REPORT NO. F694
31 AUGUST 1967

PART B1 ALTERNATIVES, ANALYSES, SELECTION

VOYAGER CAPSULE PHASE B FINAL REPORT

VOLUME III SURFACE LABORATORY SYSTEM

PREPARED FOR:
CALIFORNIA INSTITUTE OF TECHNOLOGY
JET PROPULSION LABORATORY
PASADENA, CALIFORNIA
CONTRACT NUMBER 952000

COPY NO. 23

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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 III, describes the McDonnell Douglas preferred design for the Surface Laboratory System. It is arranged in 5 parts, A through E, and bound in 8 separate documents, as noted below.

Part A	Preferred Design Concept	1 document
Part B	Alternatives, Analyses, Selection	3 documents, Parts B_1, B_2 and B_3
Part C	Subsystem Functional Descriptions	2 documents, Parts C_1 and C_2
Part D	Operational Support Equipment	1 document
Part E	Reliability	1 document

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

VOLUME III CROSS REFERENCE INDEX

SYSTEM/SUBSYSTEM	VOLUME III PARTS	PART A	APPENDIX A (TO PART A)	APPENDIX B (TO PART A)	PART B	PART C	PART D	PART E
		PREFERRED DESIGN CONCEPT Objectives, Constraints – System Description, Sequence of Operations, Subsystem Summaries.	ENVIRONMENTAL REQUIREMENTS	FUTURE MISSION CONSIDERATIONS	ALTERNATIVES, ANALYSIS, AND SELECTION Trade Studies, Supporting Analyses, and Results	SLS FUNCTIONAL DESCRIPTIONS Subsystem Descriptions	OPERATIONAL SUPPORT EQUIPMENT Equipment, Software and Trade Studies	RELIABILITY Constraints, Analysis, Results, Testing and Control
Surface Laboratory System								
Mission	Objectives	Section 1	1.1 Environmental Design Criteria	1.1 Exploration Strategies	–	–	1. Introduction & Objectives	–
Mission	Constraints	Section 2	1.1 Environmental Design Criteria	1.1 Exploration Strategies	–	–	2. Requirements & Constraints	1 – Reliability Constraints 4 – Program Requirements
Mission	Profile	Section 3.1	1.5 Mission Environmental Conditions	1.2 Mission Profile	4.7 Extended Mission	–	–	–
Mission	Operations	4.1 Sequence 4.2 Timeline 4.3 Contingency Modes	1.3 Source of Environmental Parameters	–	2 – Mission Analysis	–	8. Software	2 – Failure Mode, Effects, Criticality Analysis 3 – Quantitative Estimates
Design	General	3.2 Configuration	1.2 General 1.4 Environmental Design Requirements	–	1 – Study Approach & Analysis 3 – System Functional Requirements 4 – Major Trade Studies	–	3. Preferred Approach 3.2 Design Concept 6 – ASHE & Servicing Equipment 7 – SC Mounted SLS Equipment 10. Analyses & Trade Studies	5 – Component Part Reliability
Design	Standardization/Growth	11 – Summary	–	–	–	–	4.3.8, 4.5.8	–
Design	Weight/Physical Characteristics	5 – Summary & Supporting Data	–	1.6 Constraints	–	–	4.3.3, 4.4.3, 4.5.3	–
Reliability		6 – Philosophy, Implementation, Definitions	–	–	4.6 Resource Allocation	–	4.3.6, 4.4.6, 4.5.6	–
Planetary Quarantine		7 – Contamination Analysis, Design for Sterility	1.6 Sterilization & Decontamination	–	–	–	–	–
OSE		8 – General Description	–	–	–	–	Complete OSE Description 3.3 Equipment Summary 4 – System Level Support Equipment 4.3 STC 4.4 LCE 4.5 MDE	–
Interfaces (Also See Volume V)		9 – System Interface Summary	–	–	–	–	4.3.5, 4.4.5, 4.5.5	–
Implementation (Also See Volume VI)		10 – Schedule & Program Summary	–	–	–	–	4.3.7, 4.4.7, 4.5.7	–
Major Subsystems		Section 3.3	–	–	4.3 Analysis of SL Alternatives 5 – Subsystem Studies	Complete Subsystem Functional Descriptions	5 – SL Subsystems Level Test Equipment 5.9 Automatic Processor 5.10 Miscellaneous 9. Equipment Summary	–
Electrical Power		3.3.1 – Requirements, Equipment Description & Operation	–	1.4 Major Considerations	5.1 Power Studies	Section 1	5.3 EPS Test Set	See Part C – Section 1
Sequencer		3.3.2 – Requirements & Description	–	–	4.4 In-Flight Monitoring & Checkout 5.2 Sequencing & Timing Studies	2.1 Sequencer & Timer 2.2 Test Programmer	5.4 Sequencer Subsystem Test Set	See Part C – Section 2
Control		3.3.3 – Requirements & Description	–	–	5.3 High Gain Antenna Pointing Studies	Section 3	–	See Part C – Section 3
Telecommunications		3.3.4 – Requirements & Description	–	–	5.4 Telecommunications Studies	4. Radio Subsystem 5. Antenna Subsystem 6. Command Subsystