	90 +	–	–	–
83a	10	30	100	G	Head	0	23.29	54.2	11.5	197.9
83b		30	100	G	Head	0.3	24.21	58.7	36.0	586.8
84		30	100	G	Cross	0	24.69	56.8	12.3	211.2
43		20	100	S	Head	0	14.51	34.7	7.0	112.2
44		20	100	G	Head	0	26.84	61.8	13.3	219.3
65a		20	100	G	Head	0.1	27.15	63.4	17.4	289.9
107b		20	100	G	Head	0.2	27.61	65.3	27.8	452.3
65b		20	100	G	Head	0.3	90+	–	–	–

Notation – h_w = Wind Altitude
V_w = Wind Velocity
S = Sharp-Edged Step
G = Sharp-Edged Resonant Gust
θ = α or β, as Appropriate
$\dot{\theta}$ = q or r, as Appropriate
T_c = Time to Control to 3 deg/sec
I_c = Impulse to Control to 3 deg/sec
Aero Damping = C_{mq} or C_{nr}, as appropriate (positive values are unstable)

Figure 4.5–3

On the other hand, oscillation frequencies are lower at lower altitudes, so that attitude dynamics are more amenable to reconstruction with the specified sample rates.

Besides the effects on trajectory reconstruction, these large transient disturbances also cause loss of two or three TV pictures, and disturb the flow field to the point where stagnation pressure measurements and even base-region measurements may be affected.

4.5.4 Aerodynamic Deceleration Phase - At an altitude of 23,000 ft. the parachute is deployed, and the RCS is shut down. Six to eight seconds later, the aeroshell is separated from the Capsule Lander, taking the RCS with it.

Large attitude and rate oscillations may be expected during this phase, both from the deployment transient and from the parachute's extreme sensitivity to gust disturbances. The parachute's dynamic derivatives are not well defined; the form of the Capsule's oscillation dynamics (compund pendulum) is quite complex. For all these reasons, acceleration data will be rather noisy, and will be weighted somewhat lower than the altimeter data in the trajectory reconstruction data-processing scheme.

To define an attitude reference for terminal descent thrust initiation, the GCC begins processing of gyro rates via the "soft cage" filtering scheme described in Volume 2, Part B, Section 2.3.7. This scheme is designed to "filter out" the attitude oscillations, and determine the direction of the Capsule Lander's velocity vector. Pending further analysis of the filtering process, the attitude uncertainty at the beginning of terminal descent thrusting is presently estimated at 10-20 degrees.

4.5.5 Terminal Descent Phase - At 5000 ft. of altitude, the four throttleable (10:1) bipropellant terminal descent engines are ignited, and the parachute is separated. The four engines are throttled together for velocity control, and differentially throttled for attitude control.

Differential throttling of the large terminal engines provides very powerful attitude torques, and aerodynamic torques are rather small, due to the removal of the aeroshell. Therefore, attitude transients due to initial misalignments and wind gusts are removed very rapidly, typically 3 to 5 sec.

Prior to radar lock-on, attitude is referenced to the velocity vector, derived by the filtering process mentioned above. As a radar range/range-rate data becomes available, velocity and attitude are controlled to secure the desired cutoff conditions: Vertical attitude, zero lateral velocity, and less than 10 ft/sec vertical velocity at an altitude of 10 feet. When these conditions are achieved, the terminal descent engines are shut down, and the Capsule Lander drops to the surface of Mars.

4.5.6 Summary - Performance of the preferred attitude control system has been shown to be adequate to ESP requirements, over all mission phases, in the presence of all design conditions. Section 5.7 of Part E deals with alternative system designs and their performance, not only in terms of improving performance to meet more stringent conditions, but also in terms of reducing performance, or entirely eliminating attitude control for certain phases, to simplify the design or reduce weight.

4.6 RADAR ALTIMETER - The Radar Altimeter provides altitude data from 200,000 feet to 50 feet for the correlation of entry science data. It is also used for terminal descent event sequencing. This section discusses: 1) Development of the sequencing and entry science data requirements; 2) Physical and functional description;3) System performance and accuracy including the results of inaccuracy; 4) Elevation changes, resolution and surface profile determination;5) Development status and development requirements. A more detailed functional description is provided in Volume II, Part C, Section 10.1.

4.6.1 Development Of Requirements - Landing Radar turn-on, aerodynamic decelerator deployment and terminal propulsion engine initiation by means of radar altimeter triggering was selected over other methods, to provide reliable and accurate sequencing. Other methods rely on the ability to measure velocity, peak deceleration, static pressure or a combination of these, and relate a measurement to a specific altitude. Due to the wide variation in the postulated atmospheres, the desired measurement could occur during an excessively wide altitude interval. Landing Radar turn-on at 100,000 feet set the highest altitude of operation for sequencing marks. Operation at altitudes higher than 100,000 feet appeared desirable to enable a more accurate determination of atmospheric properties and TV subsystem imaging correlation.

Selection of the highest altitude at which altimeter data should be available was aided by a preliminary computer study. The computer runs simulated altimeter data obtained from 100, 200, 300 and 500 thousand feet and below, with a "plasma blackout" period from 400 to 150 thousand feet, for the 500 thousand foot runs. Altimeter accuracies of both 1% (or $\pm$ 200 feet), and 3% (or $\pm$ 500 feet) were used for the simulated altimeter runs. Data from the computer study shows the resultant inaccuracy in estimating altitude and atmospheric density back to 800 thousand feet. Figure 4.6-1 shows how the ability to estimate altitudes above the point where the

ALTITUDE ESTIMATION ACCURACY ($3\sigma_{HE}$) AS A FUNCTION OF ALTIMETER ACCURACY ($3\sigma_H$), AND DISTANCE OF FIRST ALTITUDE MEASUREMENT

(VM-3 ATMOSPHERE)

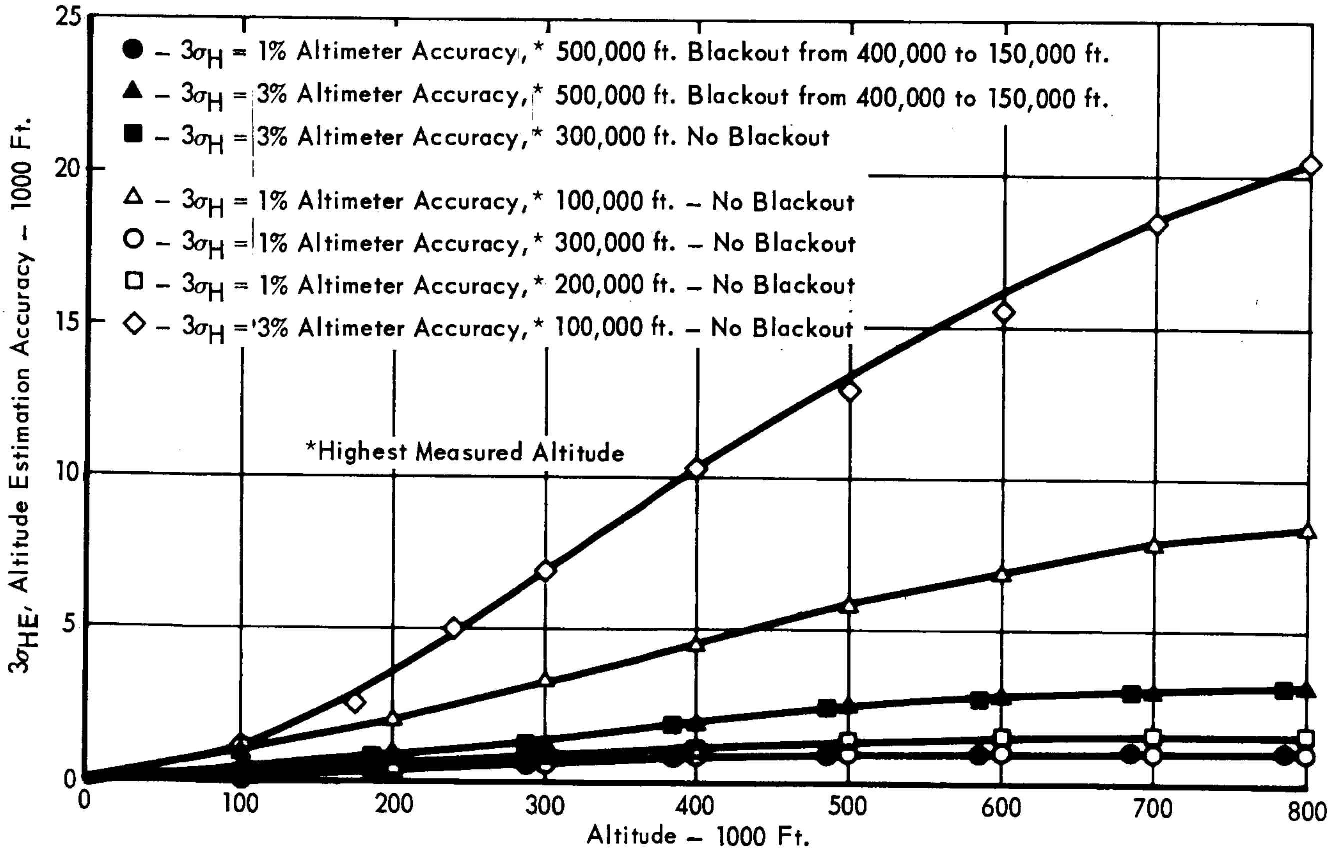


Figure 4.6-1

first measurement was made is equally dependent on highest altitude of operation and greatest altimeter accuracy. Figure 4.6-2 shows that the density estimation error is more dependent on altimeter accuracy. A maximum altitude of 200,000 feet and an altimeter accuracy of 1% (3 σ) was selected as the combination to best fulfill the entry science data requirements.

4.6.2 Physical and Functional Description - The Radar Altimeter consists of: (1) Electronics package with integral dipole antenna, (2) Conical monopole primary antenna, and (3) R.F. transmission line from electronics package to primary antenna.

Figure 4.6-3 shows the placement of the equipment on the Lander and the Aeroshell. The electronics package configuration and placement of the major elements are shown in Figure 4.6-4. Features of the primary antenna, with the Entry Science temperature and pressure equipment located inside the radiating element, are shown in Figure 4.6-5.

Major elements which make up the electronics package, and their relationships, are shown in Figure 4.6-6. The altimeter is a pulsed radar system using broad beam antenna coverage and servo controlled range tracking. System parameters selected for the altimeter are shown in Figure 4.6-7.

At time T_O, five-microsecond pulses are generated in the transmitter section (see Figure 4.6-6) and are sent through the R.F. section to the antenna. The R.F. energy travels to the surface in a spherical "packet" and illuminates the surface in an expanding circular area. The reflected signal energy returns to the altimeter at a time from T_O equal to twice the altitude divided by the speed of light. The reflected signal energy reaches a maximum at a time equal to the pulse-width (5 microseconds) later. This ramp function R.F. signal is amplified and mixed with an intermediate frequency in the RF amplifier and receiver section. The RF amplifier amplifies the resultant signal and the envelope detects and shapes the video pulse. An early-late gate is positioned around the video signal by a servo-controlled tracker. Altitude is measured by gating an accurate crystal oscillator clock into an accumulator, starting the gate with the transmitter T_O signal and ending the gate with the early-late gate signal. This provides a range count proportional to altitude which can be converted to a digital signal and transmitted to Earth by the Telecommunication subsystem.

4.6.3 System Performance and Accuracy - An analysis of the altimeter performance and accuracy based on detection and acquisition capability, basic accuracy and accuracy degradations due to rates and degraded signal returns was performed in

ATMOSPHERIC DENSITY ESTIMATION ACCURACY ($3\sigma_\rho/\rho$) AS A FUNCTION OF ALTIMETER ACCURACY ($3\sigma_H$) AND DISTANCE OF FIRST ALTITUDE MEASUREMENT

(VM-3 ATMOSPHERE)

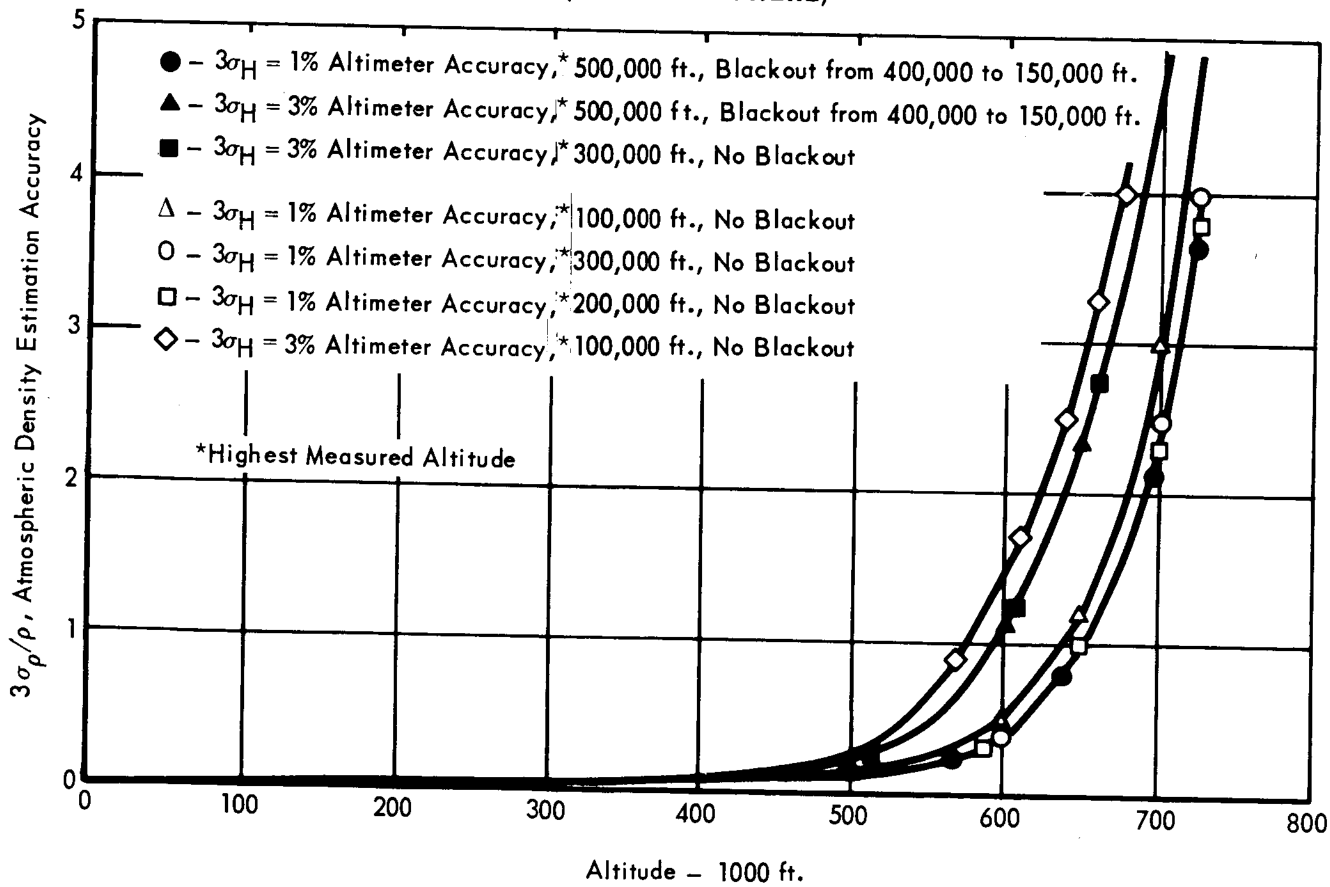


Figure 4.6-2

RADAR ALTIMETER EQUIPMENT LOCATION

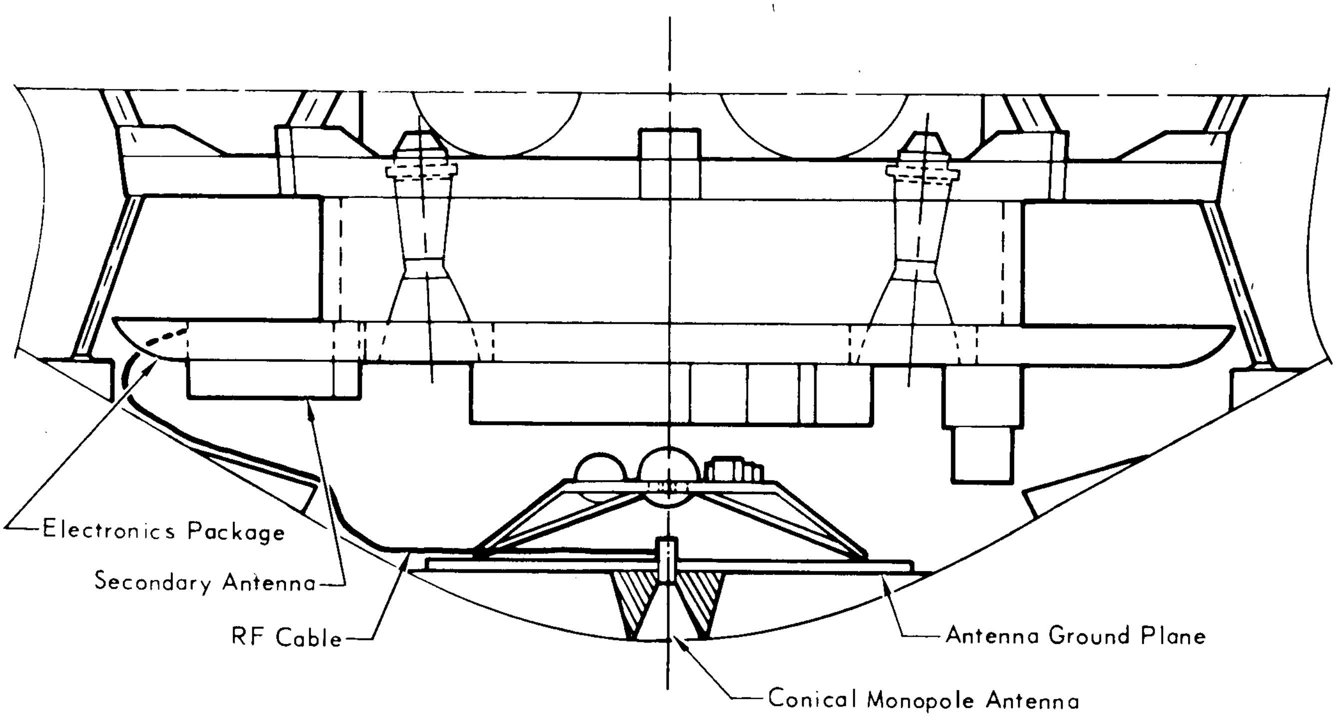


Figure 4.6-3

RADAR ALTIMETER ELECTRONICS PACKAGE

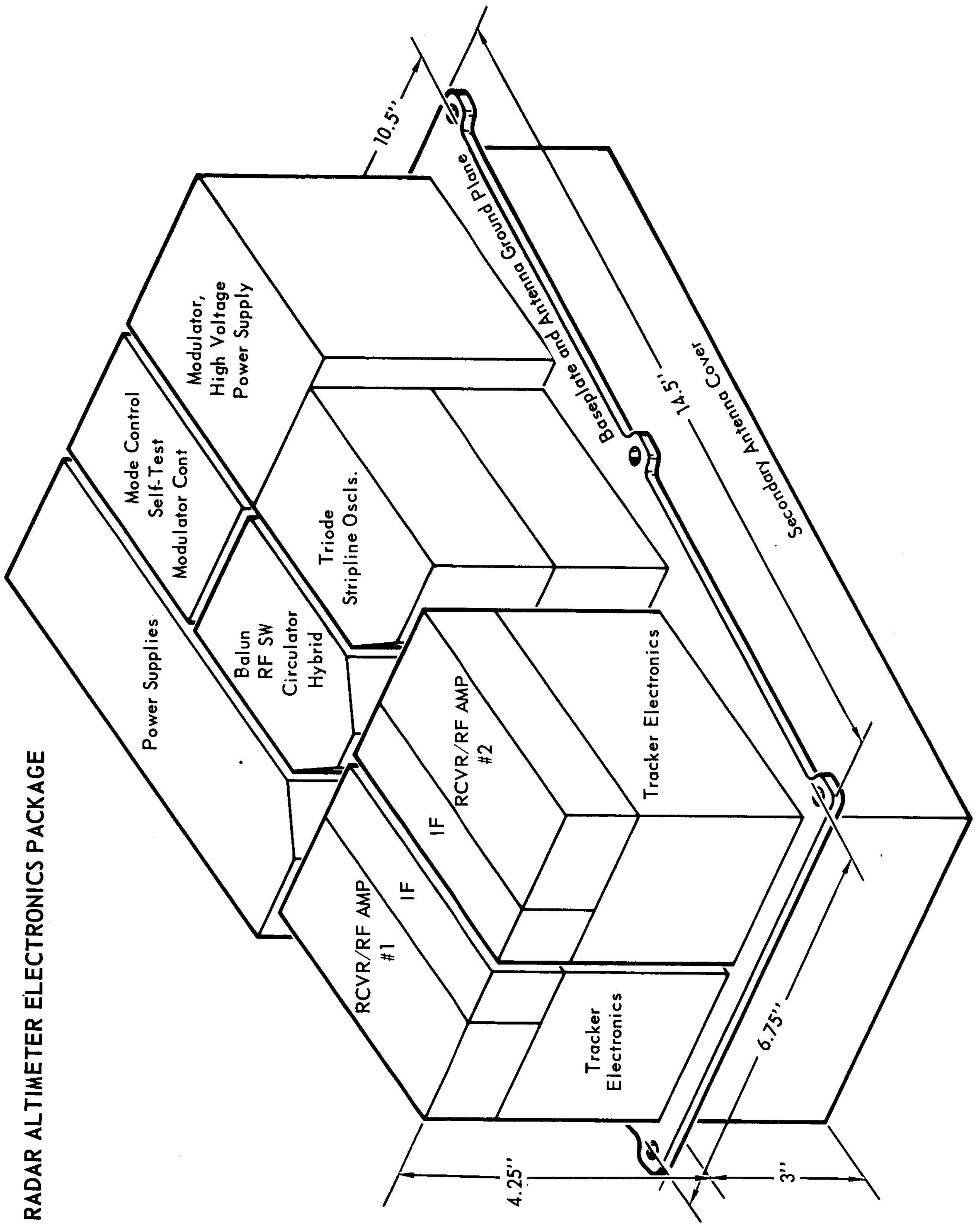


Figure 4.6-4

STAGNATION POINT INSTRUMENT HEAD

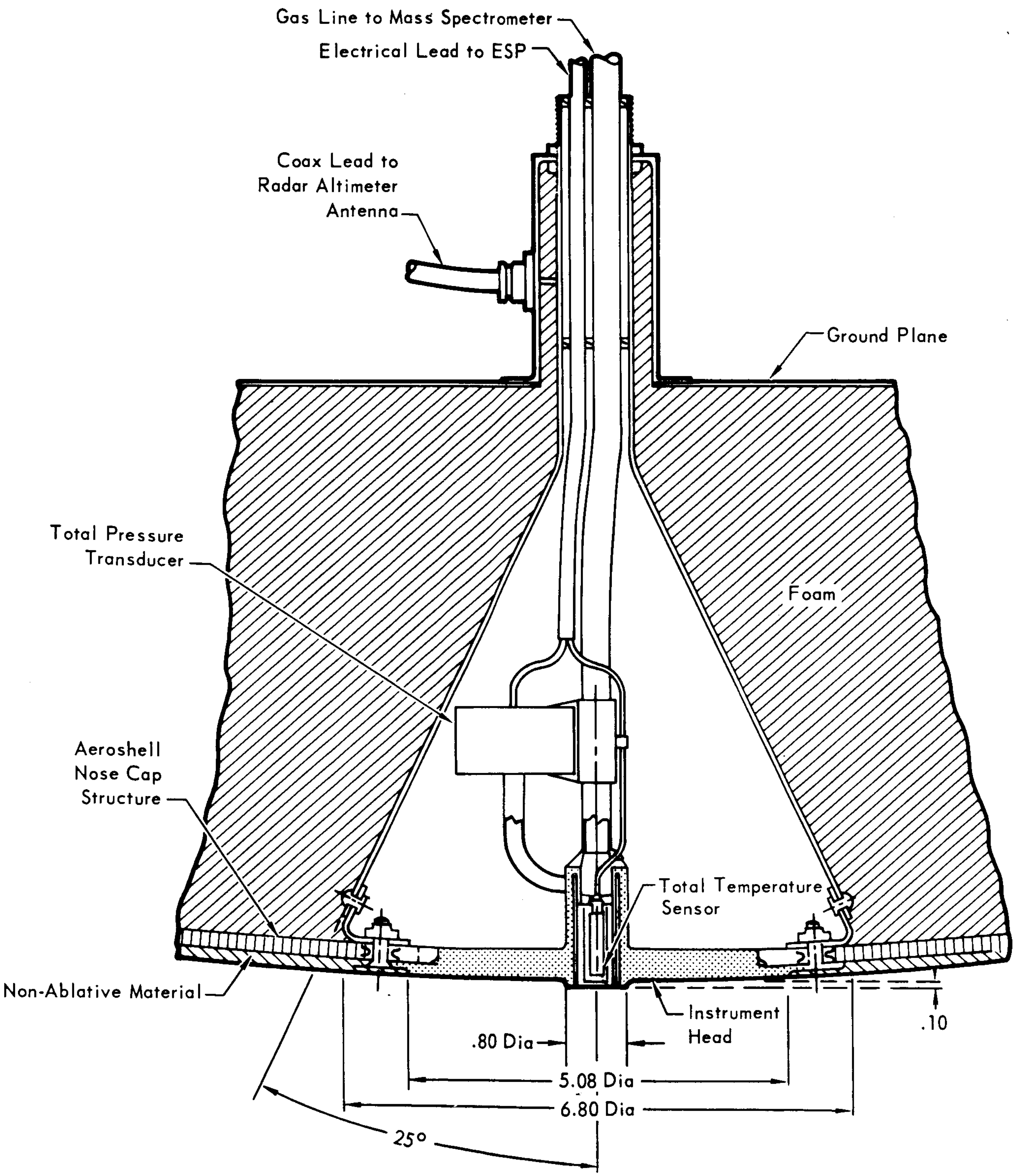


Figure 4.6-5

RADAR ALTIMETER FUNCTIONAL BLOCK DIAGRAM

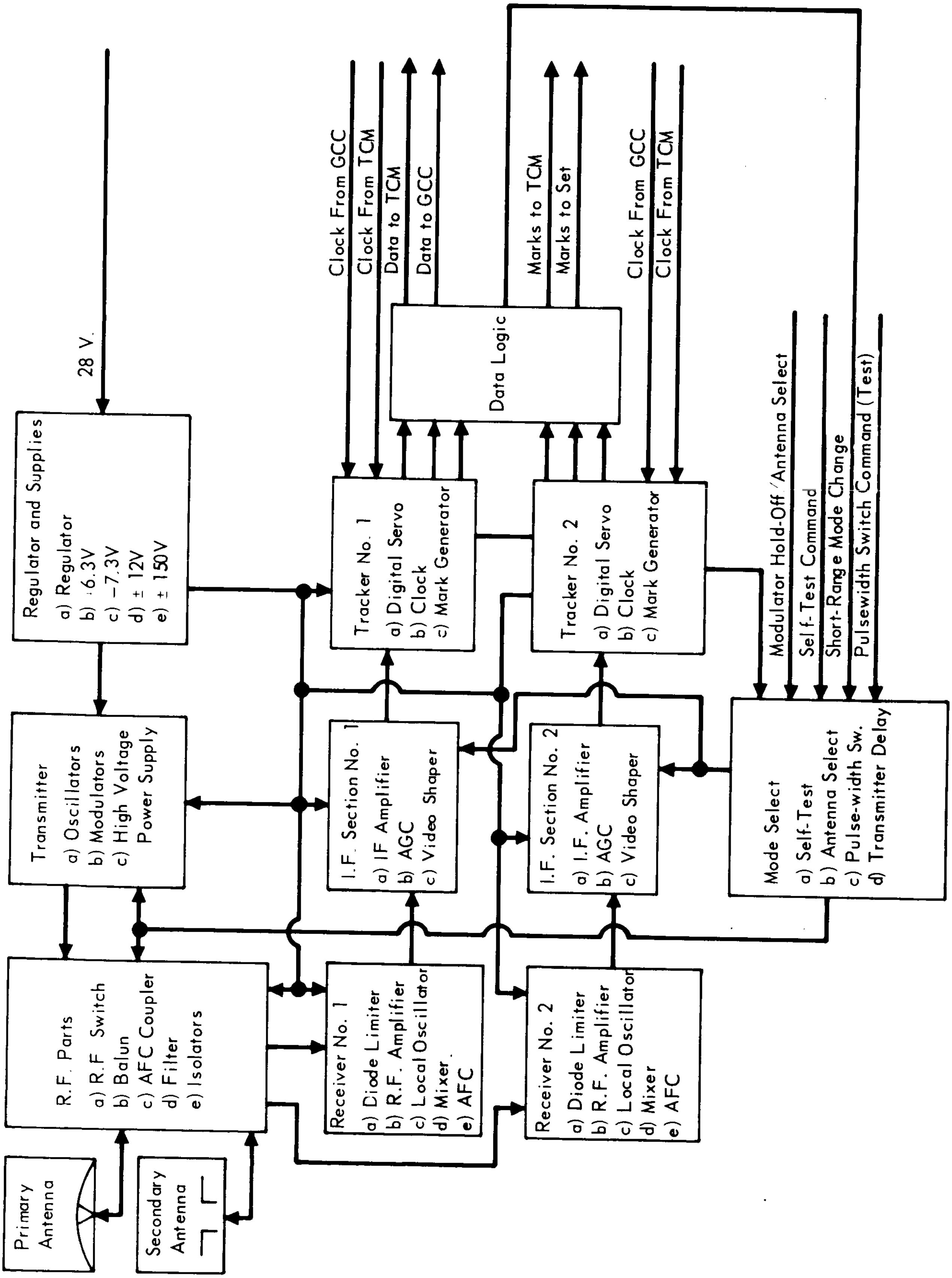


Figure 4.6–6

RADAR ALTIMETER FUNCTIONAL CHARACTERISTICS

CHARACTERISTIC	SYMBOL	VALUE
Frequency	f	1 GHz
Wavelength	λ	1 foot
Transmitter Repetition Frequency	f_R	10^3 Hz
Pulse Width	τ	5×10^{-6} seconds (Long Range) 1×10^{-7} seconds (Short Range)
Peak Transmitter Power	P_{pk}	500 watts (27 dBw)
Pulses Integrated (for acquisition)	N_i	25 – Long Range Mode 5 – Short Range Mode
Acquisition Sweep Time	T_{aq}	2 seconds
I.F. Bandwidth	B	2×10^5 Hz – Long Range 1×10^7 Hz – Short Range
Single Pulse Signal to Noise Ratio	S/N	3 db – Long Range Mode 27 db – Short Range Mode
Probability of Detection (Single-Sweep)	P_d	.95 – Long Range .99 – Short Range
False Alarm Time	T_{fa}	1 hour
System Noise Figure	N_f	6 db
System Losses (including Aeroshell)	L	6 db
System Noise Power	KT	4.14×10^{-21} watt-seconds (–204 dBw) (assumes 300°K system temperature)
Antenna Gain	G	1 (0 db) Long Range 1.59 (2 db) Short Range

Figure 4.6-7

Volume II, Part B, Section 5.9. This analysis shows a .99 or better cumulative probability of detection at 200,000 feet with a single pulse signal to noise ratio of 3 dB for the altimeter selected.

Basic accuracy of the altimeter over a flat, smooth surface, operating on a 3 dB signal-to-noise ratio at maximum altitude will be determined by calibration inaccuracies and thermal noise fluctuations. The magnitude of error can be shown to be about $\pm$ 720 feet (3 σ) for the analog tracking loop with the time constraints initially selected, and $\pm$ 320 feet (3 σ) for the digital tracking loop with different loop parameters. Other error sources that will degrade this performance are returned RF signal rise time fluctuations due to rough (but still flat) surfaces, delayed rise times caused by a null in the antenna pattern, and a deceleration error. The rise time fluctuation error is about equal in magnitude to the basic noise error at maximum range and is continually present. The deceleration error occurs during maximum deceleration and can be considered a fixed error during this time. Delayed rise time error occurrs during the 30,000 to 20,000 foot interval for VM-10 atmospheres. This error is caused by the null in the antenna pattern (about the roll axis) providing insufficient ground illumination. The magnitude of this error was determined by calculating the return signal rise time for mainlobe, and the pattern null illumination of the surface directly below the Capsule Bus. Figure 4.6-8 shows this error (ε) to be about 500 feet, due to a difference in the 50% amplitude point for the two cases of about 1 microsecond.

An estimate of the maximum error expected for the two tracking loop configurations is shown in Figure 4.6-9. This is a composite error in percent of measured altitude over the range of 200,000 to 500 feet, for all error sources previously identified. A significant error decrease can be seen in Figure 4.6-9 to ocurr shortly after parachute deployment and Aeroshell release at 20,000 feet, when any errors due to insufficient ground illumination are eliminated. Another decrease in error is seen when pulsewidth switching from 5 to 0.1 microsecond ocurrs at 5,000 feet. The total error is expected to be $\pm$ 25 feet (3 σ) from this point to 50 feet, where operation ends.

4.6.4 <u>Elevation Changes, Resolution and Surface Profile Determination</u> - The system errors shown in Figure 4.6-9 do not include the effects of resolution. Because of the broad surface area illuminated at altitudes from 200,000 to 5,000 feet by the 5 microsecond pulses, the altimeter cannot accurately resolve peaks and depressions within this area. An altitude will be measured which is approximately the distance above the mean elevation. Correlation with visual images obtained via TV will allow

RETURN SIGNAL RISETIME AND SURFACE ILLUMINATED vs TIME

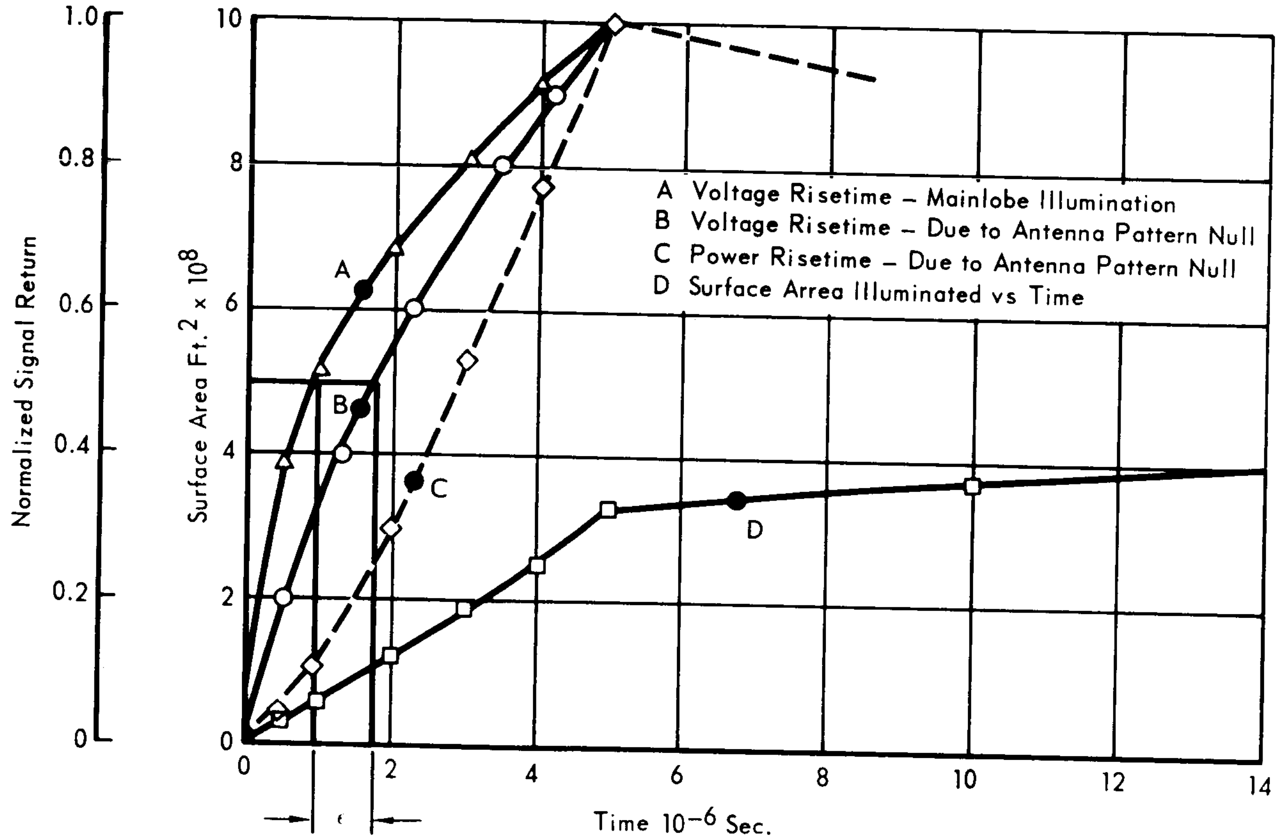


Figure 4.6–8

MAXIMUM TRACKING LOOP ERROR AS A FUNCTION OF ALTITUDE
(NON-OPTIMIZED LOOP PARAMETERS)

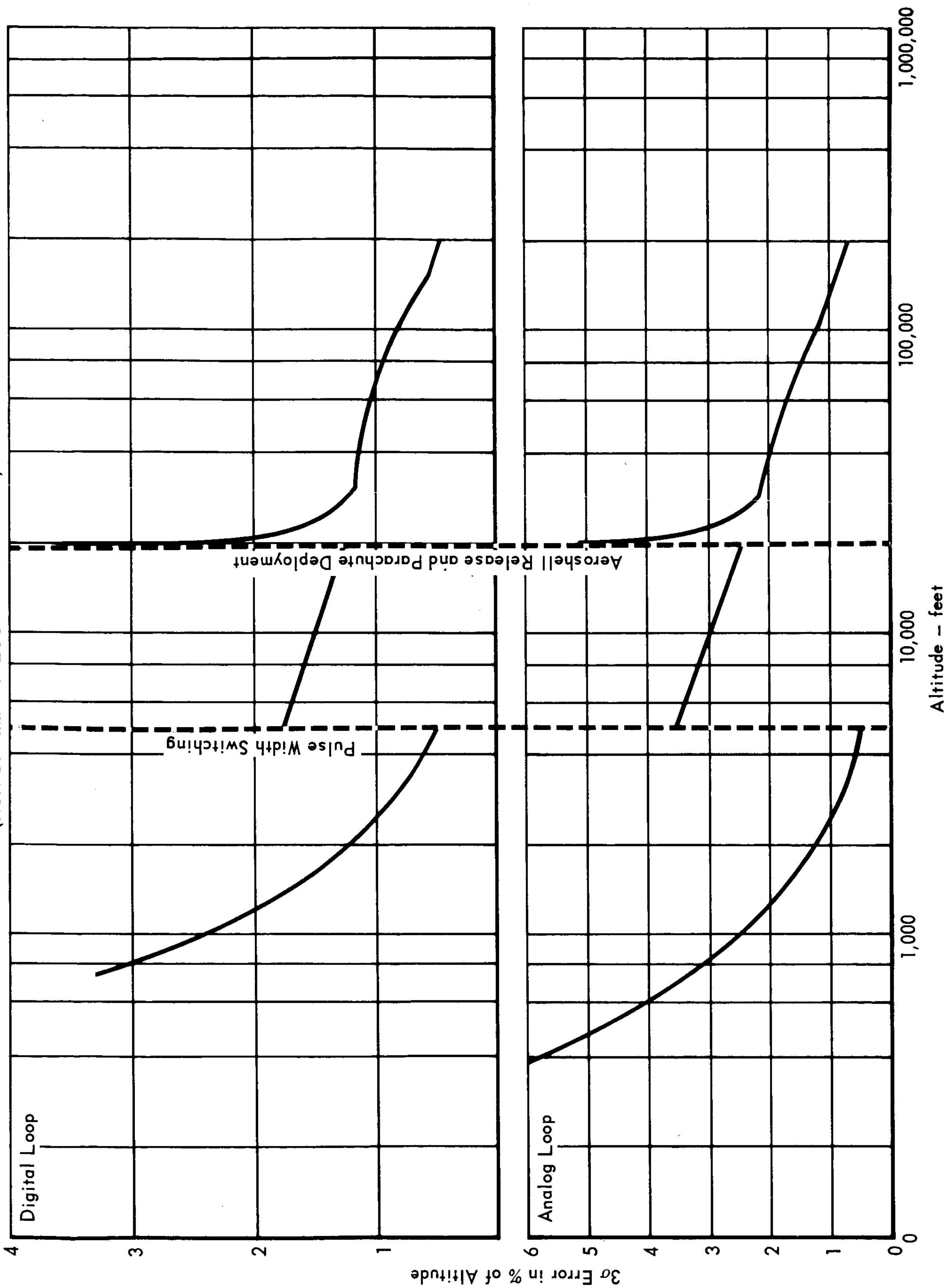


Figure 4.6– 9

the resolution of major surface disturbances to about 2,500 feet. Below 5,000 feet, the resolution improves to 50 feet due to pulsewidth switching. Improved high altitude resolution is possible if a short pulsewidth is used at high altitudes, with high gain antennas. Surface profiles could be obtained from this improved resolution, if desired. However, the narrow pulsewidth and high gain antenna combination requires vehicle stabilization or beam steering, which is not a desirable approach to the overall altimeter or Capsule Bus design.

4.6.5 Development Status and Developments Required - The preferred altimeter design is based on using space proven techniques and equipment, and previously developed improvements. The Saturn altimeter serves as a basis for the Radar Altimeter described herein. The Saturn altimeter program has demonstrated the ability of an L-Band, short-pulse altimeter using relatively wide-beam antennas to accurately measure high altitudes in a space environment. Saturn altimeter miniaturization and improvement study contracts have demonstrated digital AGC and tracking techniques, self-test features and pulsewidth switching using developed circuits. No state-of-the-art advances or new technology is required to obtain a usable radar altimeter to perform the VOYAGER mission.

4.7 LANDING AND IMMEDIATE POST LANDING OPERATIONS - Two minutes of the five minute baseline post-landed view period of the spacecraft will be available for ESP experiment measurements both as backup and adjunct to Surface Laboratory operations. The base region pressure and temperature sensors will obtain surface pressure and temperature data. The mass spectrometer inlet will be switched to a small capillary sampling tube to reduce the time constant associated with obtaining fresh atmospheric samples. The CBS and ESP interface during landing and post-landing operations consists of continuing the structural mounting support of the ESP equipment not previously jettisoned (stagnation region pressure and temperature sensors are separated along with Aeroshell, and TV cameras are jettisoned prior to landing) and the interleaving of ESP low rate data for transmission over the CBS communications link to the orbiting spacecraft.

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4-4: J. C. South, "Program Description of Approximate Solution for Supersonic Flow Past Blunt Bodies with Sharp Sonic Corners," Langley Working Paper 240.

4-5: 1973 VOYAGER Capsule System Constraints and Requirements Document (Rev. No. 2) JPL SE003BB002-2A21, 12 June 1967.

SECTION 5

MISSION OPERATIONS

The ESP mission operation requires a schedule of planned events, the means for in-flight checkout of ESP equipment, and available alternate events in case of equipment degradation. The nominal sequence of events planned for the mission is described in Section 5.1. The contingent/alternate paths in the event of malfunction are detailed in Section 5.2. Finally, Section 5.3 describes the means for equipment monitoring and checkout which will help determine whether a planned or alternate event will occur. The mission operation of the ESP is automatic. However, ESP turn-on time and the time delay following peak dynamic pressure before starting to use the mass spectrometer and the stagnation point temperature sensor can be reprogrammed in orbit prior to Capsule separation.

5.1 Entry Science Package Mission Sequence - The Entry Science Package nominal mission sequence is presented in Figure 5.1-1. The mission sequence is a detailed list of events that the Entry Science Package will perform in the successful execution of its 1973 mission. Important Flight Capsule events are included for reference purposes.

The Entry Science Package sequence consists of (1) an event listing, (2) the source of the primary signal for the initiation of each event, (3) the equipment destination of the signal, (4) the source of any backup initiating signal, and (5) the time of occurrence of each event.

5.2 Entry Science Package Operational Contingency Modes - In compliance with the design requirement that "no potential single failure mode shall cause a catastrophic effect on the mission," the Entry Science Package has a number of contingency modes incorporated into its design. The contingent/alternate paths for recourse to operation malfunctions are illustrated in Figure 5.2-1.

The contingency-mode flow chart contains the following information:

- A listing of the major operations in the ESP mission sequence.
- The source of the primary command for initiating the mission events.
- The corrective actions which are available, by design, to backup the ESP operations.
- The possible alternative actions which can be taken to offset any anomalous indications.
- The presence of block redundancy in the ESP equipment in cases where "no-mission" results are possible.

ENTRY SCIENCE PACKAGE 1973 MISSION SEQUENCE

EVENT	SIGNAL SOURCE	SIGNAL DESTINATION	BACKUP SIGNAL	TIME	REMARKS
PRELAUNCH TO CAPSULE IN-ORBIT CHECKOUT					
1. Turn on ESP Cruise Commutator	LOS	FC Pwr		$T_0 - 2$ hr	
2. Switch ESP Cruise Commutator to Cruise Mode	LOS	FC CC		$T_0 - 2$ hr	
3. Switch ESP to Internal Power	LOS	ESP Pwr		$T_0 - 7$ min	
4. Liftoff				T_0	T_0 = Liftoff
5. Parking Orbit Insertion				T_1	$T_1 = T_0 + 713.4$ sec
6. Trans-Mars Injection				T_2	$T_2 = T_1 + 37.8$ min
7. Separation of Forward Planetary Vehicle				T_3	$T_3 = T_2 + 2.6$ min
8. Separation of Aft Planetary Vehicle				T_4	$T_4 = T_3 + 15$ min
9. Switch ESP to FS Power	FS CC&S/Mos	ESP Pwr	MOS	$T_3 + 23$ min	
10. Begin ESP Battery Recharge	ESP Pwr	ESP Pwr	Mos	$T_3 + 23$ min	Automatic recharging during remainder of mission when CB receives power from FS
10. Arrival Date Separation Maneuver [1]				T_5	$T_5 = T_2 + 6.8$ days
11. First Midcourse Trajectory Correction Maneuver [1]				T_6	$T_6 = T_2 + 30$ days
12. Second Midcourse Trajectory Correction Maneuver				T_7	$T_7 = T_8 - 30$ days
13. Mars Orbit Insertion [1]				T_8	$T_8 = T_2 + 217.2$ days (minimum)
14. Mars Orbit Trim Maneuver [1]				T_9	$T_9 = T_8 + 4$ Orbits
CAPSULE IN-ORBIT CHECKOUT TO CAPSULE ENTRY					
1. Switch ESP Cruise Commutator to CB TM Control Mode	CB TP	ESP CC	MOS	$T_{10} - 5$ min	
2. Turn on CB TM Subsystem	CB TP	CB Pwr	MOS	$T_{10} - 5$ min	
3. Switch CB TM to Check-Out Mode	CB TP	CB TM	MOS	$T_{10} - 5$ min	
4. Pre-Separation CB Checkout				T_{10}	$T_{10} = T_{12} - 24$ hrs
5. Switch ESP TM Subsystem to Check-Out Mode	CB TP	ESP TM	MOS	$T_{10} + 122$ min	
6. Turn on ESP TM Subsystem	CB TP	ESP Pwr	MOS	$T_{10} + 122$ min	
7. Switch ESP Cruise Commutator to Cruise Mode	CB TP	ESP CC	MOS	$T_{10} + 129$ min	
8. Turn off ESP TM Subsystem	CB TP	ESP Pwr	MOS	$T_{10} + 129$ min	
9. Turn off CB TM	CB TP	CB Pwr	MOS	$T_{10} + 129$ min	
10. Enable SL Power Reserve for ESP	CB S&T	SL Pwr	MOS	$T_{12} - 64$ min	
11. Switch ESP to Internal Power	CB S&T	ESP Pwr	MOS	$T_{12} - 64$ min	
12. Deploy FS-Mounted UHF High-Gain Antenna	FS CC&S	FS Pyro	MOS	$T_{12} - 53$ min	
13. Turn on CB TM	CB S & T	CB Pwr	MOS	$T_{12} - 6$ min	
14. Switch ESP Cruise Commutator to CB TM Control Mode	CB S & T	ESP CC	MOS	$T_{12} - 7$ min	

Figure 5.1-1

15. Sterilization Canister Separation	CB S & T			T_{11}	$T_{11} = T_{12} - 5$ min
16. Spacecraft – Capsule Separation				T_{12}	$T_{12} = T_9 + 7\frac{1}{2}$ orbits (minimum when Mars-orbit trim occurs)
17. Ignite CB De-orbit Motor	CB S & T	CB TM		T_{13}	$T_{13} = T_{12} + 20$ min
CAPSULE ENTRY THROUGH CAPSULE LANDING					
1. Turn on FS–Mounted High-Rate UHF Receiver	FS CC&S	FS Pwr	MOS	$T_{14} - 10$ min.	
2. Turn on Descent TV (Warmup)	CB S & T	ESP Pwr	.05g Sensor CB RA	$T_{14} - 5$ min	
3. Entry				T_{14}	h = 800,000 ft $T_{14} = T_{13} + X$ hrs, $X \sim 4$; X is updated prior to FS–FC separation.
4. Turn on ESP TM Subsystem	CB S & T	ESP Pwr	.05g Sensor CB RA	T_{14}	
5. Switch ESP TM Subsystem to Entry Mode	CB S & T	ESP TM	.05g Sensor CB RA	T_{14}	
6. Turn on Base Pressure Experiment	CB S & T	ESP Pwr	.05g Sensor CB RA	T_{14}	
7. Turn on Accelerometer Experiment	CB S & T	ESP Pwr	.05g Sensor CB RA	T_{14}	
8. Turn on Stagnation Pressure Experiment	CB S & T	ESP Pwr	.05g Sensor CB RA	T_{14}	
9. Turn on ESP Radio Subsystem	CB S & T	ESP Pwr	.05g Sensor CB RA	T_{14}	
10. Turn on ESP Data Storage Subsystem	CB S & T	ESP Pwr	.05g Sensor CB RA	T_{14}	
11. Initiate Descent TV Sequencing	CB S & T	ESP Sci	.05g Sensor CB RA	T_{14}	
12. Sense .05g Deceleration Level				T_{15}	
13. Begin Radar Altimeter Tracking				T_{16}	h = 200,000 ft
14. Landing Radar Turnon				T_{17}	h = 100,000 ft
15. Signal Mach 5	ESP Sensor			T_{18}	h > 30,000 ft 30 sec after peak stagnation pressure.

16. Initiate Mass Spectrometer Experiment	ESP Sensor	ESP Sci	CB RA	T_{18}	
17. Initiate Stagnation Temperature Experiment	ESP Sensor	ESP Sci	CB RA	T_{18}	
18. Initiate Base Temperature Experiment	ESP Sensor	ESP Sci	CB RA	T_{18}	
19. Deploy Aerodecelerator (Parachute)				T_{19}	h = 23,000 ft
20. Separate CB Aeroshell				T_{20}	h ~ 15, 500 ft;
21. Switch ESP TM Subsystem to Terminal Descent Mode	CB Sensor	ESP TM		T_{20}	$T_{20} = T_{19} + 12$ sec.
22. Ignite Terminal Propulsion Motors				T_{21}	h = 5,000 ft
23. Intercept Programmed CB Decelerator Profile				T_{22}	
24. Landing Radar Scale Change				T_{23}	h = 2,500 ft
25. Release Descent TV Cameras	CB RA	CB Pyro	CB L R	T_{24}	h = 90 ft
26. Switch to Inertial Attitude Control Over CB				T_{25}	V_z = 10 fps
27. Begin Constant Velocity Descent				T_{25}	
28. Terminate CB Radar Altimeter Operation				T_{25}	h = 50 Ft
29. Terminate CB Terminal-Propulsion – Motor Burn				T_{26}	h = 10 ft
30. Touchdown on Surface of Mars				T_{27}	
31. Shut Down Entry Science Package (End of ESP Mission)	CB S & T	ESP Pwr		T_{26} + 10 min	

TIME DETONATIONS

T_0 – Liftoff
T_1 – Parking-Orbit Insertion
T_2 – Trans-Mars Injection
T_3 – Separation of Forward Planetary Vehicle
T_4 – Separation of Aft Planetary Vehicle
T_5 – Arrival Date Separation Maneuver
T_6 – First Interplanetary Midcourse Correction
T_7 – Second Interplanetary Midcourse Correction
T_8 – Mars-Orbit Insertion
T_9 – Mars-Orbit Trim
T_{10} – Pre-Separation FC Checkout
T_{11} – Sterilization Canister Separation
T_{12} – FS–FC Separation
T_{13} – Ignition of De-orbit Motor
T_{14} – Mars (800,000 ft.) Entry
T_{15} – Sense .05g Deceleration Level
T_{16} – Begin Radar Altimeter Tracking
T_{17} – Landing Radar Turnon
T_{18} – Sense Mach 5
T_{19} – Deploy Aerodecelerator
T_{20} – Separate CB Aeroshell
T_{21} – Ignite Terminal Propulsion Motors
T_{22} – Intercept Programmed Descent Profile
T_{23} – Landing Radar Range Scele Change
T_{24} – Begin Constant–Velocity Descent
T_{25} – Terminate CB Radar Altimeter Operation
T_{26} – Terminate Terminal Thrust
T_{27} – Touchdown on Surface of Mars

ABBREVIATIONS

CB – Capsule Bus
CC – Cruise Commutator (ESP, CB, SL, or FC Adapter)
CC & S – Central Computer and Sequencer (FS)
ESP – Entry Science Package
FC – Flight Capsule
FS – Flight Spacecraft
LOS – Launch Operation System
MOS – Mission Operations System
Pwr – Electrical Power Subsystem (FS, CB, ESP or SL)
RA – Radar Altimeter (CB)
SL – Surface Laboratory
Sci – Science Subsystem (ESP or SL)
TM – Telemetry Subsystem (ESP or SL)
TP – Test Programmer (CB or SL)

1. ESP Switched to Internal Power Before PV Maneuvers and Switched Back to Power After PV Maneuvers.

ENTRY SCIENCE PACKAGE CONTINGENCY MODES
LIFTOFF THROUGH END OF ESP MISSION

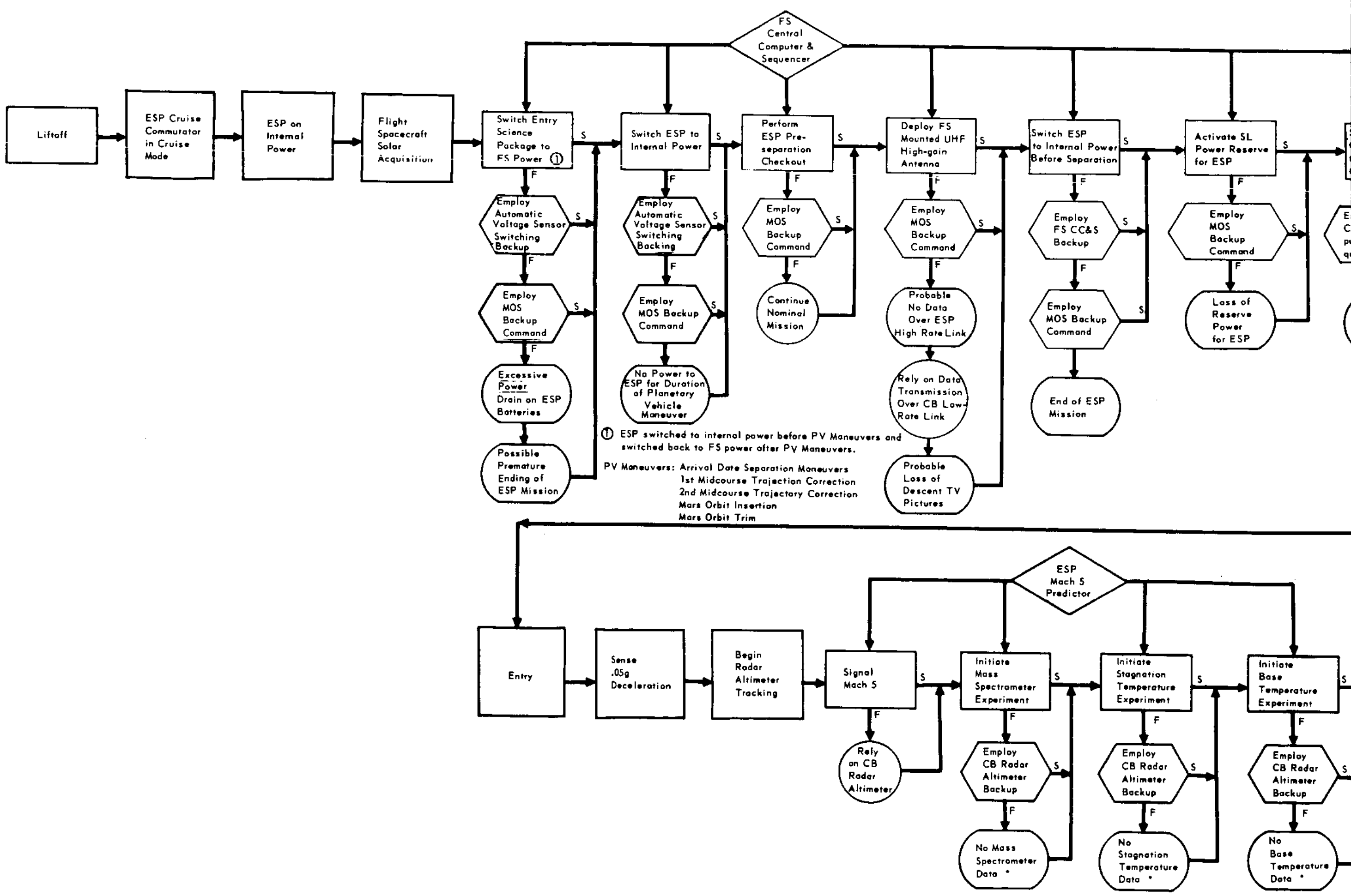
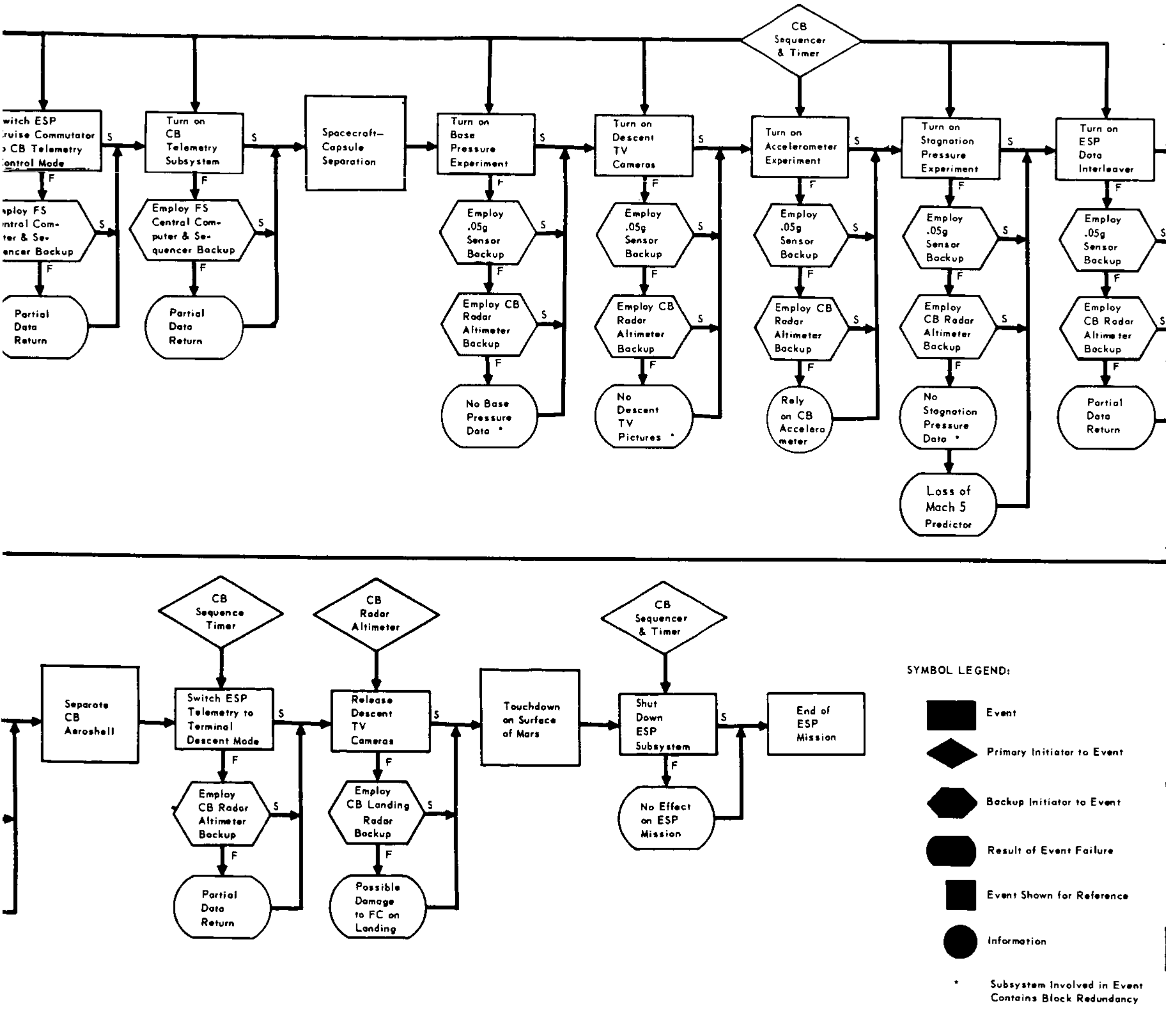
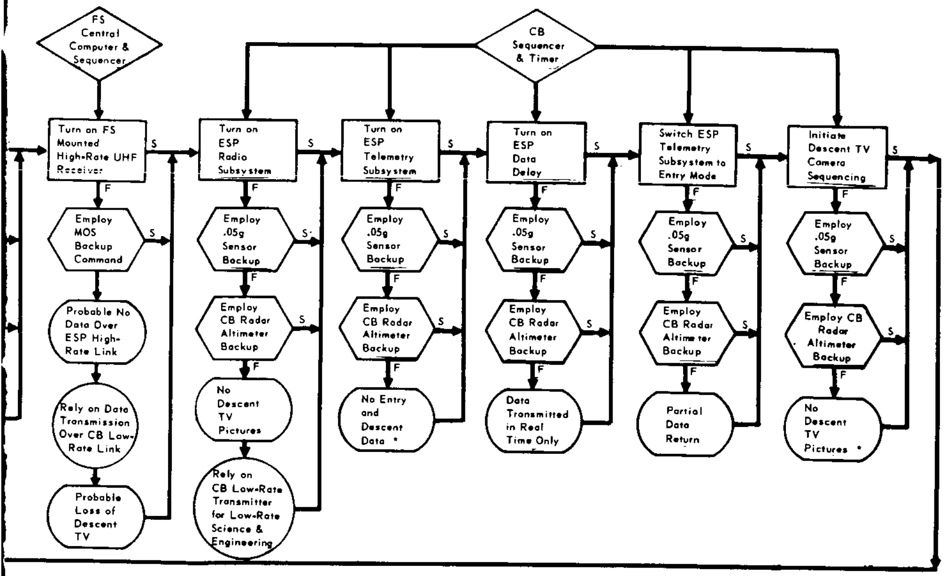


Figure 5.2-1


CB Sequencer & Timer
witch ESP
ruise Commutator
CB Telemetry
ontrol Mode
S
F
ploy FS
ntral Com-
ter & Se-
encer Backup
S
F
Partial Data Return
Turn on CB Telemetry Subsystem
S
F
Employ FS Central Computer & Sequencer Backup
S
F
Partial Data Return
Spacecraft–Capsule Separation
Turn on Base Pressure Experiment
S
F
Employ .05g Sensor Backup
S
F
Employ CB Radar Altimeter Backup
S
F
No Base Pressure Data *
Turn on Descent TV Cameras
S
F
Employ .05g Sensor Backup
S
F
Employ CB Radar Altimeter Backup
S
F
No Descent TV Pictures *
Turn on Accelerometer Experiment
S
F
Employ .05g Sensor Backup
S
F
Employ CB Radar Altimeter Backup
S
F
Rely on CB Accelerometer
Turn on Stagnation Pressure Experiment
S
F
Employ .05g Sensor Backup
S
F
Employ CB Radar Altimeter Backup
S
F
No Stagnation Pressure Data *
Loss of Mach 5 Predictor
Turn on ESP Data Interleaver
F
Employ .05g Sensor Backup
S
F
Employ CB Radar Altimeter Backup
S
F
Partial Data Return
CB Sequence Timer
CB Radar Altimeter
CB Sequencer & Timer
Separate CB Aeroshell
Switch ESP Telemetry to Terminal Descent Mode
S
F
Employ CB Radar Altimeter Backup
S
F
Partial Data Return
Release Descent TV Cameras
S
F
Employ CB Landing Radar Backup
S
F
Possible Damage to FC on Landing
Touchdown on Surface of Mars
Shut Down ESP Subsystem
S
F
No Effect on ESP Mission
End of ESP Mission
SYMBOL LEGEND:
Event
Primary Initiator to Event
Backup Initiator to Event
Result of Event Failure
Event Shown for Reference
Information
* Subsystem Involved in Event Contains Block Redundancy



ABBREVIATIONS:
- CB – Capsule Bus
- ESP – Entry Science Package
- FC – Flight Capsule
- MOS – Mission Operations System
- PV – Planetary Vehicle
- S – Success
- F – Failure

An example of the use of partial functional redundancy in the Entry Science Package to provide a contingent mode of operation is found in the turn-on of the ESP for start of operations. In the normal sequence of events, the ESP subsystems (other than its cruise commutators and automatically operating heaters) are turned on by the Capsule Bus Sequencer and Timer (S&T) at entry, 300 seconds prior to reaching 800,000 feet above the mean Martian surface. As illustrated in Figure 5.2-1 two alternate back-up initiating sources are available in the Capsule Bus. The first of these involves the use of the CB guidance and control accelerometer readings. When a .05 g deceleration level is sensed, the ESP will be given a turn-on command. The altitude range for initiation would then be from 180,000 feet to 480,000 feet depending on the atmosphere encountered. This back-up is partially redundant, but is still dependent upon a prior enabling command to the accelerometers.

The second alternate means for initiating the Entry Science Package makes use of the CB radar altimeter. Here, the initiating signal would be given to the ESP at an altitude of 100,000 feet, making use of the same signal used for turn-on of the Landing Radar. Here again, this is only partially redundant, being dependent upon a prior enabling command to the altimeter.

In both cases, some mission degradation will occur due to the lower altitudes at which the ESP is turned on. However, ESP turn-on occurs at a sufficiently high altitude to perform a mission and acquire much of the desired entry and descent data.

5.3 In-Flight Monitoring and Checkout - An ESP in-flight status monitor/checkout/control plan has been developed in parallel with the ESP engineering and science design and has been integrated into the overall Mission Support Plan. The automatic monitor/checkout activity includes:

- Continuous passive monitoring from Earth launch through Mars orbit (interplanetary cruise).
- Science instruments and engineering subsystems activation and performance checkout prior to Flight Capsule/Spacecraft separation.
- Monitoring of cruise parameters from separation to entry (800,000 feet altitude), but at 10 times faster sampling rate.
- Continuous monitoring of all subsystem operational parameters during the atmospheric descent profile.

All of this data is automatically generated and telemetered by the Spacecraft to the Earth stations, where the ESP mission operations personnel analyze and judge the integrity and/or performance of the equipment. These same personnel have recourse to corrective/preventive equipment control actions or mission sequence modifications via Earth to Spacecraft command, until Flight Capsule separation from the Spacecraft.

Figures 5.3-1, -2 and -3 present functional descriptions of the status monitor/checkout/control activities for all ESP mission phases. It is noted that data is continuously gathered on the ESP equipment; cruise parameter monitoring continues in orbit both before and after the pre-separation checkout period. The methods of all data reception, distribution, and analyses by the mission operations personnel at the Deep Space Instrumentation Facility and Space Flight Operations Facility are discussed in detail in Section J4.5.

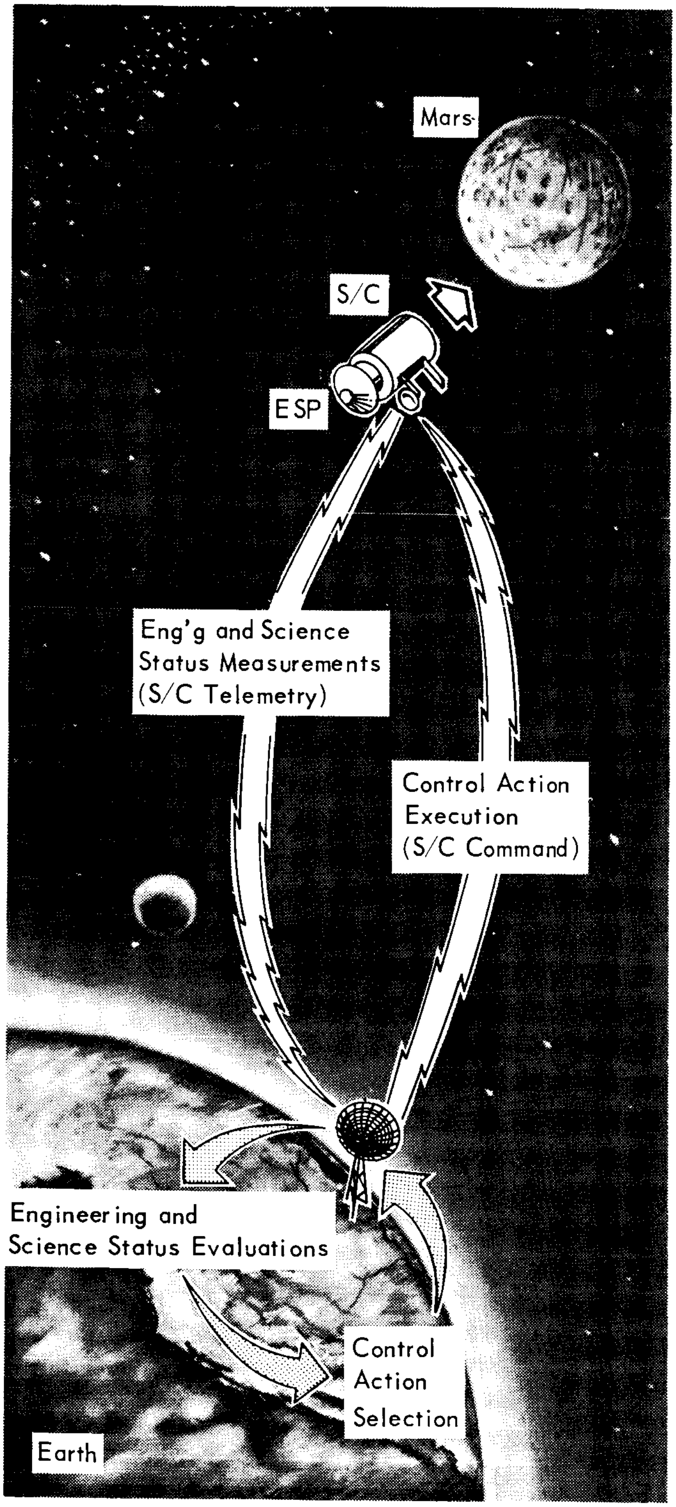
5.3.1 Test Purpose and Selection Criteria - The purpose of ESP in-flight monitoring and checkout is to maximize the probability of ESP and overall mission success. The condition of the ESP, as determined from the monitor and checkout data, is used to determine command actions which safeguard equipment and/or modify the final mission sequence. The test selection criteria establish the tests necessary to perform this status evaluation function with proven engineering techniques.

Interplanetary Cruise Monitoring - Continuous passive monitoring of equipment temperatures and the electrical power subsystem status are required to maintain confidence of ESP survival during transit to Mars. The confidence level of the received data is established by having the telemetry unit also monitor its own operating conditions. In the event of off-normal conditions or malfunctions, the ground personnel can elect contingent status control actions to correct the problem condition or assess the remaining mission capability. Early detection will result in the best control action selection.

Pre-Separation Checkout Tests - Prior to the Flight Capsule/Spacecraft separation the ESP engineering and science equipment will be activated and tested under simulated mission inputs (where practical). The test sequence and equipment operation is under the control of an onboard, pre-programmed, automatic Test Programmer. Equipment operational parameters and test responses are evaluated by the mission operations personnel and are used to:

- Determine the operational performance of all ESP subsystems prior to mission commitment
- Fault isolate to the experiment or engineering subsystem module level
- Select ESP back-up modes of operation and/or mission event timing changes
- Select Capsule Bus and/or Surface Laboratory mission modifications
- Provide correlation data to facilitate post-flight analyses and compare with pre-launch calibration data.

ESP INTERPLANETARY CRUISE STATUS MONITOR/CONTROL PLAN



Engineering and Science Equipment Status Measurements

- Continuous Passive Monitoring

Status Evaluation (Mission Operations Personnel)

- Engineering and Science Integrity Verification
- Impending Failures or Failures
- Ground Equipment Accuracy Verification

Control Action Selection (Mission Operations Personnel)

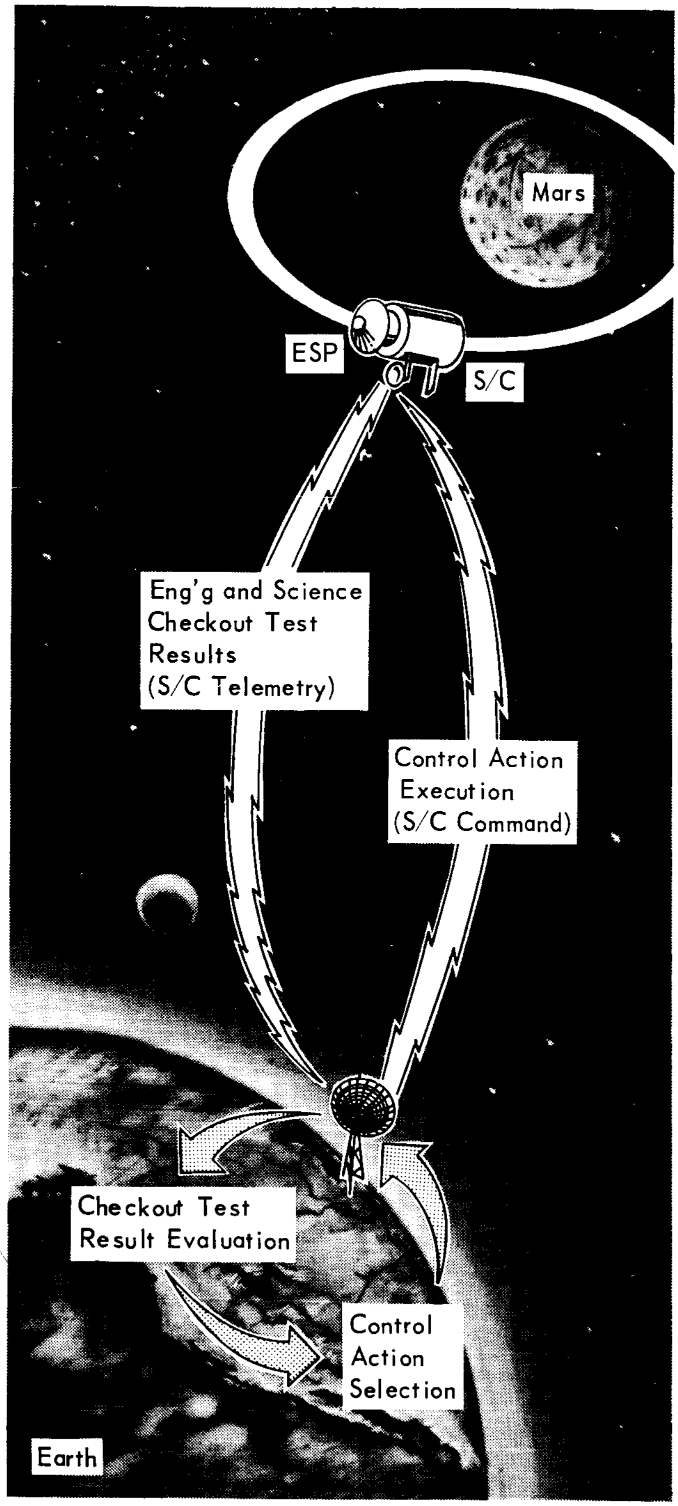
- Choose Equipment Corrective/Preventive Action
- Select Mission Contingency Plan
- Check Ground Equipment

Control Action Execution

- Send Command to Spacecraft-to-Capsule Bus-to-ESP
- Send Command to Spacecraft-to-Capsule Bus-to-ESP

Figure 5.3-1

ESP PRE-SEPARATION CHECKOUT TEST/CONTROL PLAN



ESP Checkout Test Results

- Equipment Simulated Inputs and Responses Monitored
  - Test Sequence Automatically Controlled by the Capsule Bus Test Programmer

Checkout Test Result Evaluation

(Mission Operations Personnel)

- Equipment Operation/Calibration Verification
- Fault Isolation
- Ground Equipment Accuracy Verification

Control Action Selection (Mission Operations Personnel)

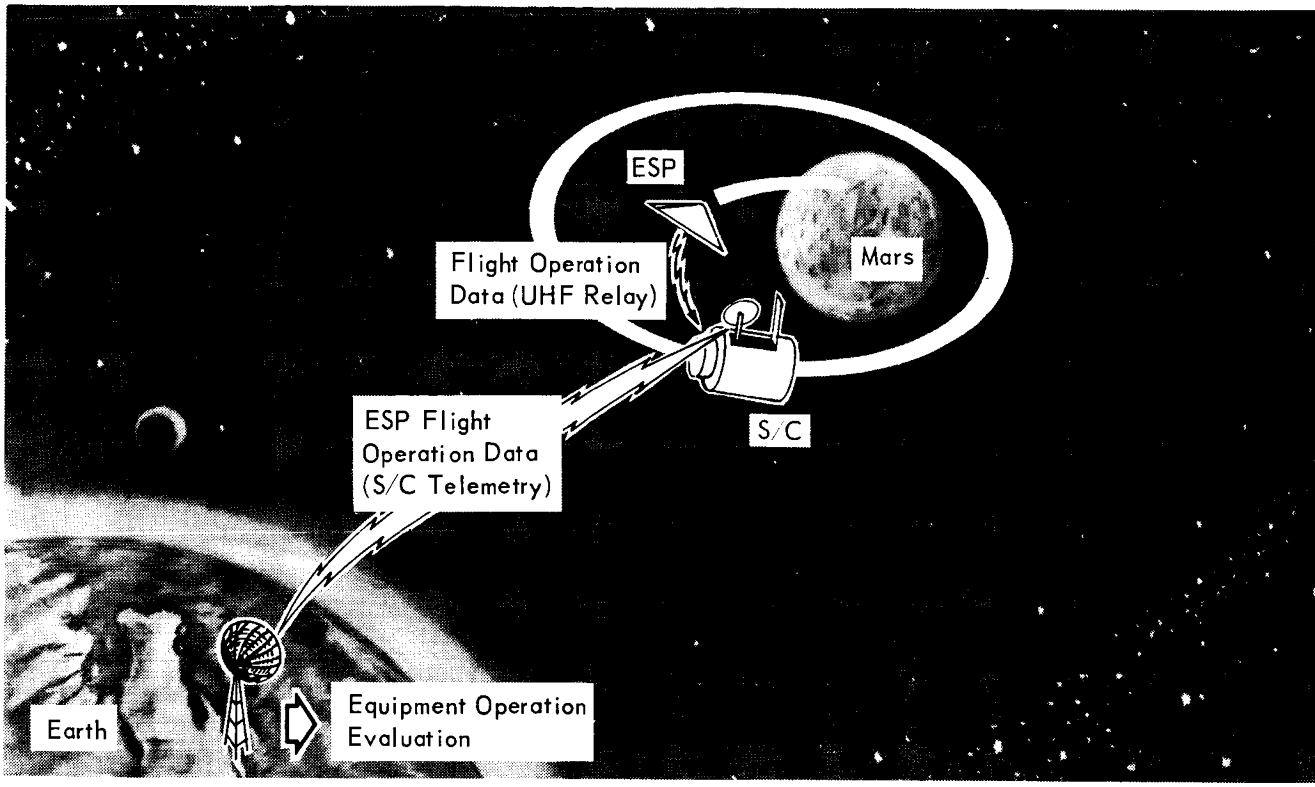
- Select Flight Capsule Mission Contingency Plan
- Change ESP Mission Sequence
- Repeat Particular Checkout Test

Control Action Execution

- Send Command to Spacecraft -to-Capsule Bus -to-ESP
- Send Command to Spacecraft-to-Capsule Bus
- Send Command to Spacecraft-to-Capsule Bus Test Programmer
- Send Command to Spacecraft-to-Surface Laboratory

Figure 5.3-2

ESP MISSION FLIGHT OPERATION MONITOR PLAN



Equipment Flight Operation Results

- Monitor Cruise Parameters from Separation to Entry (800,000 ft Altitude)
- Continuous Monitor of ESP Operational Parameters After Entry

Equipment Operation Evaluation (Mission Operations Personnel)

- Record Data for Post-Flight Analyses
- Cursory Atmospheric Model Construction and Imagery Analysis to Update Entry Profile of Second Flight Capsule

Figure 5.3-3

The test details are constrained by the requirements for minimizing reliability degradation, test equipment complexity, and consumption of mission power as a result of testing.

An extensive equipment built-in, self-test capability is required for remote pre-launch checkout after Flight Capsule sterilization. Many of the same equipment capabilities will be employed again to attain considerable in-flight test depth.

Mission Operation Monitoring - The cruise parameter commutation speed is increased by a factor of 10 from separation to entry (800,000 feet altitude). This passive monitoring is sufficient since the ESP is not activated until entry; the sampling rate is increased to measure the effects of Capsule Bus activities on the ESP.

Critical operational parameters of all the engineering and experiment subsystems are continuously monitored during the science data gathering mission phase. This data will be used to:

- Establish the confidence level of the transmitted science data
- Determine any desirable experiment modifications (i.e., response time, dynamic range, etc.) to increase measurement sensitivity and capability on future missions
- Isolate fault causes between the interplanetary cruise and atmospheric entry environments.

The equipment operational parameters monitored during entry operation are largely the same as those monitored during the checkout tests, which utilized simulated mission inputs. The sampling rates of some of the parameters are increased during mission flight to detect transient conditions over wider input ranges.

5.3.2 Subsystem/Experiments Design Implications - The decision to include continuous passive monitoring during interplanetary cruise and the interval from post-separation to entry has resulted in the design of a special purpose, dual mode telemetry commutator. This cruise telemetry commutator is a low power, high reliability unit using proven design techniques. The equipment telemetry transducers are simple in design and few in number for this function.

The decision to include the onboard dynamic subsystems checkout test capability has resulted in a parallel design requirement for subsystem/experiment compatibility with the required synthetic test input stimuli, the Test Programmer, and the Telemetry Subsystem. Due to this early integration on the test planning, the equipment designers have played an important role in deciding on each of the test parameters, the special test equipment design, and interface definitions. This onboard, automated test capability is also required for remote pre-launch equipment checkout after Flight Capsule sterilization. The equipment test stimuli generators have been chosen according to each selected pre-launch and in-flight test on each element and are self-contained within the primary equipment. This internal packaging concept has been chosen to minimize equipment/test stimuli design compatibility and integration problems. The ESP equipment is required to interface with the Capsule Bus Test Programmer. This Test Programmer automatically commands the ESP elements into their test states and cycles the test stimuli according to a pre-programmed event/time schedule (test sequence). The data requirements to evaluate the equipment test responses and validate the proper test stimuli and test programmer outputs have been included in the analyses to determine the Telemetry Subsystem modes and capacity.

No additional telemetry channels are required for the selected equipment parameters during the entry phase; those utilized for pre-separation checkout are employed again.

5.3.3 ESP Monitor and Checkout Test Descriptions/Discussion - Figure 4.2-2 of Section E4.2 (ESP telemetry instrumentation list) presents the data which are transmitted to Earth during each ESP mission phase. Also shown are the accuracy of these measurements and their nominal sampling period and rates. These five different data acquisition modes provide continuous ESP subsystems data; the cruise data continues in Mars orbit both before and after the preseparation checkout tests. The use of the Cruise Commutator during the de-orbit cruise eliminates the transmission of operational parameter data on inactive systems during this approximately 5 hour period.

Figures 5.3-4 and 5.3-5 present functional descriptions and test objectives for all the dynamic checkout tests performed on each subsystem and science instrument during the pre-separation phase. The actual subsystem and sensor data gathered during these tests is listed in Figure 4.2-2 of Section E4.2 under the heading of pre-separation checkout.

The total time to perform all the described ESP checkout tests is 6 minutes. All the science experiments are checked concurrently for 5 minutes on Spacecraft power at a power rating of 107 watts (200 watt limitation). The UHF Radio Subsystem test takes one minute and is performed on internal battery power at a level of 280 watts; all other elements on during the test (i.e., Spacecraft receiver, test programmer, etc.) are on Spacecraft power at a level of 45 watts. This test consumes approximately two percent (2%) of the ESP battery capacity. This battery energy is regained after testing by putting the battery back into the charge mode.

ESP ENGINEERING SUBSYSTEM PRE-SEPARATION CHECKOUT TESTS

TEST NAME	TEST DESCRIPTION	TEST OBJECTIVE/CORRECTIVE ACTION/ SPECIAL REMARKS
UHF Relay Radio & Antenna Subsystems ESP-to-Spacecraft-to-Earth Verification of ESP High Data Rate Radio Link	• Special test words (simulated data) are used to modulate the high rate transmitter. • The ESP antenna radiates RF into a parasitic antenna (which is left with spacecraft at separation). • The parasitic signal is coupled into the spacecraft high rate UHF receiver behind the spacecraft receiving antenna via a coaxial directional coupler • Spacecraft telemetry sends received test words to earth. • Low rate test word checks low rate data interleaving onto high rate link.	• Earth reception of special test words verifies the ESP/Spacecraft/Earth relay high data rate radio link (except the spacecraft high rate UHF receiving antenna) and the low rate data interleaving on this link. • Telemetered engineering parameters allow fault isolation between ESP transmitter and Spacecraft receiver. • Low rate data cross-strap technique provides all low rate data even if complete failure of low rate link. • Loss of high rate link means loss of all entry TV pictures – could delay orbital descent in favor of de-orbiting other flight capsule first.
Electrical Power Subsystem Battery Condition Check	• The battery temperature and open/closed circuit output voltage/current and charging current are monitored. • Same performance parameters as monitored during interplanetary cruise – sampling rates are increased during checkout because they are used to furnish power for UHF radio test described above.	• Verifies battery charge state, cell status, and detects any self-discharge. • This data allows correlation with other test data to isolate cause of failure (i.e., out-of-tolerance voltage). • Surface laboratory power can be distributed to ESP in event of power failure with subsequent decreased landed mission energy capacity.
Thermal Control Subsystem Heaters Performance Monitor	• ESP thermal control subsystem consists of two constant power electrical heaters – one to ESP module and one to common entry TV case. • Eight strategic ESP structure temperatures are monitored. • Same performance parameters as monitored during interplanetary cruise.	• Monitored structural temperatures verify electrical heaters performance or fault isolates • The continuous interplanetary cruise log of these elements performance can be used to correlate with Entry TV checkout data if TV camera degradation is observed.
Telemetry Subsystem Linearity Test	• Multi-level reference voltages are applied to all low level amplifiers and A/D converters to map the linearity (accuracy) of these signal conditioners over their entire input range.	• Test data indicates how to bias interpretation of observed systems data if linearity degradation is observed. • This test also performed during interplanetary cruise and orbital descent and entry
Data Storage Subsystem Data Storage/Delay Checkout	• All low rate science experiments checkout data is also stored and delayed transmitted by 50 seconds and 150 seconds.	• This test verifies the data storage and delayed transmission function • The real time low rate science data is compared with the 50 second delay data to verify the data storage/delay accuracy. • Loss of data storage will result in complete loss of data through blackout regime.

Figure 5.3-4

ESP SCIENCE INSTRUMENTS PRESEPARATION CHECKOUT TESTS

TEST NAME	TEST DESCRIPTION	TEST OBJECTIVE/CORRECTIVE ACTION/ SPECIAL REMARKS
Television		
1. Camera Calibration (2 Cameras)	• Each camera is turned on and generates a picture of an internal illuminated target grid. • Camera operational parameters are monitored during test • Picture of the test pattern is sent to earth • Test target grid is behind camera shutter and lens.	• Provides vidicon transfer function and distortion check. • Does not test camera lens and shutter control.
Descent Atmosphere Properties		
1. Stagnation Temperature Probe Check	• Turn on sensor and measure ambient temperature.	• Coarse sensor calibration by comparing probe output with structural temperature gauge outputs in near vicinity.
2. Stagnation Pressure Transducer Check	• Turn on transducer and monitor output in known ambient vacuum environment in orbit	• Measurements in known vacuum environment provide reference against which flight measurements can be compared – provides sensor bias.
3. Base Temperature Transducer Check	• Turn on sensor and measure ambient temperature	• Coarse sensor calibration by comparing probe output with structural temperature gauge outputs in near vicinity.
4. Base Pressure Transducer Check	• Turn on transducer and monitor output in known ambient vacuum environment in orbit	• Measurements in known vacuum environment provide reference against which flight measurements can be compared – provides sensor bias.
5. Science Accelerometers Checkout	• Calibrated current source is applied to auxiliary torquer of pulse rebalance accelerometers. • Two accelerometers on Z-axis for dual range capability.	• Verifies accelerometers operation. • Failure of accelerometers implies loss of atmosphere density model construction capability.
6. Mass Spectrometer Check	• Instrument is turned on and one scan of ambient environment is made. • Engineering operational parameters monitored (i.e., pump pressure, diode temperature, etc.)	• Engineering parameters verify integrity of electronics and pump. • Sample measurement provides comparison basis for flight measurements.

Figure 5.3–5

The in-flight checkout test duty cycle and turn-on/off reliability considerations for all equipment has been included in the overall ESP mission reliability calculations. It has been determined that the change in probability of equipment failure of the ESP is less than 1/2% as a result of these test operations. This change is insignificant relative to the enhancement of mission success as a result of the test information and command capability to update the mission sequence.

The ESP pre-separation checkout test period is between the Capsule Bus and Surface Laboratory checkout periods. The 6 minute ESP test interval occurs approximately 22 hours before Flight Capsule separation. This integrated test phasing and timeline are designed to allow adequate time for selecting any desirable mission updates. The cited corrective actions in Figures 5.3-4 and 5.3-5 are considered examples, because the test results of one system can affect the mission decisions for other systems. For example, the Surface Laboratory batteries may be used to power the ESP, with subsequent decrease in landed mission capacity. Moreover, test re-runs might be selected for any of the systems. The selected time margin from test initiation to planned separation is ample to decide and implement the optimum available (Capsule Bus, ESP, Surface Laboratory) mission plan.

5.3.4 ESP Monitor/Checkout Test Data/Command Interfaces - The data generation/gathering techniques and rates differ, as also does the command interface to perform status control for each of the ESP mission phases. Figure 5.3-6 presents the data/command interfaces of the ESP, Capsule Bus, and Spacecraft equipment for all mission phases.

Data Interface Description - From launch to the pre-separation checkout tests, the cruise commutator is the only ESP data source. Its own internal condition monitor data, ESP temperatures and battery status data are hardlined to the Spacecraft, which transmits the data to Earth.

The Capsule Bus Test Programmer automatically initiates the ESP checkout routine after completing its routine for the Capsule Bus checkout. A detailed description of this test programmer is given in Volume II, Part C, Section 8.3. The Capsule Bus Test Programmer, Sequencer and Timer, and Selector-Driver Unit design approach evolved primarily from CB checkout design considerations. It has been decided to use this same equipment combination to checkout the ESP.

The test programmer outputs result in accomplishing the tests of Figures 5.3-4 and 5.3-5 according to the 6 minute schedule discussed in the last subsection. The test programmer first commands the ESP telemetry subsystem into the checkout mode; this selects all the checkout test parameter channels. The test programmer then

ESP IN-FLIGHT TEST DATA AND COMMAND INTERFACE DIAGRAM

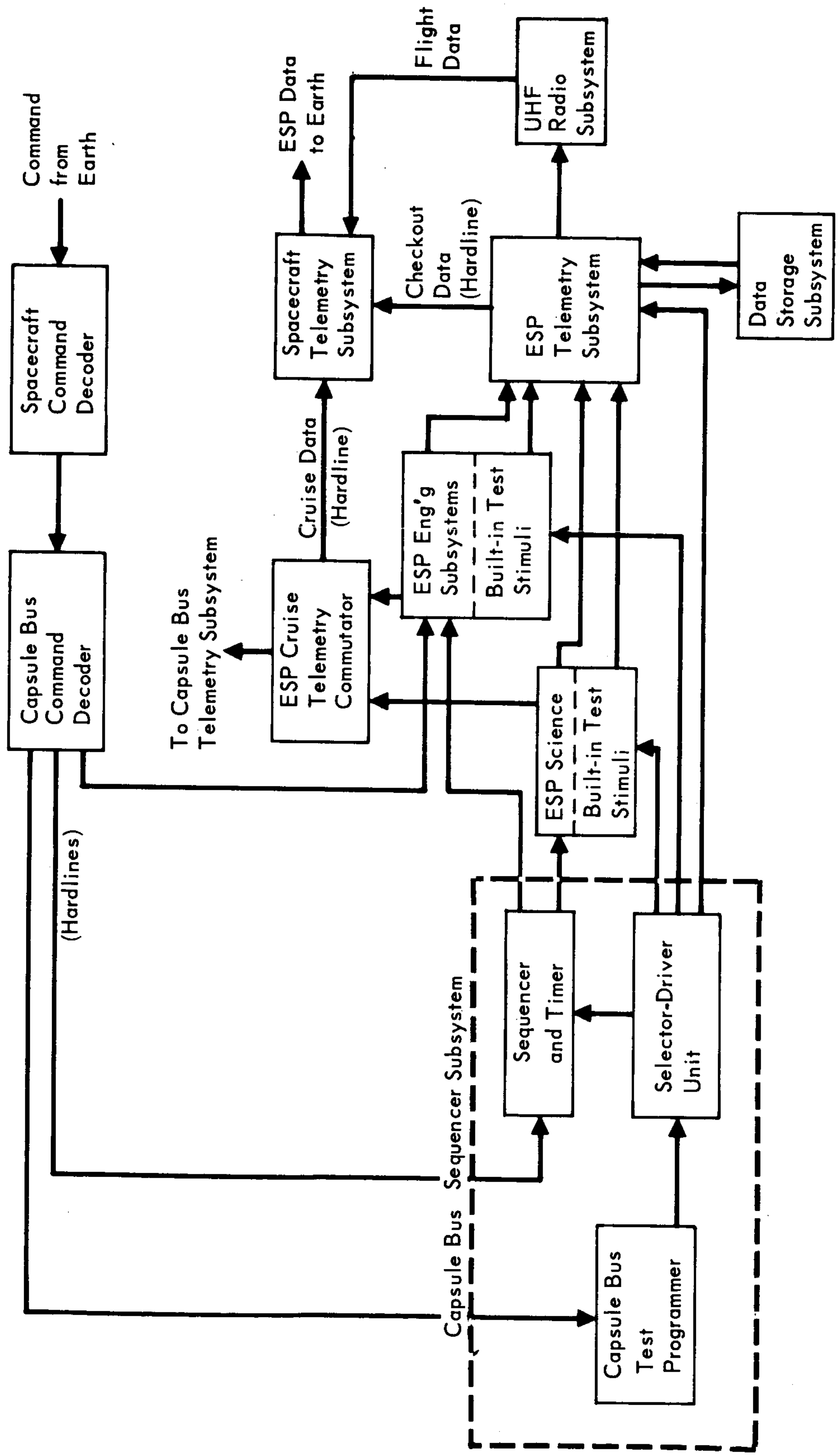


Figure 5.3–6

commands the science instruments and subsystems on and appropriately cycles the internal equipment test stimuli (i.e., activates entry TV camera illumination test pattern, cycles calibrated current source inputs to accelerometers, etc.) The test programmer outputs are time tagged commands which are decoded by the Selector Driver Unit and result in either Sequencer and Timer or Selector-Driver Unit output circuit closures. The command time delay between successive steps in any one test is based on estimated times for equipment stabilization; for example, the entry TV test picture take command is issued 4 minutes after camera turn on to allow vidicon tube temperature stabilization. All the checkout data is sent hardline from the ESP telemetry subsystem to the Spacecraft telemetry subsystem, which controls transmission to the Earth stations.

From separation to 800,000 feet altitude (de-orbit cruise), the ESP cruise telemetry commutator outputs are transmitted by the active CB telemetry subsystem. This technique eliminates the need for turn-on of the large capacity ESP telemetry subsystem during this 5 hour inactive period for the ESP.

The ESP equipment is turned on and warmed up for 5 minutes prior to science data transmission initiation at entry (800,000 feet altitude). The equipment monitor parameters are thereafter interleaved with the science data in the ESP Telemetry subsystem. The total real time telemetry outputs are relayed to the Spacecraft by the UHF radio subsystem, and the Spacecraft transmits the data to Earth stations. The ESP data is also delay transmitted using the Data Storage Subsystem so that ESP information is not lost through the entry communications blackout regime.

Command Interface Description - The ESP mission is designed to be automatic from Earth launch to Mars landing. However, a back-up command change capability is designed to alter its status and/or modify its mission.

The ESP mission modification command capabilities are to 1) change its time of turn on for operation at 800,000 feet and 2) change the nominal 30 second delay between peak dynamic pressure and low Mach number sensors turn-on. These changes would result primarily from any command changes in the pre-programmed Capsule Bus de-orbit profile, rather than ESP considerations.

The back-up commands for ESP equipment status/mode control are limited to the electrical power, thermal control, and telemetry subsystems. The battery can be commanded both on or off charge. The electrical heaters can be commanded on or off. The cruise commutator and telemetry subsystem can be placed in each of their respective modes. No back-up command capability exists to exercise the science instruments, except that available by exercising the test programmer.

The test programmer is capable of complete update. This allows test routine time changes between successive test steps and the repeat of any single test. This capability is required to allow equipment stabilization during checkout tests, if the first test data shows any pre-programmed estimated times are inadequate. The test programmer can also be employed to increase equipment temperature during cruise, if so desired, by turning on equipment with subsequent power dissipation and warming. If need be, an early checkout test routine can be initiated via command to the test programmer (hardline from Spacecraft).

There is no ESP command capability after Flight Capsule separation from the Spacecraft.

SECTION 6

IMPLEMENTATION

Successful development of the VOYAGER Flight Capsule requires a plan of action that assures a mission-quality vehicle. The Implementation Plan complements the technical approach in the previous sections of this volume, reflects consideration of the mission objectives, requirements, and constraints, and integrates twenty-one element plans into one plan of attack.

Basic assumptions and ground rules applies to the development of this plan are:

a. Provide four identical Flight Capsules for the 1973 opportunity. This includes two vehicles for the flight mission, and two units as backup spares to assure meeting the fixed launch period successfully.

b. The Capsule Bus System Contractor integrates the Surface Laboratory and the Entry Science Package to produce a Flight Capsule. This includes test articles, flight hardware, OSE, and personnel to support subsystems and systems integrated acceptance activities.

c. A launch period from 1 August to 6 September 1973 is utilized for the preferred design. However, if a landing near the evening terminator were used, the launch period is from 16 July to 6 September 1973. To retain a time contingency, the 16 July date is used for all planning and schedules.

d. Phase C starts 1 June 1968 and ends 29 February 1969. Phase D starts 1 March 1969 and continues to VOYAGER Program objective accomplishment.

SCOPE - The Implementation Plan, presented in Volume VI, provides a definitive description of the planned effort, the facilities, and the management controls that are necessary to produce a mission-ready Flight Capsule. It specifically establishes the time-phased sequence of events necessary for the design, development, fabrication, assembly, and test of the Capsule.

The hardware implementation aspects of the Entry Science Package are discussed in Section 3, Part B of Volume VI. A summary of that discussion is presented in subsequent paragraphs of this section. Individual implementation plans are presented, in summary, for twenty-one project functional elements. These element plans are organized into three basic project functions of Product Development, Product Support, and Management Control as illustrated in Figure 6-1. Each of the plans, as discussed in Volume VI, define the tasks, events, and activities necessary for Flight Capsule development within their respective function.

ELEMENT PLANS OF THE VOYAGER FLIGHT CAPSULE IMPLEMENTATION PLAN

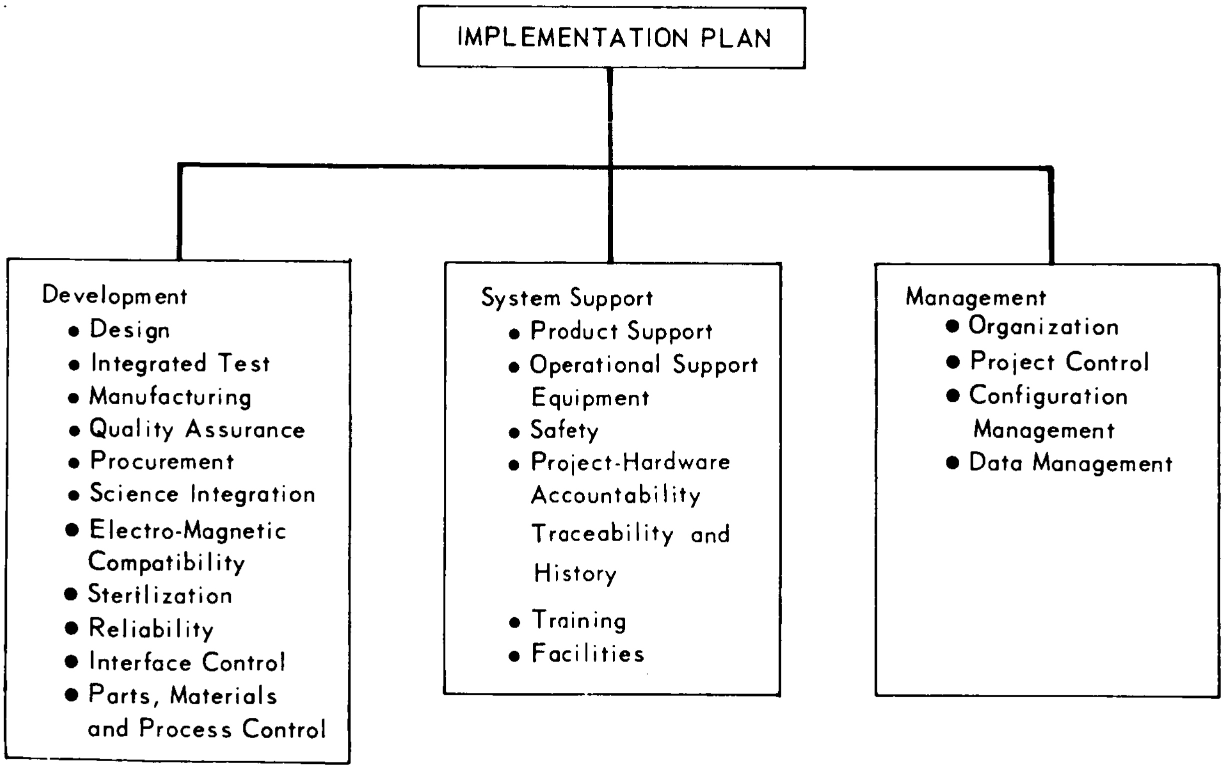


Figure 6-1

MAJOR CONSTRAINTS - Flight Capsule stufies have identified four major constraints that affect the implementation planning.

- o The inflexible launch period
- o Planetary quarantine requirements
- o Science-experiment integration
- o Interface multiplicity

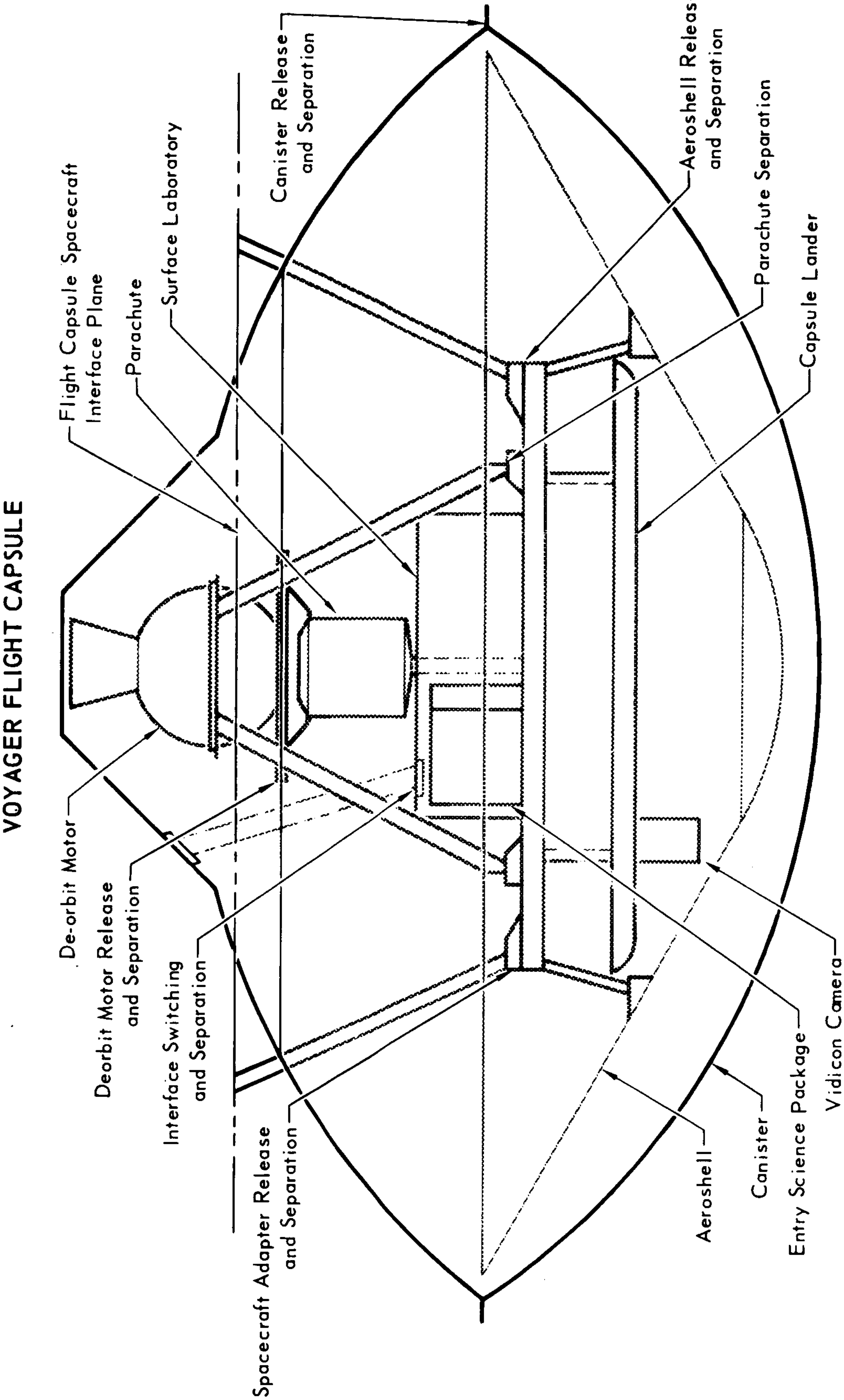
Launch Period - The inflexible launch period demands that precise schedules be established and controls exercised to insure that the Flight Capsule is flightworthy before the first day of the launch opportunity. The master Flight Capsule schedule must have sufficient flexibility for contingencies, which past experience has taught us to expect, and must be well coordinated with the schedule of other major systems.

Planetary Quarantine - The planetary quarantine requirements increase the time and cost required for total system development. The impact of the sterilization requirement is initially felt at the part/piece level, since part selection must be more stringent than in previous programs. At subsystem, system, and final assembly levels of fabrication and test, the microbiological monitoring, the decontamination, and the cleanliness control that are required increase the complexity of techniques, procedures, OSE, and facilities. Extensive parts identification and control are required so that parts traceability, microbiological loading, and reliability data can be provided on rapid recall.

Science Experiment Integration - The restrictive launch period and the planetary quarantine constraints require that experiment integration be accomplished with particular care, and in conjunction with the development of other subsystems. Prototype, qualification, and flight acceptance hardware is required at the appropriate time in the proper configuration, in accordance with the Integrated Test Plan.

Interfaces - Entry Science Package interface coordination with other systems requires a constant information flow. This means providing software and hardware intersystem interface control, with assistance, on time, in depth, as required.

6.1 BASIC SUBSYSTEMS AND MODULES - The preferred VOYAGER Flight Capsule concept configuration is composed of twenty basic subsystems assembled into seven major modules. The basic concept configuration is illustrated in Figure 6-2. The Entry Science Package basic subsystems are illustrated in Figure 6-3. Of the twenty basic subsystems in the Flight Capsule, twelve are germane to the Entry Science Package. These twelve subsystems are assembled into one major module (ESP principal unit), an accelerometer unit, a stagnation point sensor and inlet unit, and a camera



VOYAGER FLIGHT CAPSULE
Flight Capsule Spacecraft Interface Plane
Parachute
Surface Laboratory
Canister Release and Separation
Aeroshell Release and Separation
Parachute Separation
Capsule Lander
De-orbit Motor
Deorbit Motor Release and Separation
Interface Switching and Separation
Spacecraft Adapter Release and Separation
Aeroshell
Canister
Entry Science Package
Vidicon Camera

Figure 6–2

ENTRY SCIENCE PACKAGE SUBSYSTEMS

- Entry Package Science Subsystem
  - Descent TV
    - Camera (2)
    - Electronics
    - Shutter, Filter, Solenoid
  - Accelerometer – 3 Axis
  - Press. Transducer
  - Temp. Probe
  - Mass Spectrometer
- Power Subsystem
  - Battery
  - Battery Charger
  - Power Switching & Logic
- Radio Subsystem
  - High Rate Transmitters
    - Modulator
    - Power Amplifier
    - Spacecraft Mounted Receivers
- Antenna Subsystem
  - Antenna
  - Hybrids, etc.
  - Checkout Components
  - Spacecraft Mounted Antenna
- Telemetry Subsystem
  - Instrumentation
    - Transducers
    - Signal Conditioning
    - Power Supply
  - Equipment
    - Science and Monitoring Data Commutator
    - Cruise Commutator
    - Programmer
    - Multiplexer Encoder
    - Data Combiner
- Data Storage Subsystem
  - Data Storage Buffer
- Structural Mech. Subsystem
  - Structure
  - Mechanisms
- Cabling
- Thermal Control Subsystem
  - Heaters
  - Thermostats
  - Insulation
  - Coatings

SPACECRAFT MOUNTED EQUIPMENT

- Radio Subsystem (Ref)
  - High Rate Equipment
    - Receiver
    - Synchronizer
- Antenna Subsystem (Ref)
  - Antenna(s)
  - Hybrids, Diplexer, etc.
- Command Subsystem (Thru Capsule Bus Equipment)

Figure 6-3

equipment unit, as illustrated in Figure 6-4.

6.2 SCHEDULES AND ANALYSIS - A summary Master Schedule for the Entry Science Package is included as Figure 6-5. This section highlights the key events planned in this schedule, the time critical subsystems identified through our schedule analysis and the provisions for contingency included in our planning.

6.2.1 Key Schedule Dates - During the first two months of Phase C, a detailed review of subsystem development requirements is conducted to augment our preliminary analysis and define the time critical subsystems. This early identification permits adequate time during Phase C to begin breadboard and development of these critical subsystems. Detailed subsystem design is scheduled to complete two months prior to the Part I CEI submittal date. This is to allow ample time for incorporation of design data into the specification prior to the Preliminary Design Review.

During Phase D the ciritical design review is scheduled to begin 17 months after go-ahead and 11 months prior to the start of the Entry Science Package flight vehicle #1 structural fabrication. This scheduling is late enough to permit design maturity, yet early enough to effectively implement changes in the development and manufacturing operations. The first production test article is available in October 1970. Early availability of test vehicles will allow a progressive series of component, subsystem, and system testing, permitting advanced identification of system development problems. All qualification testing is scheduled to complete prior to delivery of the first flight vehicle. This allows any problems discovered in qualification testing to be investigated and appropriate changes incorporated into the first flight vehicle. Operational support equipment for the factory has been scheduled to be validated and ready for operation approximately three months prior to its first usage.

Delivery of all production test and flight vehicles has been scheduled to be delivered to the integrating Contractor six weeks prior to their installation date into the Flight Capsule. This time period allows the integrating Contractor to perform the normal inspection functions and integration tasks prior to installing the Entry Science Package into the Flight Capsule.

6.2.2 Critical Subsystems - The milestones identified on the Master Schedule were analyzed through the development of numerous plans and schedules and integrated through the processing and review of over 5000 PERT network activities. The following subsystems were identified as time critical:

ENTRY SCIENCE PACKAGE INSTALLATION

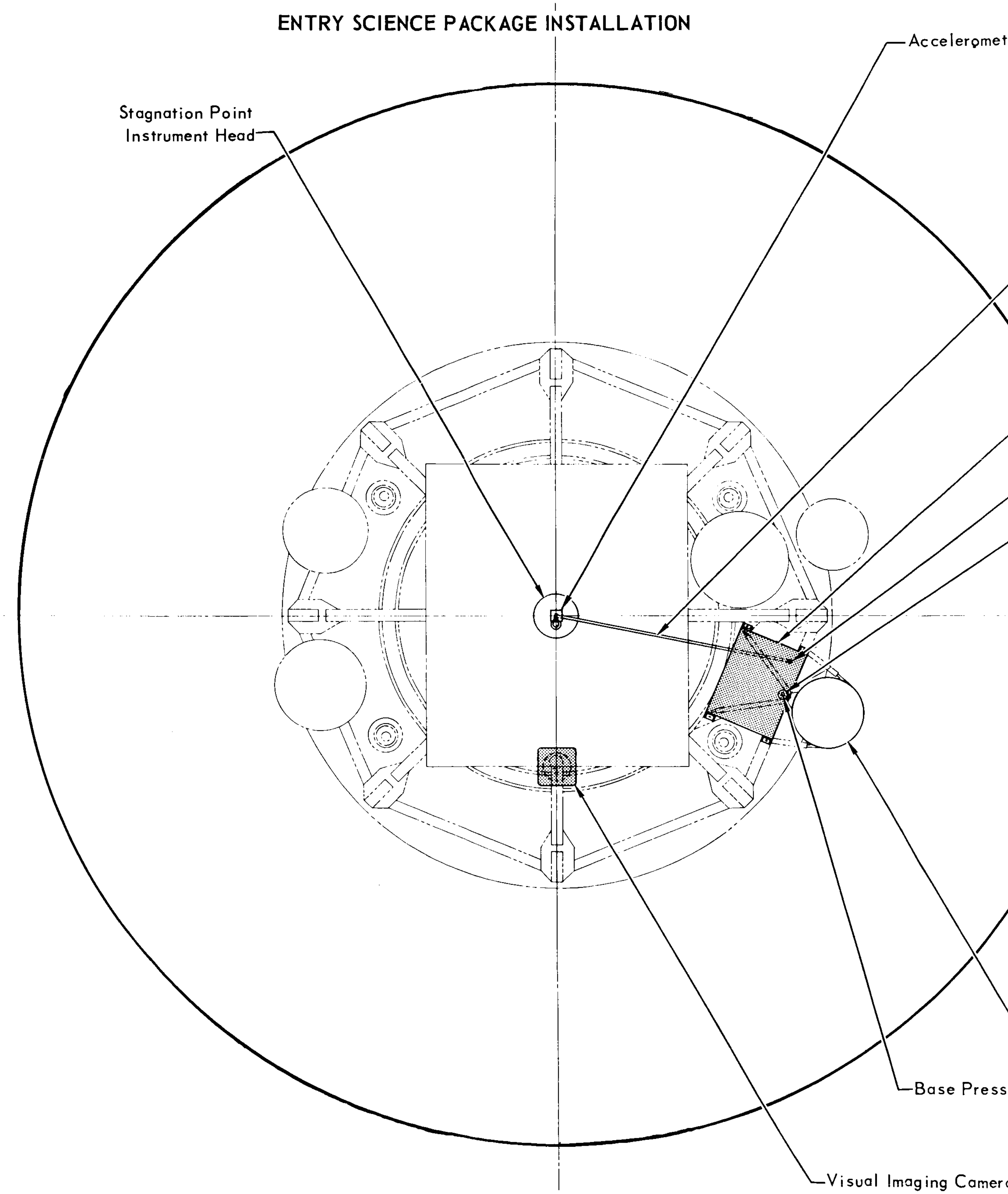
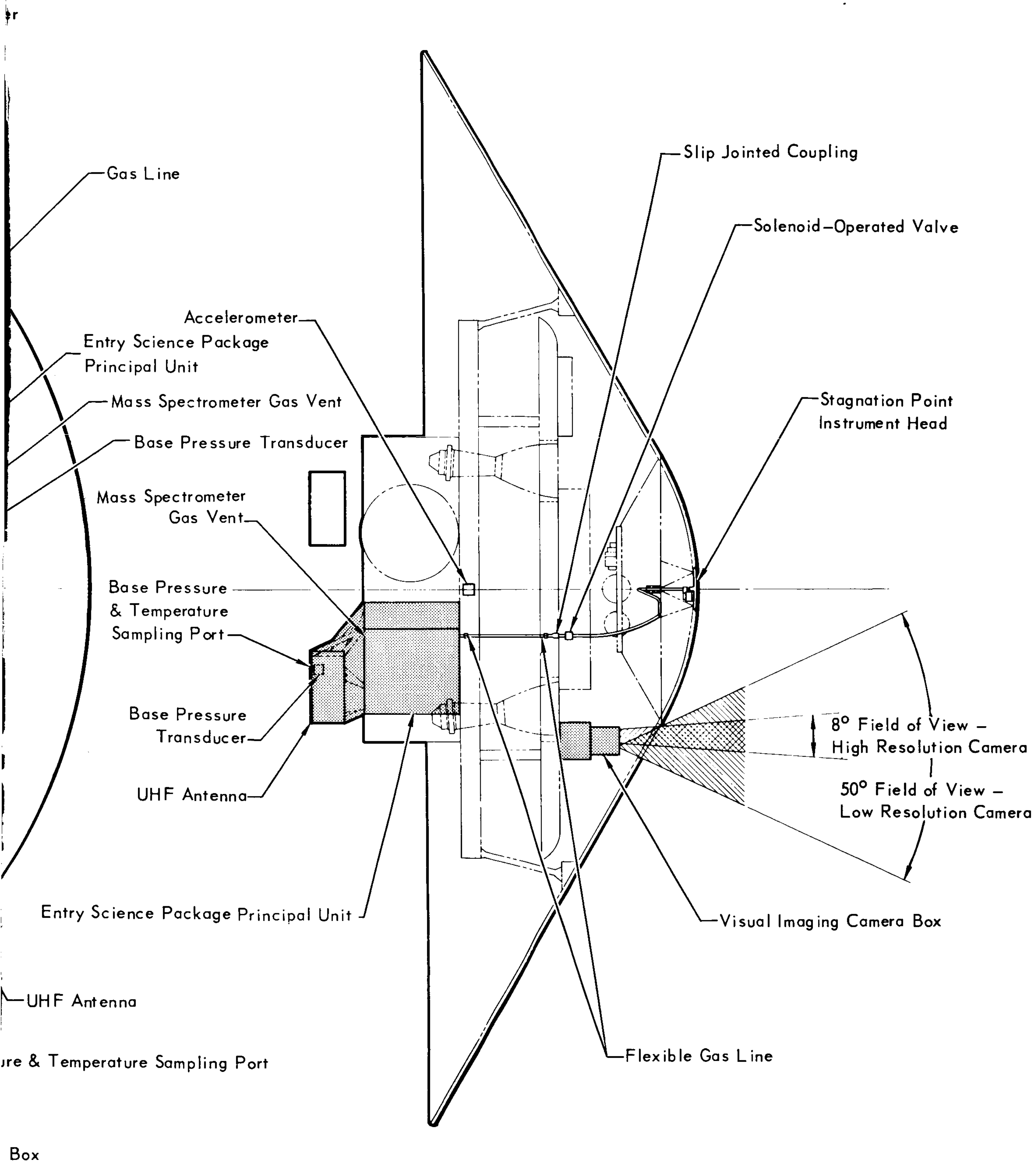


Figure 6-4

6-7-1


Gas Line
Accelerometer
Entry Science Package
Principal Unit
Mass Spectrometer Gas Vent
Base Pressure Transducer
Mass Spectrometer
Gas Vent
Base Pressure
& Temperature
Sampling Port
Base Pressure
Transducer
UHF Antenna
Entry Science Package Principal Unit
UHF Antenna
ıre & Temperature Sampling Port
Box
Slip Jointed Coupling
Solenoid-Operated Valve
Stagnation Point
Instrument Head
8° Field of View –
High Resolution Camera
50° Field of View –
Low Resolution Camera
Visual Imaging Camera Box
Flexible Gas Line

VOYAGER ENTRY SCIENCE PACKAGE – SUMMARY SCHEDULE

	ACTIVITY
1	PHASE C
2	
3	Preliminary System Design Completed
4	
5	System Layouts Completed
6	
7	Detailed Subsystem Design Complete
8	
9	Part I CEI Specs Completed and Submitted
10	
11	Preliminary Design Review
12	
13	Final Definition of Critical Subsystems
14	
15	Critical Subsystem Breadboard Fabrication
16	
17	Critical Subsystems Breadboard Testing
18	
19	PHASE D
20	
21	Engineering
22	90% Design Release
23	Critical Design Review
24	Part II CEI Specs Submitted
25	First Article Configuration Inspection
26	
27	Subsystem Development
28	Purchase Orders (PO) Placed
29	Subsystems Development Testing
30	Subsystems Qualification Testing
31	Flight Vehicle No. 1 Subsystems Available
32	
33	Manufacturing
34	Begin Detailed Parts Manufacture
35	Begin Test Article Fabrication
36	Flight Vehicle No. 1 Structural Assembly
37	Flight Vehicle No. 1 Subsystems & Equipment Instl
38	Flight Vehicle No. 1 Systems Tests
39	Flight Vehicle Deliveries to Integ Contr
40	
41	Qualification Testing
42	Perform Static, Dyanmic & Thermal Testing
43	
44	Operational Support Equipment (OSE)
45	Factory – Installation & Validation
46	
	MONTHS AFTER GO-AHEAD
	CALENDAR

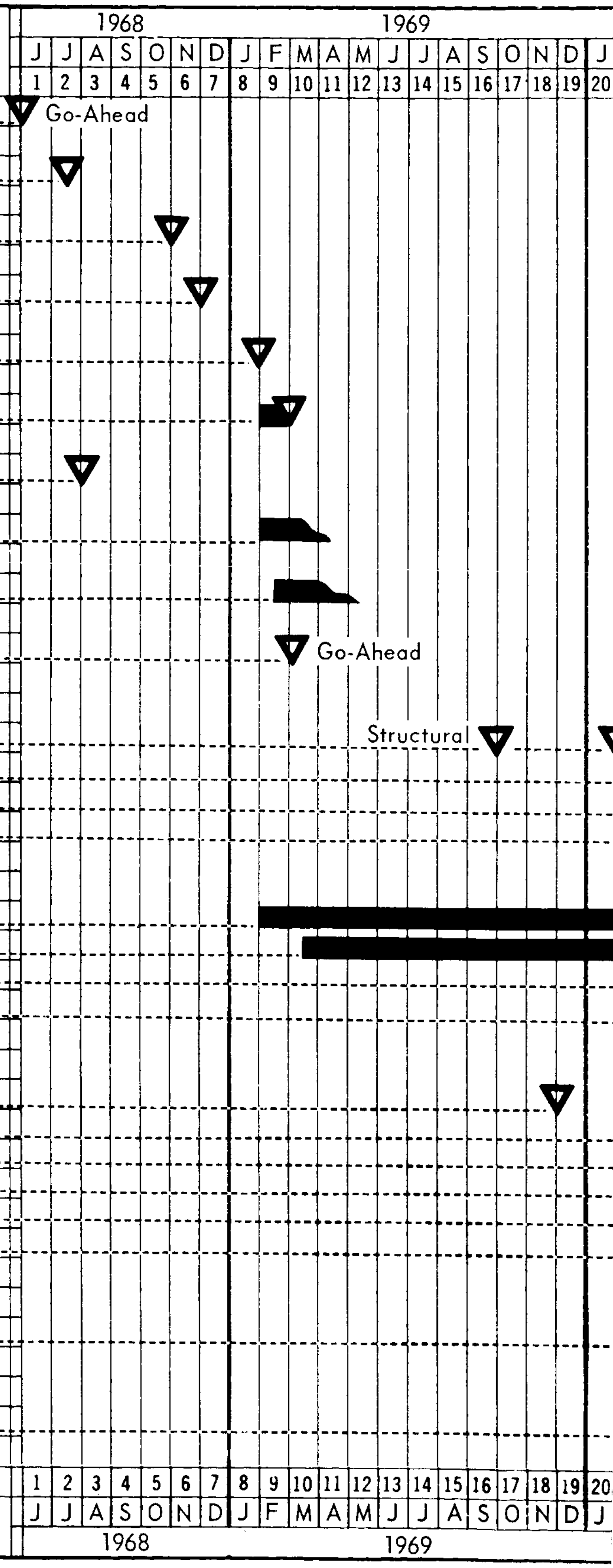
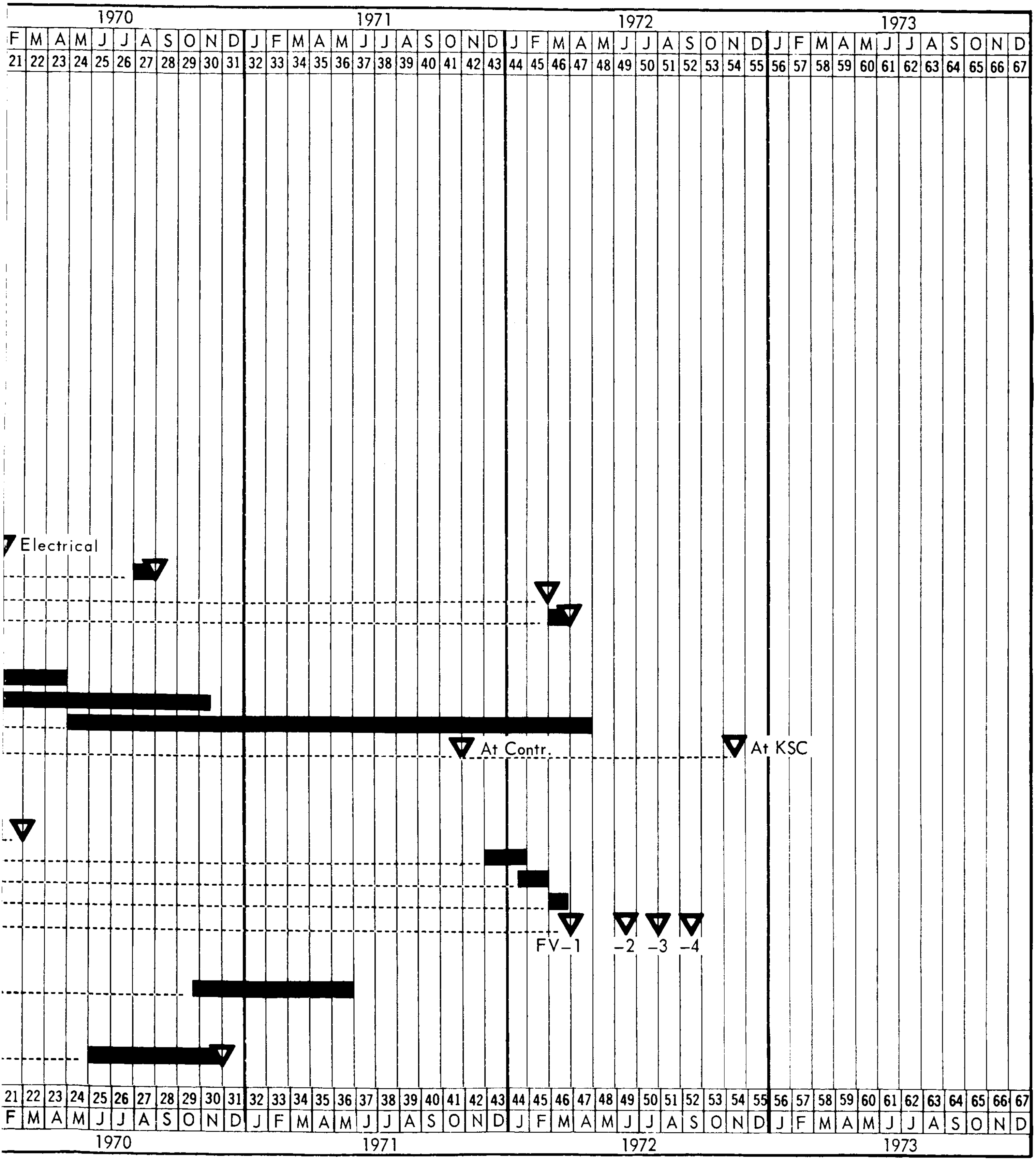


Figure 6–5


1970
1971
1972
1973
F M A M J J A S O N D J F M A M J J A S O N D J F M A M J J A S O N D J F M A M J J A S O N D
21 22 23 24 25 26 27 28 29 30 31 32 33 34 35 36 37 38 39 40 41 42 43 44 45 46 47 48 49 50 51 52 53 54 55 56 57 58 59 60 61 62 63 64 65 66 67
Electrical
At Contr.
At KSC
FV–1
–2
–3
–4

Critical Subsystem	Critical Item	Approximate Weeks Criticality
Telemetry	Programmer development and testing	
Data Storage	Development and testing of memory core units	
Radio	Development and testing of 400MH transmitter and bit synchronizer	

The critical subsystems noted above were those remaining after all feasible Phase D replanning efforts had been completed. It was, therefore, determined that the components within these subsystems must begin development during Phase C. It is recognized that the criticality for these subsystems is small and requires further investigation during the early part of Phase C to substantiate this advanced effort.

6.2.3 Contingencies - The Master Schedule, and the more detailed schedules supporting it, include planned provisions for recovery in the event of unexpected delays in the program. These provisions can be eliminated at increased risk and/or cost.

Subsystem Development - The subsystem development flow consists of constraints of breadboard performance and qualification testing on the release of design for prototype and production hardware. For example, the design release for manufacturing prototype hardware is constrained by a completion of 50% of the performance and sterlization testing. A relaxation of this constraint to only 25% completion of this testing would allow an earlier buildup and delivery of test and flight articles for many of the subsystems.

System Assembly - Assembly operations are planned for a two-shift, 40 hour work week. The capability, therefore, exists for third-shift and extended work week operations. The fabrication of the production vehicles reflects no learning. Should an 85% learning curve be accomplished the availability of the first flight vehicle could be approximately 6 weeks earlier.

System Test - Structural, dynamic and thermal testing has been scheduled to allow approximately 10 months for re-design, fabrication and re-testing should any major failures occur during testing, and still permit appropriate changes to be incorporated in the first flight vehicle prior to its delivery to the integrating Contractor. It should be noted that the ESP delivery would move downstream thirteen weeks if it were an integral portion of the Capsule Bus rather than being supplied as a separate item. This is because the contingency lead time and unit pre-installation checkout time would not be required.

6.3 Manufacturing Schedule and Flow Plan - This section contains the manufacturing schedule and flow plan for the Entry Science Package (ESP). It shows the scheduling of functions required to produce the ESP, starting with engineering design and proceeding through the production of all test and flight articles. It illustrates the manufacturing flow of a typical flight article through the various stages of production and acceptance testing, including the point at which the ESP is ready for delivery for integration into the Capsule Bus.

6.3.1 Manufacturing Schedule - The manufacturing schedule for the Entry Science Package is shown on Figure 6-6. Development of this schedule was based on the requirements of the ESP and the integrating Contractor. These requirements include providing test articles, tooling, qualification units, and flight vehicles at the appropriate time for ESP needs and also the needs of the Capsule Bus Contractor.

6.3.2 Flow Plan - Figure 6-7 shows the flow plan for a typical flight article of the Entry Science Package. A pictorial representation of this plan is displayed in Figure 6-8.

6.4 Integrated Test Plan - The purpose of this comprehensive test plan, including Feasibility, Development, Qualification, Pre-Delivery Acceptance and Flight Acceptance Test Phases, is to demonstrate the ability of the Flight Capsule, including the ESP, to perform to the requirements of the mission. Testing under conditions representative of those to be experienced during the mission provides a realistic evaluation of the performance of the equipment. The VOYAGER Integrated Test Plan is an evolution of the plans employed on the previous Mercury, ASSET, and Gemini spacecraft programs. The arrangement of the test phases and the selection of the test models places emphasis on thorough evaluation tests with attention given to the exclusion of repetitive tests which produce little design improvement or engineering confidence. The test phases are interrelated in that the progress of each phase is a constraint on the subsequent phases. Figure 6-9 presents the time phasing of the requires series of tests. Descriptions of each test phase are presented in the following paragraphs.

6.4.1 Feasibility Tests - The purpose of this current test phase is to evaluate materials and design approaches. It has been in progress for the past two years and is to be completed early in Phase C. The results of the tests to date have been used as the foundation for the selection of many aspects of the preferred concept design.

The major portion of the feasibility testing effort has been devoted to the following:

a. Martian surface environment simulation
b. Microbiological research and related investigation of sterilization problems and techniques

ENTRY SCIENCE PACKAGE MANUFACTURING SCHEDULE

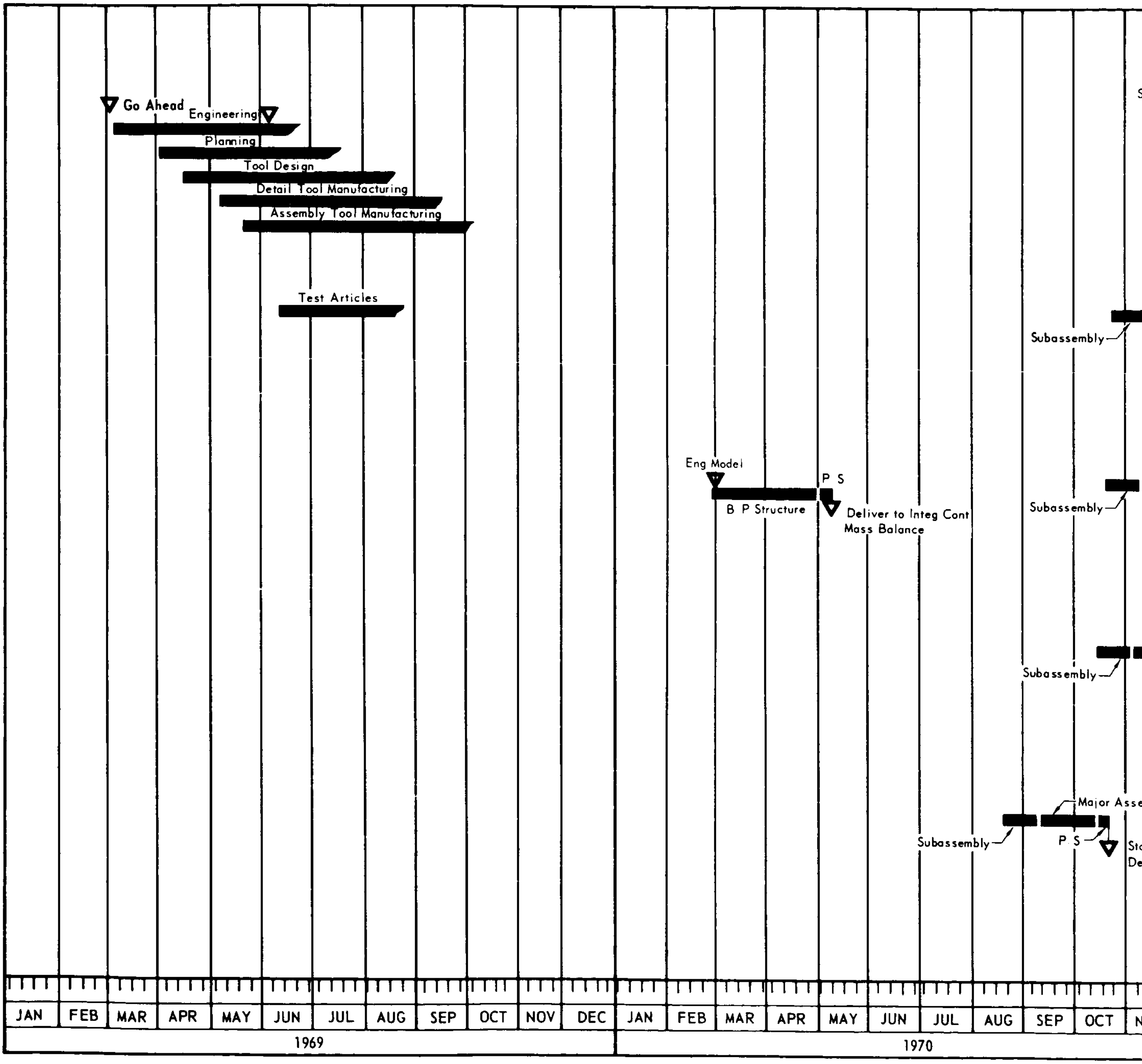
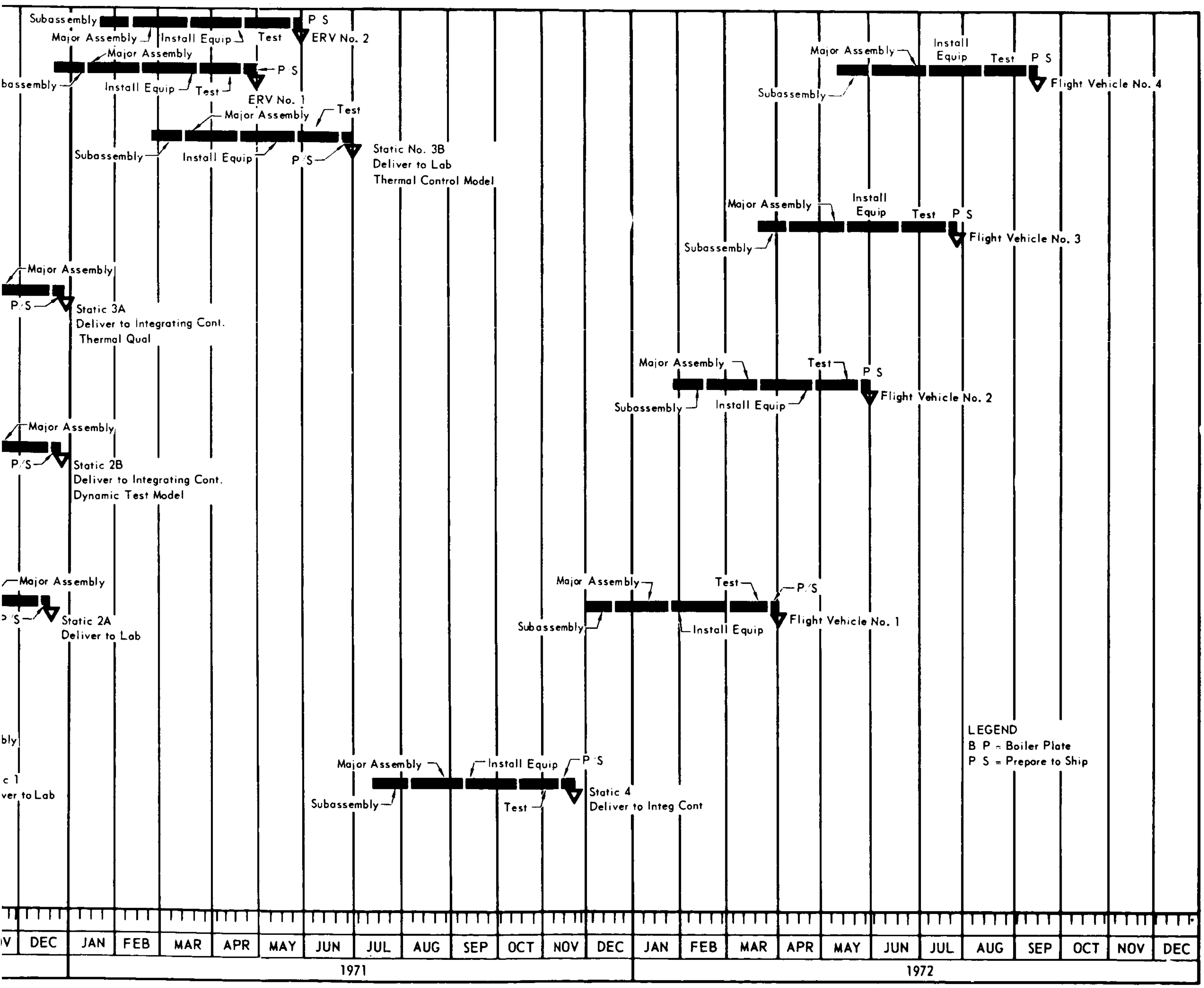


Figure 6-6

6-11 -1


Subassembly
Major Assembly
Install Equip
Test
P S
ERV No. 2
ERV No. 1
Static No. 3B
Deliver to Lab
Thermal Control Model
Static 3A
Deliver to Integrating Cont.
Thermal Qual
Static 2B
Deliver to Integrating Cont.
Dynamic Test Model
Static 2A
Deliver to Lab
Static 4
Deliver to Integ Cont
Flight Vehicle No. 1
Flight Vehicle No. 2
Flight Vehicle No. 3
Flight Vehicle No. 4
LEGEND
B P = Boiler Plate
P S = Prepare to Ship
DEC JAN FEB MAR APR MAY JUN JUL AUG SEP OCT NOV DEC JAN FEB MAR APR MAY JUN JUL AUG SEP OCT NOV DEC
1971
1972

ENTRY SCIENCE PACKAGE

MANUFACTURING FLOW

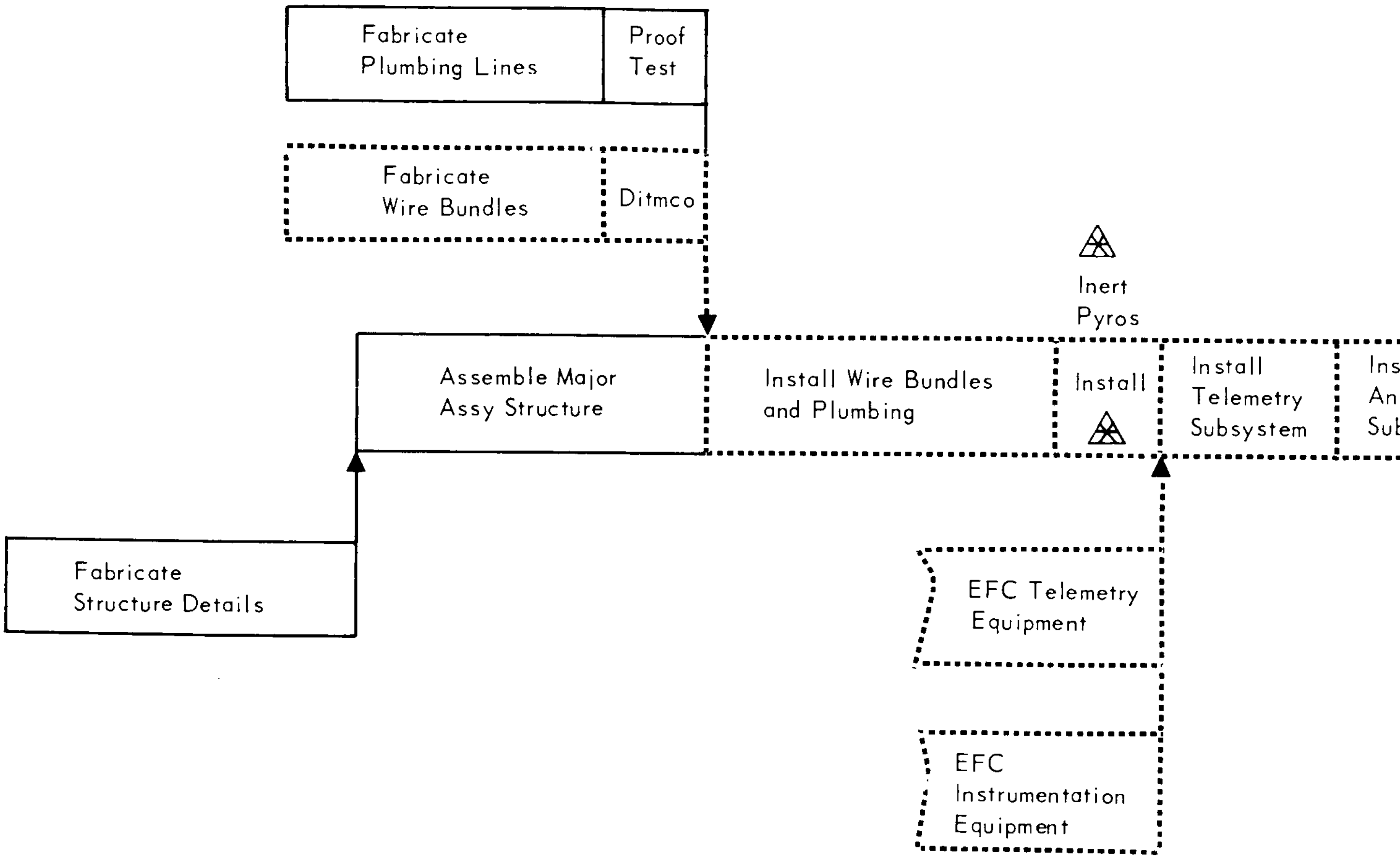
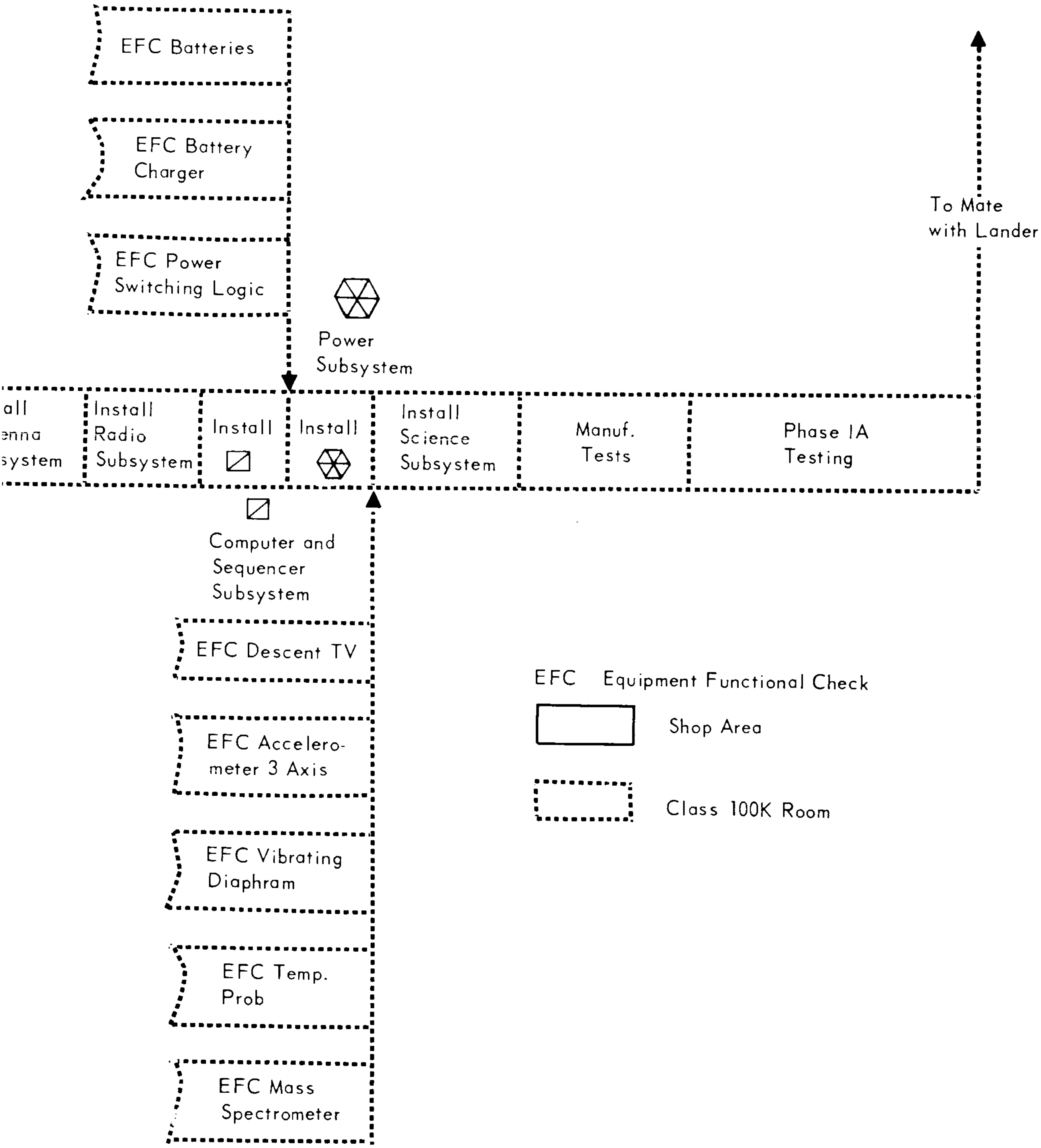


Figure 6–7


EFC Batteries
EFC Battery Charger
EFC Power Switching Logic
Power Subsystem
To Mate with Lander
all
enna
system
Install Radio Subsystem
Install
Install
Install Science Subsystem
Manuf. Tests
Phase IA Testing
Computer and Sequencer Subsystem
EFC Descent TV
EFC Accelero-meter 3 Axis
EFC Vibrating Diaphram
EFC Temp. Prob
EFC Mass Spectrometer
EFC Equipment Functional Check
Shop Area
Class 100K Room

ENTRY SCIENCE PACKAGE PRINCIPAL UNIT MANUFACTURING FLOW

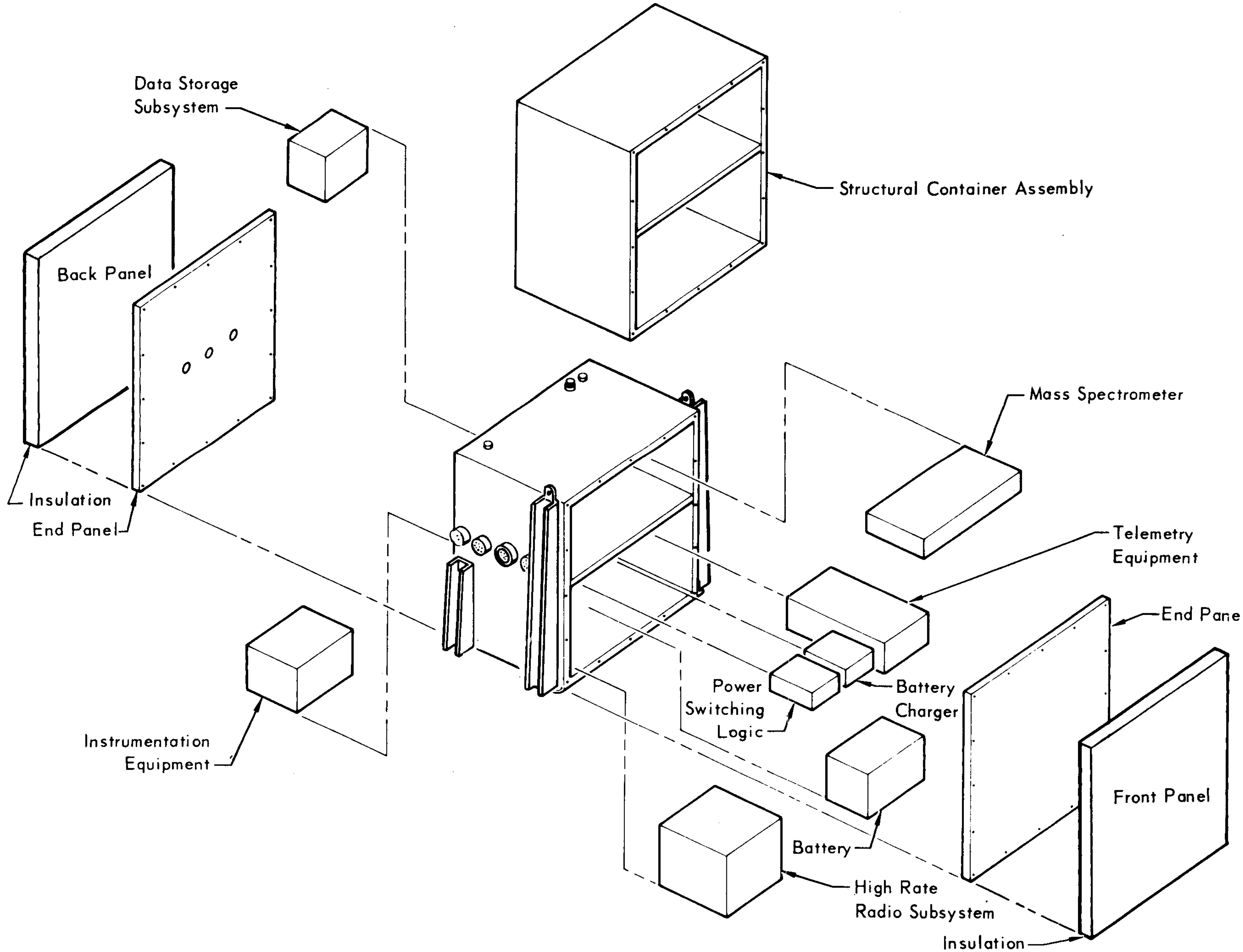


Figure 6-8

VOYAGER CAPSULE BUS INTEGRATED TEST PLAN – SUMMARY

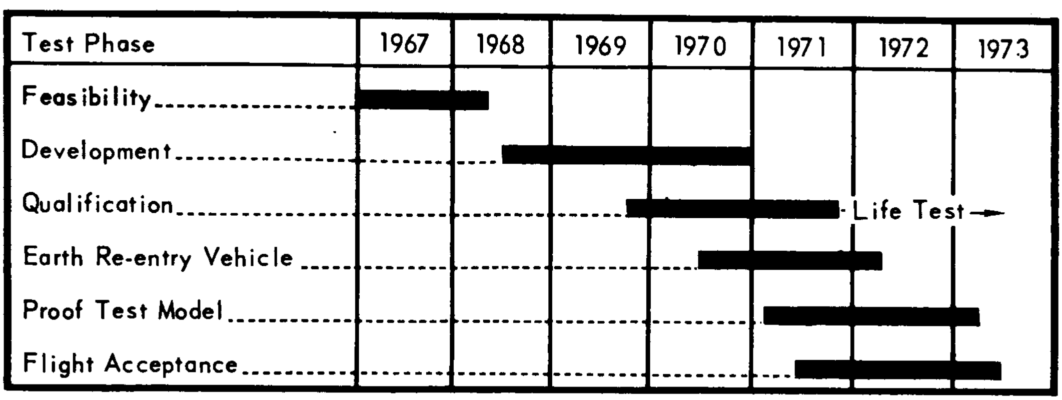


Figure 6–9

c. Soft and hard landing concepts evaluation

d. Structural design evaluation

e. Telecommunications entry and surface characteristics

f. Long term hard vacuum exposure effects

g. Real time versus reduced time test

Figure 6-10 shows the scope and duration of the feasibility test to date.

6.4.2 <u>Subsystem Tests - Development and Qualification</u> - The subsystem development tests are programmed to provide design information, analysis verification, and a demonstration of design adequacy early in the program. Subsystem qualification testing is thus conducted to demonstrate hardware adequacy with margins of performance that are significantly greater than those expected for ground and flight extremes. See Figure 6-11 for the time phasing of the tests. The various subsystem development and qualification tests employ hardware in breadboard, engineering prototype, and manufacturing prototype configurations. Figure 6-12 is a matrix, showing the application of the tests performed to the ESP.

6.4.3 <u>System Test - Development and Qualification</u> - Those tests requiring a relatively complete (including the ESP) representation of the 1973 flight configuration hardware procedures, support equipment and other elements needed to achieve the defined test objectives. The design verification tests described are similar to the Pre-Delivery Acceptance (PDA) or Flight Acceptance Tests (FAT) and to each other in flow and form, providing a build-up of training, experience, ability, and confidence in the pre-mission preparations. Figure 6-13 shows the timing and sequence of these tests. Test categories are:

a. Engineering Model Tests are performed to establish levels of acceptable integrated system and subsystem performance using components of engineering prototype configuration.

b. Thermal Control Subsystem Tests are performed to verify that the subsystem will properly function in all of the simulated mission environments. All other subsystems are installed using components of manufacturing prototype configurations.

c. System Life Tests demonstrate systems operations capability, compatibility, and endurance employing the hardware previously used for the thermal control tests. One of two programs, the simulation mission test, verifies design integrity as regards function and compatibility. The other, systems compatibility and endurance tests, evaluates the

MANUFACTURING TEST SCHEDULE

TEST APPLICIBILITY ESP	SLS	CBS	SECTION NUMBER	ACTIVITY
			1.1.1	MATERIALS EVALUATION
		X	1.1.1.1	Ablative Material Tests in Simulated Martian Entry Environments
		X	1.1.1.2	Effect of Sterilization and Vacuum Environments on Ablative Materials
		X	1.1.1.3	Ablative Material Weight Loss in a High Vacuum
	X	X	1.1.1.4	Effect of Time at Elevated Temperature on Energy Absorption Materials
	X	X	1.1.1.5	Evaluation of ORPEAM and Balsa-Cube Binder
	X	X	1.1.1.6	Lockalloy Mechanical Properties Test
	X	X	1.1.1.7	Effect of Sterilization on Resistance Welded Titanium
X	X	X	1.1.1.8	Long Term Material Exposure (ETO, Heat, Vacuum)
X	X	X	1.1.1.9	Accelerated Testing of Materials
		X	1.1.1.10	Heat Shield Materials Evaluation Program
			1.1.2	STRUCTURAL DESIGN EVALUATION
		X	1.1.2.1	Entry Cone Structural Panel Fabrication and Ablator Evaluation
		X	1.1.2.2	Static Test of Titanium Panels for Effect of Load on Ablator Band
		X	1.1.2.3	Static Test to Determine the Buckling Strength of Rings in 120° Conical Shapes
			1.1.3	LANDING SYSTEMS EVALUATION
	X	X	1.1.3.1	Sphere Penetration in Sand and Dust
		X	1.1.3.2	Erection Test of ¼ Scale Spherical Lander Model
		X	1.1.3.3	Deflection and Profile Tests on Flexible Inflated Torus Tank
		X	1.1.3.4	Drop Tests of ¼ Scale Torus Lander
		X	1.1.3.5	Overturning-Stability and Drop Tests of 1/10 Scale Legged Landers
		X	1.1.3.6	Determination of Stability of 1/10 Scale Uni-Disc Lander
		X	1.1.3.7	Development of Dynamic Impact Facility for Testing Energy Absorption Materials
		X	1.1.3.8	Static and Dynamic Energy Absorption

1965					1966												1967								
A	S	O	N	D	J	F	M	A	M	J	J	A	S	O	N	D	J	F	M	A	M	J	J	A	S
1	2	3	4	5	6	7	8	9	10	11	12	13	14	15	16	17	18	19	20	21	22	23	24	25	26

Figure 6–10

6-16-2

				Honeycomb, Aluminum Flexcore and Aluminum Crosscore Before and After Sterilization
			1.1.4	AERODYNAMICS
X		X	1.1.4.1	Entry Capsule Force and Moment Tests in Polysonic Wind Tunnel
X		X	1.1.4.2	Aerodynamic Decelerator Static Force and Moment Test in Trisonic Wind Tunnel
			1.1.5	TELECOMMUNICATION
X	X	X	1.1.5.1	Antenna Breakdown Tests in Simulated Martian Atmospheres
X	X		1.1.5.2	S-Band Antenna Radiation Pattern Tests
X		X	1.1.5.3	Flow Field Seeding for Entry Ionization Suppression
X		X	1.1.5.4	Bit Synchronization in a Multipath Environment
X	X	X	1.1.5.5	RF Systems Multipaction Tests
X		X	1.1.5.6	RF Transmissibility Test
X	X	X	1.1.5.7	UHF Antenna Systems Radiation Pattern Tests
			1.1.6	PROPULSION
		X	1.1.6.1	Feasibility of Sterilizing Liquid Propellant Systems
		X	1.1.6.2	Material Compatibility During Sterilization of Anhydrous Hydrazine (N_2H_4)
		X	1.1.6.3	Evaluation of Sterilization Effects on Mono-Propellant Engine Performance
			1.1.7	PYROTECHNICS
	X	X	1.1.7.1	Effect of Sterilization and Long Storage Life on Pyrotechnic Actuation Devices
			1.1.8	SURFACE ENVIRONMENT SIMULATION AND THERMAL CONTROL EVALUATION
	X		1.1.8.1	Development of a Martian Environmental Simulation Facility
	X		1.1.8.2	Dust Particle Behavior in a Simulated Martian Atmosphere
	X		1.1.8.3	Behavior and Characteristics of Simulated

				Martian Sand and Dust Storms
	X		1.1.8.4	Wind Blown Sand and Dust Tests
	X		1.1.8.5	Effect of Voyager Mission Requirements on Thermal Control Coatings
	X		1.1.8.6	ETO Effects on Thermal Control Coatings
	X	X	1.1.8.7	Heat Pipe Demonstration
	X	X	1.1.8.8	Heat Pipe Control Valve Test
X	X	X	1.1.8.9	Measurement of Wind Velocity at Low Pressure
	X		1.1.8.10	Investigation of Martian Surface Phenomena
			1.1.9	RELIABILITY
X	X	X	1.1.9.1	Environmental Effects on Electronic Parts
			1.1.10	BIO-CONTAMINATION CONTROL TECHNIQUES EVALUATION
X	X	X	1.1.10.1	Microbiological Research
X	X	X	1.1.10.2	Sterile Assembly
X	X	X	1.1.10.3	Class 100 Facility Operation
			1.1.11	EXPERIMENTAL INVESTIGATION
X	X	X	1.1.11.1	Large Amplitude Con ing in a Single Degree-of-Freedom Gyro.
X	X		1.1.11.2	Efficiency of Molecular Separators for Interfacing a Gas Chromatograph With a Mass Spectrometer
X	X		1.1.11.3	Pryolysis – Gas Chromatograph for Chemical Analysis
X	X		1.1.11.4	Digitization of Time-of-Flight Mass Spectra
X	X		1.1.11.5	Advantages of a Multi-Diameter Separation Column in Gas Chromatographic Analysis of Organics

6-16-3

TIME PHASING OF TESTS

ACTIVITY

PHASE C
PHASE D
1.0 Material & Processes
- Material Selection Tests
- Fabrication & Assembly Processes

2.0 Thermal Control Subsystem
- Insulation Tests
- Heater and Thermostat Tests
- Thermal Control Coatings Test
- Thermal Control Simulator Test (Included in Capsule Lander Thermal Control Simulator Test)

3.0 Structural/Mechanical Subsystem
- Qualification Testing
  - Structural Tests (Static No. 1)
  - Dynamic Model Tests (Static No. 2)

4.0 Typical Development Tests for Electronic Subsystems
- Breadboard Testing ⚠1
  - Performance Check
  - Performance (Temperature & Power Extremes)
  - Trial Sterilization Cycles
  - Performance Margin Tests
  - EMI Search Test
  - Selected Mission Environments
- Engineering Prototype Testing ⚠1
  - Performance – Acceptance
  - Performance – Sterilization Cycles
  - Performance – Margin Test
  - Performance – Selected Mission Environments
  - EMI Margin Test
- Integrated Electronic Subsystems Bench Testing ⚠1
  - Power Up Testing & Systems Operation
  - Interface Verification Tests
  - EMI Margin Test
  - Performance Margin Tests
  - In-flight Checkout
- Qualification Testing ⚠1
  - Manufacturing Prototype Hardware
    - Performance – Acceptance
    - Performance – Sterilization Cycles
    - Performance – Mission Environments
    - Performance – Margin Test
    - Performance – Mission Life Tests

5.0 Typical Development Tests for Science Subsystems
- Breadboard Testing ⚠1
  - Performance Check
  - Performance (Temperature & Power Extremes)
  - Trial Sterilization Cycles
  - Performance Margin Tests
  - EMI Search Test
  - Selected Mission Environments

⚠1 The following tests are including in the testing

ACTIVITY

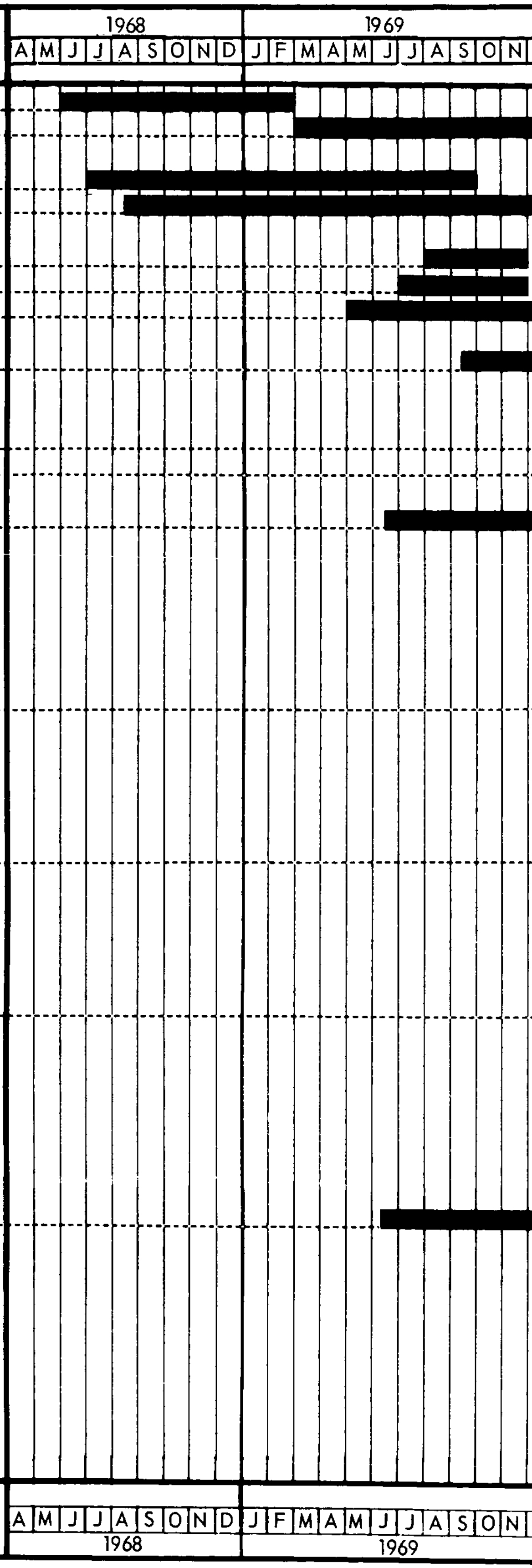
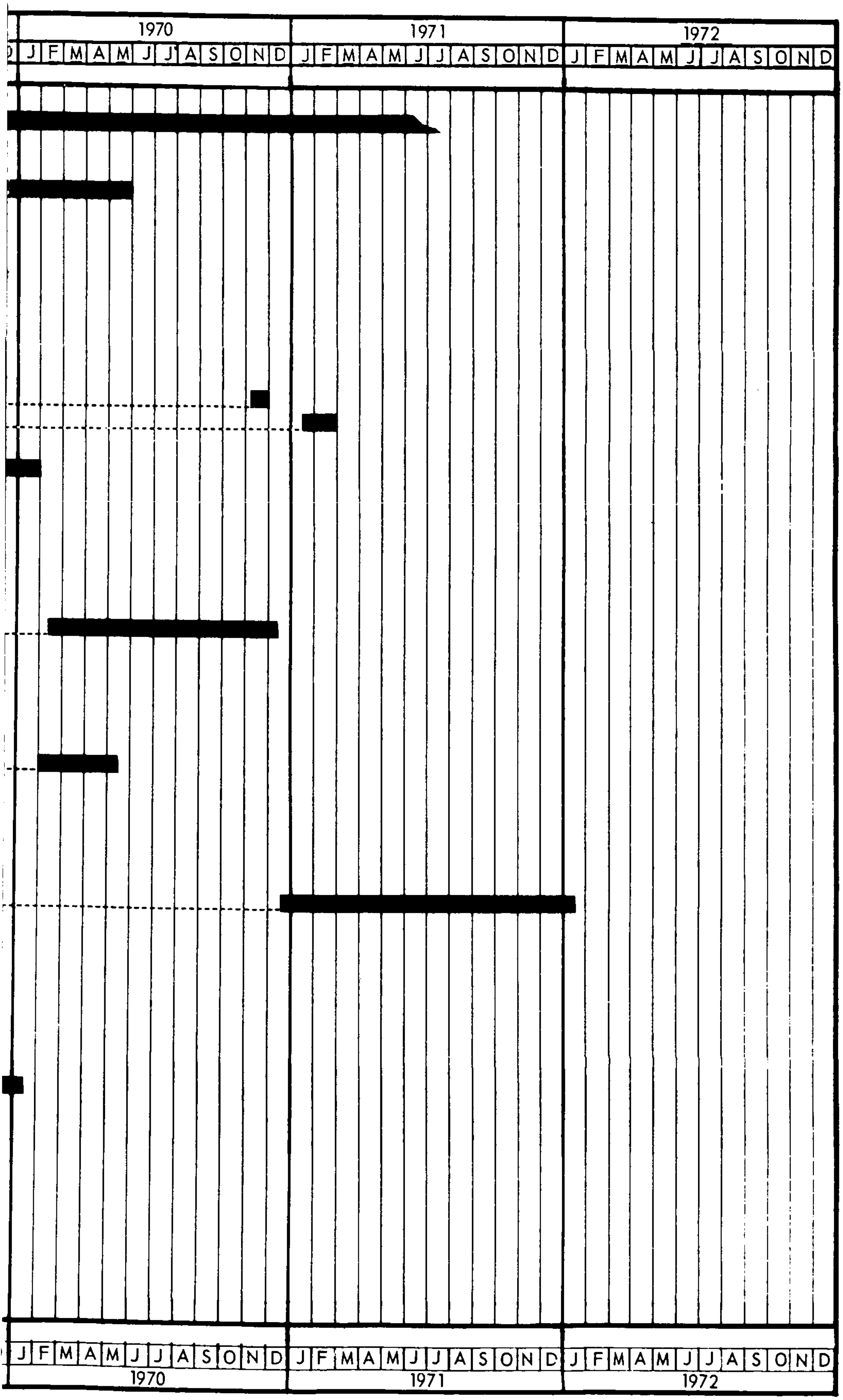


Figure 6–11



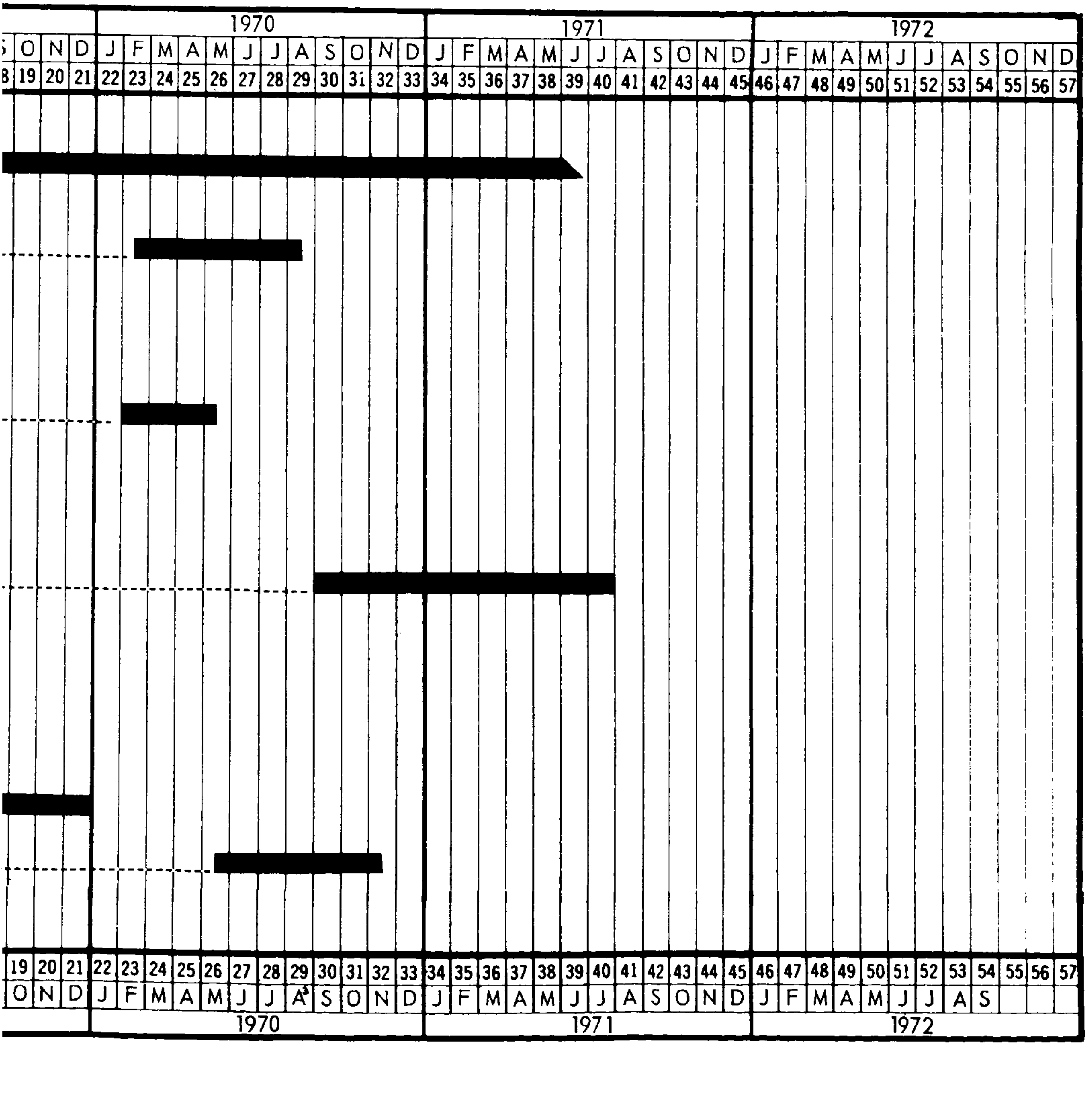
6-17-2

TIME PHASING OF TESTS (Continued)

ACTIVITY	A 1	M 2	J 3	J 4	A 5	S 6	O 7	N 8	D 9	J 10	F 11	M 12	A 13	M 14	J 15	J 16	A 17
	1968									1969							
PHASE C			■	■	■	■	■	■	■	■	■						
PHASE D												■	■	■	■	■	■
5.0 Typical Development Tests for Science Subsystem (continued)																	
Engineering Prototype Testing ⚠1																	
Performance – Acceptance																	
Performance – Sterilization Cycles																	
Performance – Margin Test																	
Performance – Selected Mission Environments																	
EMI Margin Test																	
Integrated Science Subsystem Bench Testing ⚠1																	
Power Up Testing & Systems Operation																	
Interface Verification Tests																	
EMI Margin Test																	
Performance Margin Tests																	
In-flight Checkout																	
Qualification Testing-Components ⚠1																	
Manufacturing Prototype Hardware																	
Performance – Acceptance																	
Performance – Sterilization Cycles																	
Performance – Mission Environments																	
Performance – Margin Test																	
Performance – Mission Life Test																	
6.0 Cabling Subsystem																	
Component Evaluation																	■
Prototype Wire Bundles Functional Evaluation (Engineer Model Tests)																	

⚠1 The following tests are included in the testing.

Figure 6–11 (Continued)


1970
1971
1972
O N D J F M A M J J A S O N D J F M A M J J A S O N D J F M A M J J A S O N D
19 20 21 22 23 24 25 26 27 28 29 30 31 32 33 34 35 36 37 38 39 40 41 42 43 44 45 46 47 48 49 50 51 52 53 54 55 56 57
19 20 21 22 23 24 25 26 27 28 29 30 31 32 33 34 35 36 37 38 39 40 41 42 43 44 45 46 47 48 49 50 51 52 53 54 55 56 57
O N D J F M A M J J A S O N D J F M A M J J A S O N D J F M A M J J A S
1970
1971
1972

6-18-2

FLIGHT CAPSULE TEST MATRIX

TEST TITLE	SECTION NUMBER	CAPSULE BUS SYSTEM	SURFACE LABORATORY SYSTEM	ENTRY SCIENCE PACKAGE
Material Tests	1.2.1	X	X	X
Wind Tunnel Tests	1.2.2	X		
Aerodynamic Decelerator Tests	1.2.3	X		
Thermal Control Subsystem	1.2.4			
Thermal Control Blankets	1.2.4.1	X	X	X
Heaters & Thermostats	1.2.4.2	X	X	X
Thermal Coatings	1.2.4.3	X	X	X
Heat Pipes	1.2.4.4		X	
Thermal Control Simulator	1.2.4.4	X	X	X
Structural/Mechanical Subsystems	1.2.5			
Structural Tests	1.2.5.1	X	X	X
Mechanical Devices	1.2.5.2		X	
Dynamic Tests	1.2.5.3	X	X	X
Heat Shield Tests	1.2.5.4	X		
Canister Pneumatic Tests	1.2.5.5	X		
Propulsion Subsystems	1.2.6			
Reaction Control	1.2.6.1	X		
Terminal Propulsion	1.2.6.2	X		
De-orbit Motor	1.2.6.3	X		
Pyrotechnic Tests	1.2.7	X	X	
Electronic Subsystems	1.2.8			
Guidance & Control		X		
Power		X	X	X
Antenna		X	X	X
Radio		X	X	X
Telemetry		X	X	X
Data Storage		X	X	X
Guidance Sensor		X		
Command		X	X	
Control (Antenna Steering)			X	
Sequencer		X	X	
Science Subsystem Test	1.2.9		X	X
Cabling Subsystem Test	1.2.10	X	X	X

Figure 6–12

FLIGHT CAPSULE SYSTEM TEST SCHEDULE (DEV & QUAL)

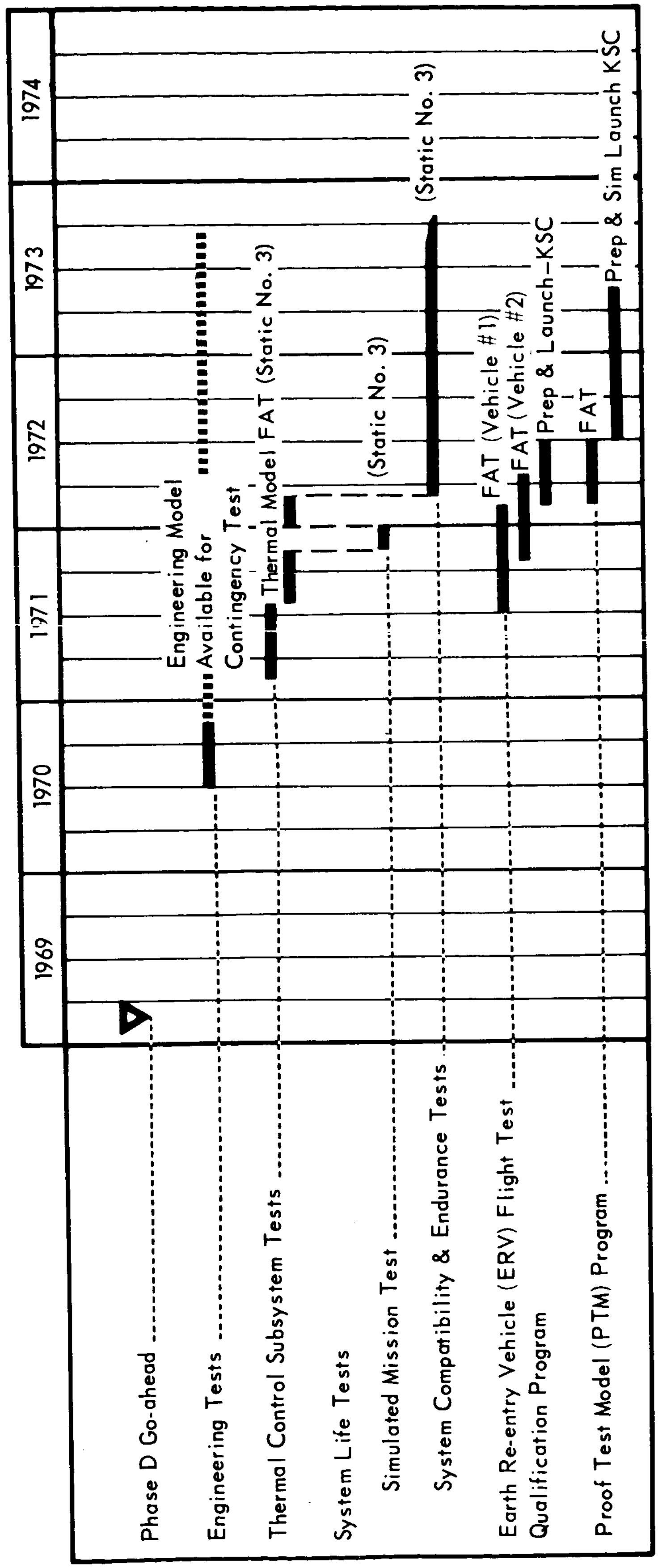


Figure 6–13

6–20

time dependency of the physical and functional characteristics of the subsystems, determining the causes and means of degradation as it occurs. These tests are performed by the CBS Contractor, supported in part by ESP personnel.

d. Proof Test Model (PTM) Test is a final qualification system test that uses flight qualifiable hardware. It demonstrates system performance capabilities and OSE compatibility, verifies flight hardware test procedures, and provides test organization training. From module testing to simulated launch, the PTM is tested within the Flight Acceptance Test format to qualification level standards and parameters.

6.4.4 Pre-Delivery Acceptance (PDA) Tests - These tests are conducted on each flight ESP, prior to delivery to the Capsule Bus integrating facility, to demonstrate the capability of the system to perform to the mission requirements. These tests are conducted in a manner consistent with Flight Capsule Flight Acceptance requirements and the test results are considered as the initial Flight Acceptance Test data.

The sequence of tests and periods of preparation are presented by Figure 6-14; test descriptions are on Figure 6-15.

6.4.5 Flight Acceptance Tests - These tests are conducted on each Flight Capsule, including the ESP, to demonstrate the capability of the systems to perform in accordance with the mission requirements. The ESP undergoes an Equipment Functional Check (EFC) prior to being installed in the Capsule Bus. The Flight Acceptance Test (FAT) Plan presented by Figure 6-16 establishes the sequence of tests and periods of preparation necessary to assure "mission readiness" at the time of launch. It is based upon the minimum amount of disassembly required and the exclusion of repetitive tests which produce little system improvement or engineering confidence.

Commonality of the test procedures and equipment at the SLS and ESP facilities, the Capsule Bus facility, and the launch site minimizes rejection of equipment due to variations in test configuration. Test descriptions are on Figure 6-17.

This test plan reflects the McDonnell "factory-to-pad" policy - the delivery of a Flight Capsule to the launch site in a "flight ready" condition, requiring only servicing and integration with the other elements of the space vehicle and ensuring an efficient launch site launch preparation program. This "factory-to-pad" approach makes maximum utilization of the Contractor capabilities possible and ensures a high level of confidence in mission success.

The FAT test teams are comprised of the personnel most knowledgeable about the

E.S.P. PRE-DELIVERY ACCEPTANCE (PDA) TEST PLAN
(NUMBER INSIDE SQUARE DENOTES TEST DESCRIPTION NUMBER)

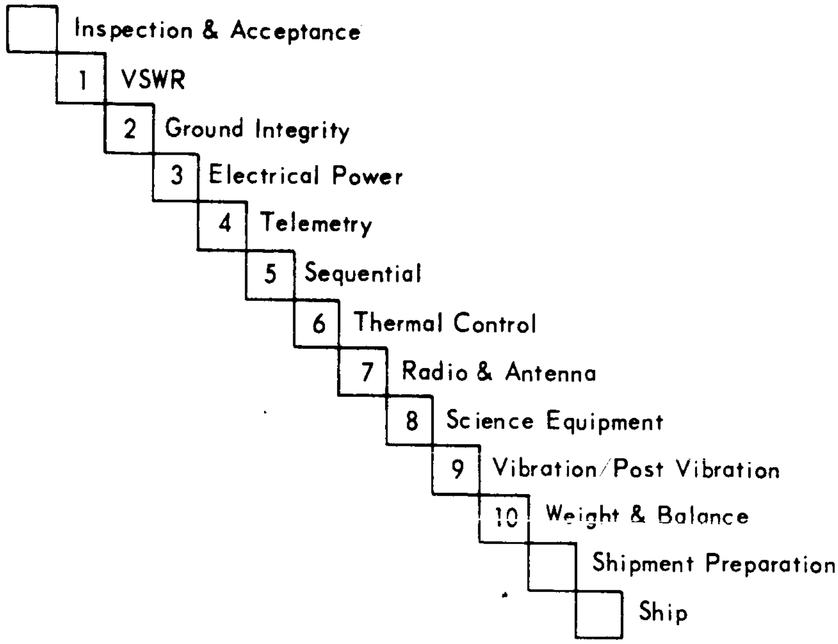


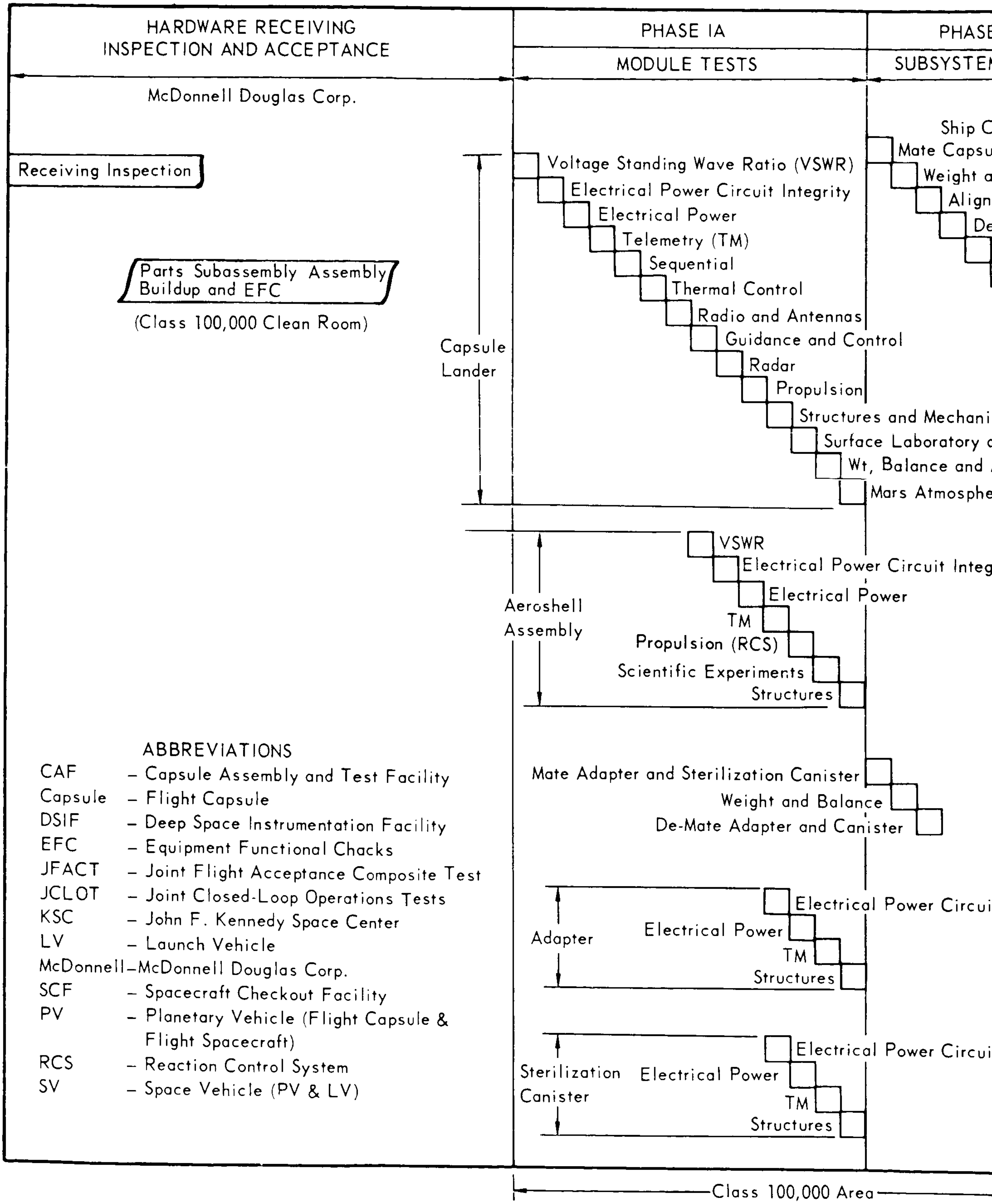
Figure 6–14

ESP PRE-DELIVERY ACCEPTANCE TEST DESCRIPTIONS

TEST NO. AND TITLE	OBJECTIVE
1. Voltage Standing Wave Ratio (VSWR)	To evaluate the characteristics of the RF paths within the ESP prior to energizing the Radio and Antenna Subsystems. Insertion loss measurements, VSWR values, and antenna phasing characteristics will be recorded for future reference.
2. Electrical Power Circuit Ground Integrity	To verify that the Power Subsystem is correctly connected and that is will be safe to energize the ESP in the next test. Fuses will be checked and tests performed to verify that no circuits are shorted to ground. Stray voltage and shield continuity tests will be performed on the pyrotechnic circuits.
3. Electrical Power	To verify prime power distribution, check buses, and power subsystems control devices with the Power Subsystem energized. To perform stray voltage tests on the pyrotechnic circuits in an energized environment. To verify the ability of the Subsystem Test Equipment (SSTE) to control the application and removal of power to the ESP. This test verifies that it will be safe to energize the other subsystems.
4. Telemetry (TM)	To evaluate the basic subsystem capability to monitor voltage, pressure, temperature, and other condition indicators. To verify that the SSTE provides satisfactory monitoring operation. This test provides assurance that the other subsystems will have TM monitor support when they are energized.
5. Sequential	To verify proper operation of sequencing devices and check tuning functions. To check pyrotechnic circuitry by means of squib simulators connected in place of the pyrotechnic devices. To evaluate redundant and back-up circuit. This test will verify proper operation of the Sequencer Subsystem prior to energizing the other subsystems.
6. Thermal Control	To verify satisfactory operation of the Thermal Control Subsystem. To check that portion of the subsystem which can be operated with only the Power and Telemetry Subsystems in operation. Verify integrity and operation of Thermal Control Subsystems prior to energizing the remaining subsystems.
7. Radio and Antenna	To evaluate the performance of the Radio and Antenna. Subsystem with power applied to the transmitters, antennas, and subsystem control devices. To evaluate signal strength, phase characteristics, power division, and subsystem performance. Record values of measurements to be used as reference data in subsequent tests.
8. Science Equipment	To verify proper operation of the Science Equipment. Perform pre-launch, cruise, and pre-separation checkout to evaluate condition of equipment. Operate experiment in mission sequence to verify electrical operation.
9. Vibration Post-Vibration	To demonstrate the ability of the ESP to withstand the loads imposed by simulated launch phase vibration. These subsystems which will be operating during the launch phase will be energized. Exitation will be applied along the horizontal and vertical axes. After the vibration tests all subsystems will be energized to verify that there has been no system degradation. The integrity of the squib simulators will be verified.
10. Weight, Balance, and Alignment	To determine the actual weight and C.G. location.

Figure 6-15

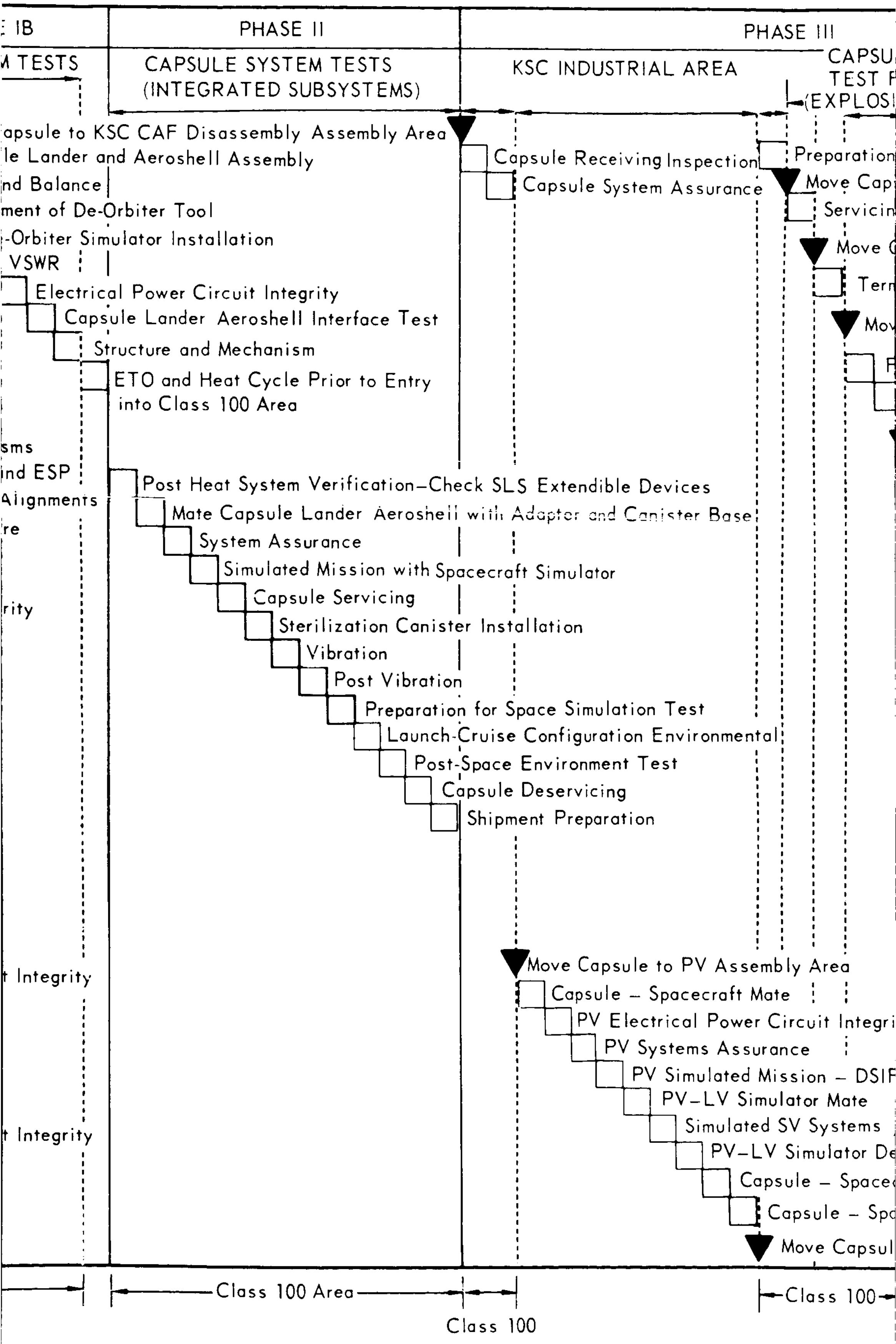
FLIGHT ACCEPTANCE TEST PLAN



ABBREVIATIONS

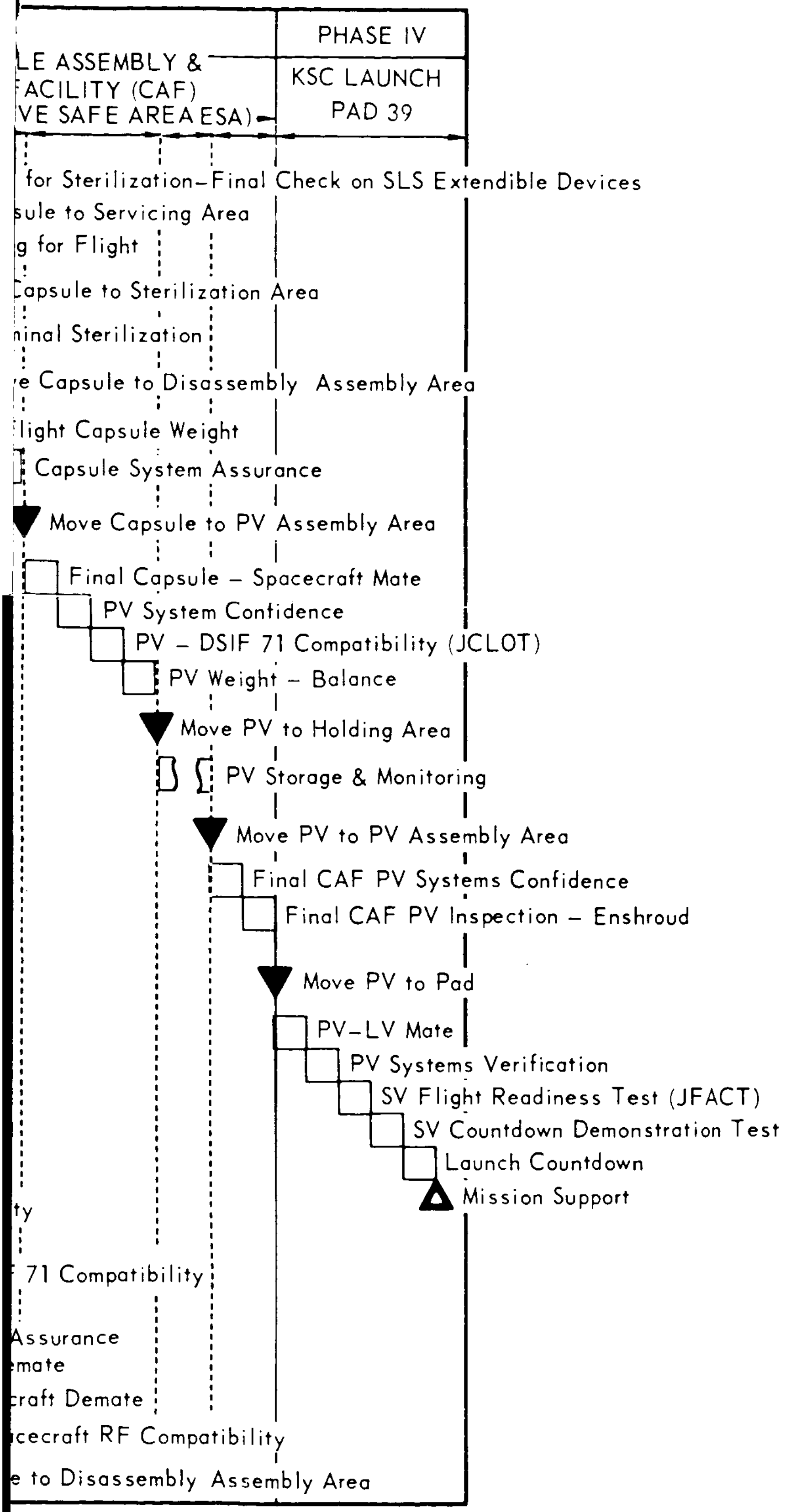
CAF – Capsule Assembly and Test Facility
Capsule – Flight Capsule
DSIF – Deep Space Instrumentation Facility
EFC – Equipment Functional Chacks
JFACT – Joint Flight Acceptance Composite Test
JCLOT – Joint Closed-Loop Operations Tests
KSC – John F. Kennedy Space Center
LV – Launch Vehicle
McDonnell–McDonnell Douglas Corp.
SCF – Spacecraft Checkout Facility
PV – Planetary Vehicle (Flight Capsule & Flight Spacecraft)
RCS – Reaction Control System
SV – Space Vehicle (PV & LV)

Figure 6–16



: IB
PHASE II
PHASE III
M TESTS
CAPSULE SYSTEM TESTS
(INTEGRATED SUBSYSTEMS)
KSC INDUSTRIAL AREA
CAPSU
TEST F
(EXPLOS
apsule to KSC CAF Disassembly Assembly Area
le Lander and Aeroshell Assembly
nd Balance
ment of De-Orbiter Tool
-Orbiter Simulator Installation
VSWR
Electrical Power Circuit Integrity
Capsule Lander Aeroshell Interface Test
Structure and Mechanism
ETO and Heat Cycle Prior to Entry into Class 100 Area
sms
ind ESP
Alignments
re
rity
Post Heat System Verification–Check SLS Extendible Devices
Mate Capsule Lander Aeroshell with Adapter and Canister Base
System Assurance
Simulated Mission with Spacecraft Simulator
Capsule Servicing
Sterilization Canister Installation
Vibration
Post Vibration
Preparation for Space Simulation Test
Launch-Cruise Configuration Environmental
Post-Space Environment Test
Capsule Deservicing
Shipment Preparation
Capsule Receiving Inspection
Capsule System Assurance
Preparation
Move Cap
Servicin
Move C
Term
Mov
F
t Integrity
t Integrity
Move Capsule to PV Assembly Area
Capsule – Spacecraft Mate
PV Electrical Power Circuit Integri
PV Systems Assurance
PV Simulated Mission – DSIF
PV–LV Simulator Mate
Simulated SV Systems
PV–LV Simulator De
Capsule – Spacec
Capsule – Spa
Move Capsul
Class 100 Area
Class 100
Class 100

6-24-2


PHASE IV
LE ASSEMBLY &
ACILITY (CAF)
VE SAFE AREA ESA)
KSC LAUNCH
PAD 39
for Sterilization–Final Check on SLS Extendible Devices
sule to Servicing Area
g for Flight
Capsule to Sterilization Area
ninal Sterilization
e Capsule to Disassembly Assembly Area
light Capsule Weight
Capsule System Assurance
Move Capsule to PV Assembly Area
Final Capsule – Spacecraft Mate
PV System Confidence
PV – DSIF 71 Compatibility (JCLOT)
PV Weight – Balance
Move PV to Holding Area
PV Storage & Monitoring
Move PV to PV Assembly Area
Final CAF PV Systems Confidence
Final CAF PV Inspection – Enshroud
Move PV to Pad
PV–LV Mate
PV Systems Verification
SV Flight Readiness Test (JFACT)
SV Countdown Demonstration Test
Launch Countdown
Mission Support
ty
71 Compatibility
Assurance
mate
raft Demate
cecraft RF Compatibility
e to Disassembly Assembly Area

FLIGHT ACCEPTANCE TEST DESCRIPTIONS

PHASE	TEST NO. AND TITLE	OBJECTIVE
1A	1. Voltage Standing Wave Ratio (VSWR)	To evaluate the characteristics of the RF paths within the Capsule Bus prior to energizing the Radio and Antenna Subsystems. Insertion loss measurements, VSWR values and antenna phasing characteristics are recorded for future reference.
	2. Electrical Power Circuit Ground Integrity	To verify that the Power Subsystem is correctly connected and that it is safe to energize the Capsule Bus in the next test. Fuses are checked and tests performed to verify that no circuits are shorted to ground. Stray voltage and shield continuity tests are performed on the pyrotechnic circuits.
	3. Electrical Power	To verify prime power distribution, check buses and power subsystems control devices with the Power Subsystem energized. To perform stray voltage tests on the pyrotechnic circuits in an energized environment. To verify the ability of the Subsystem Test Equipment (SSTE) to control the application and removal of power to the Capsule Bus. This test verifies that it is safe to energize the other subsystems.
	4. Telemetry (TM)	To evaluate the basic subsystem capability to monitor voltage, pressure, temperature, and other condition indicators. To verify that the SSTE provides satisfactory monitoring operation. This test provides assurance that the other subsystems have TM monitor support when they are energized.
	5. Sequential	To verify proper operation of sequencing devices and check timing functions. To check pyrotechnic circuitry by means of squib simulators connected in place of the pyrotechnic devices. To evaluate redundant and back-up circuits. This test verifies proper operation of the Sequencer Subsystem prior to energizing the other subsystems.
	6. Thermal Control	To verify satisfactory operation of the Thermal Control Subsystem. To check that portion of the subsystem which can be operated with only the Power and Telemetry Subsystems in operation. Verify integrity and operation of Thermal Control Subsystems prior to energizing the remaining subsystems.
	7. Radio and Antenna	To evaluate the performance of the Radio and Antenna Subsystem with power applied to the transmitters, antennas, and subsystem control devices. To evaluate signal strength, phase characteristics, power division and subsystem performance. Record values of measurements to be used as reference data in subsequent tests.
	8. Guidance and Control	To operate and evaluate the G and C subsystem. To verify proper power regulation (AC and DC) of the power supply, performance of the Inertial Measuring Unit (IMU), and the operation of the Guidance and Control Computer.
	9. Radar	To verify the ability of the Landing Radar to measure range and rate. To operate the Radar Altimeter and verify that the sequence of operation is correct and that the timing is accurate.
	10. Propulsion	To verify the integrity and operation of the propulsion subsystem. To leak check gas and liquid portions of the subsystems. To check pressure regulation, operation of transducers, and proper signals to the thrusters from the G & C electronics.
	11. Structures and Mechanisms	To verify that all mechanical devices operate properly and that the design clearances with structure are maintained.
	12. Surface Laboratory and Entry Science Package	To integrate the SLS and ESP with the Capsule Bus and verify proper operation. Perform pre-launch, cruise and pre-separation checkout to evaluate condition of experiments prior to deployment. Deploy experiments in mission sequence to verify electrical and mechanical operation.
	13. Weight, Balance and Alignment	To determine the actual weight and C.G. location. Alignment checks are performed on the IMU, the S-band antenna and other extendable devices in the SLS.
IA	14. Mars Atmosphere	To verify the operation of the SLS when subjected to a simulated Mars atmosphere. The Lander subsystems are not energized. The SLS is energized and the extendible devices deployed in mission sequence to verify

Figure 6–17

		that the system operates satisfactorily.
IB	15. Mate Capsule Lander and Aeroshell Assembly	To prepare for interface test.
	16. Weight and Balance	To determine the actual weight and Center of Gravity (C.G.) location of the Flight Capsule.
	17. Alignment of De-Orbiter Tool	To adjust the De-Orbiter Motor support links.
	18. De-Orbiter Simulator Installation	To install a Simulator representative of the flight article with weight, C.G. and physical configuration.
	19. Voltage Standing Wave Ratio (VSWR)	This test insures that the co-axial cables between the Capsule Lander and Aeroshell are satisfactorily engaged. This also permits end to end insertion loss measurements.
	20. Electrical Power Circuit Integrity	To verify that the mating of the Capsule Lancer and Aeroshell did nct cause a bus short to ground. The Pyrotechnic circuits shield continuity across the interface also are verified. To verify that multipath ground circuits do not exist.
	21. Capsule Lander/Aeroshell Interface Test	To verify all interface functions between the Capsule Lander and Aeroshell. Evaluate all subsystems operation for interaction between subsystems. Perform simulated mission to observe operating characteristics of the Capsule Bus.
	22. Structure and Mechanisms	To verify that all mechanical devices operate properly and that design clearances with structure are maintained. To verify that the proper clearance between the Capsule Lander and Aeroshell is maintained to assure satisfactory separation during the mission. Install squib simulators for subsequent tests.
IB	23. ETO and Heat Cycle	To decontaminate prior to entry into the Class 100 clean room.
II	24. Post Heat Cycle System Verification (Capsule Lander/Aeroshell)	To verify proper operation of all subsystems after exposure to Ethylene Oxide (ETO) and heat cycle. The test is essentially a repeat of the Capsule Lander/Aeroshell interface test performed just prior to entry in the Class 100 Clear Room. Additional tests are performed by partially extending the SLS devices to provide assurance that they operate after being subjected to the heat cycle.
	25. Mate Capsule Lander/Aeroshell with Adapter and Canister Base	To prepare for test of total system.
	26. System Assurance	To verify the satisfactory operation of all subsystems after mating the Capsule Lander/Aeroshell with the Adapter and Canister Base section. This is an integrated test of all subsystems except that portion which is contained in the removable Canister section.
	27. Simulated Mission (with Spacecraft Simulator)	To verify proper operation of all subsystems throughout a simulated mission. The sequence and timing of controlled functions is verified. The Flight Capsule/Spacecraft interface functions are verified by use of the Spacecraft simulator. The checkout is similar to that experienced in the launch and mission phases.

FLIGHT ACCEPTANCE TEST DESCRIPTIONS (Continued)

PHASE	TEST NO. AND TITLE	OBJECTIVE
	28. Flight Capsule Servicing	This is a preparation period rather than a test. The Flight Capsule is serviced with inert mediums to provide a condition as representative as possible to that which exists during the launch phase. Squib simulators are installed to represent the flight pyrotechnic devices to be verified after the Vibration Test. Test batteries are installed during this servicing period.
	29. Sterilization Canister Installation	To install forward Canister assembly.
	30. Vibration	The test demonstrates the ability of the Flight Capsule to withstand the loads imposed by simulated launch phase vibration. Those subsystems which are operating during the launch phase are energized. Exitation is applied along horizontal axis and the vertical axes.
	31. Post Vibration Test	To verify that there has been no system degradation. All subsystems are energized and the integrity of the squib simulators are verified by use of the OSE connector.
	32. Preparation for Space Simulation Test	To prepare the Flight Capsule and Space Chamber for test.
	33. Launch Cruise Configuration Environmental Test	To verify that the Flight Capsule operates satisfactorily under simulated flight environment.
	34. Post-Space Environment Test	To verify that there has been no system degradation.
	35. Capsule Deservicing	To deservice the Flight Capsule in preparation for delivery to the launch site. The forward Canister is removed and the inert mediums deserviced. The test batteries and squib simulators are removed at the launch site upon the installation of the flight units. This eliminates an unnecessary disassembly and assembly cycle.
II	36. Shipment Preparation	This is also a preparation period. The forward Canister is installed and the Flight Capsule charged with dry nitrogen to maintain a positive pressure within the Canister until the arrival at the launch site Class 100 room.
III	37. Capsule Receiving Inspection	To verify no damage was incurred in shipment.
	38. Capsule System Assurance	This test verifies the operation of the Capsule Bus after shipping and demonstrates the operational and Electro-Mechanical (EM) compatibility of the Flight Capsule and the System Test Complex (STC) at the launch site.
	39. Capsule – Spacecraft Mate	To prepare for test.
	40. PV Electrical Power Circuit Integrity	To verify that no multiple ground paths exist between the Flight Capsule and the Spacecraft.
	41. PV Systems Assurance	To verify the Flight Capsule/Spacecraft compatibility. To demonstrate that the PV operates with no subsystem interaction between the assemblies.
	42. PV Simulated Mission – Deep Space Instrumentation Facility (DSIF–71) Compatibility	To demonstrate the performance of the PV throughout the terminal launch count and an abbreviated mission. Demonstrate the capability of the DSIF to transmit to the Spacecraft and the relay of the commands to the Flight Capsule.
	43. PV–LV Simulator Mate	To prepare for test.

Figure 6–17 (Continued)

	44. Simulated Space Vehicle Systems Assurance	To demonstrate operation of the Flight Capsule system with the other elements of the Space Vehicle. This test verifies the compatibility of the Space Vehicle with all systems in operation. The Launch Vehicle simulator provides the L/V functions.
	45. PV–LV Simulator Demate	To prepare for test.
	46. Capsule – Spacecraft Demate	To prepare for test.
	47. Capsule – Spacecraft RF Compatibility	To demonstrate the capability to transmit and receive between the Capsule and the Spacecraft when physically separated.
	48. Preparation for Sterilization-Final Check on SLS Extendables	The forward Canister is removed in preparation for sterilization. The squib simulators, batteries, and de-orbiter simulator installed in-plant for the Vibration Test are removed. The SLS is energized and the final operational check on the extendable devices (particularly the S-band antenna) are performed.
	49. Servicing for Flight	To install flight batteries, flight pyrotechnics and service the propulsion subsystems.
	50. Terminal Sterilization	To decontaminate the Flight Capsules.
	51. Flight Capsule Weight	To determine actual weight after servicing.
	52. Flight Capsule System Assurance (Post Sterilization)	To demonstrate that the Flight Capsule system performs satisfactorily after the terminal sterilization.
	53. Final Capsule – Spacecraft Mate	To prepare for flight.
	54. PV Systems Confidence	To verify the proper operation of the PV with the Flight Capsule and Spacecraft mated for flight.
	55. PV DSIF–71 Compatibility	To verify compatibility of operation of the PV and the DSIF. This is the final check of this interface before going to the pad.
	56. PV Weight and Balance	To determine weight and C.G. location.
	57. Final CAF PV Systems Confidence	To verify satisfactory operation of PV prior to installation the shroud.
III	58. Final CAF PV Inspection – Enshroud	To prepare for move to the Launch Pad.
IV	59. PV–LV Mate	To prepare for Pad tests.
	60. PV Systems Verification	To demonstrate the satisfactory operation of the PV System on the pad. To verify the proper connection of the launch site OSE with the PV and satisfactory operation of the STC.
	61. SV Flight Readiness Test	This test demonstrates the operation of all elements of the SV (PV and LV). The PV–LV interface connection is verified in this test.
	62. SV Countdown Demonstration Test (CDDT)	This is a dress rehearsal of the launch countdown. This test verifies that the precise phasing of preparations, servicing, testing, removal of the MSS and all operational elements of the launch are in a ready state.
IV	63. Launch Countdown	To perform final tests and launch the SV.

Flight Capsule. These personnel are assigned from three sources, (1) the Project Design group, (2) the in-plant test operations group, and (3) the launch site operations group. Each of these groups is nearing the end of design effort, equipment and in-plant checkout procedures preparation, or launch site facility tests, requirements and preparation. The assignment of equal ratios of personnel from these three areas, plus selected ESP personnel, provides a balanced group with a high degree of versatility. Three teams are assigned to the PTM and to each of the four Flight Capsules from the initial in-plant tests until the time of launch.

Subsystems specialists from the launch teams are assigned to support NASA in monitoring Flight Capsule performance throughout the mission.

6.4.6 Element Plans - Those implementation element plans not previously discussed in this section are:

a. Design Engineering - The preferred concept has been selected to provide mission success and the major engineering milestones necessary to implement this concept are incorporated on the Master Schedule.

b. Quality Assurance Plan - This plan ensures the compliance of the flight article with the design requirements. Experience in stringent operational requirements on previous space programs is the basis for this plan.

c. Procurement Plan - The objective of this plan is to provide products on time and with the required technical excellence at the lowest cost. The selection of suppliers is based upon maximum use of experience on related products.

d. Science Integration - The Science Subsystem requires special emphasis. Experience on previous programs has shown that experiments are often the critical path. Interface definitions, test objectives, equipment qualification and delivery requirements consistent with those of the Flight Capsule are required.

e. Electromagnetic Compatibility (EMC) - Design and test requirements have been established to ensure electromagnetic compatibility within the Flight Capsule, and with the Spacecraft, Launch Vehicle, Launch Complex, and the Operational Support Equipment.

f. Sterilization Plan - This plan defines the controls required to assure observance of the Planetary Quarantine.

g. Reliability Plan - The control techniques and procedures to provide Reliability Assurance throughout the program have been prepared based upon the preferred concept.

h. Interface Control - This control Plan encompasses VOYAGER System-to-System interfaces versus Contractor responsibility, use and control of formal specifications, organization, and schedule requirements for the VOYAGER Flight Capsule.

i. Parts, Materials, and Processes Plan - This plan describes the Project leve activity regarding selection and control of parts, materials and processes applied to the VOYAGER Flight Capsule.

j. Operational Support Equipment (OSE) Implementation - Outlines policies, procedures, and scope of management and implementation effort required to design, procure, fabricate, test, install, and validate OSE.

k. Logistics Support - Provides the logistics direction and control that assure facilities, personnel, and equipment capable of performing the programmed tests and operational mission.

l. Facilities - Identifies the overall Flight Capsule facilities requirements.

m. Project-Hardware Accountability/Traceability and History (PATH) Systems - Provides basic requirements, procedures, and operations of data information systems (manual and automated).

n. Safety - Describes the safety organization, planning and procedures for in-plant and remote sites.

o. Training - Provides in-plant training requirements (sterilization procedures, equipment and system familiarization, and personnel proficiency evaluation).

p. Project Control - Describes the integrated use of the work breakdown structure, PERT, schedule interface log, cost-performance analysis, and a project communication center.

q. Data Management - Highlights the methods for establishing data requirements and the techniques for controlling and disseminating this data.

r. Configuration Management - Defines the approach for establishing the various configuration baselines and the means by which changes to these baselines will be controlled.

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3. Rosemount Engineering Co.; Minneapolis, Minn.
4. Servonic Instruments, Inc.; Costa Mesa, Calif.
5. Trans-Sonics, Inc.; Burlington, Mass.

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1. Rosemount Engineering Co.; Minneapolis, Minn.
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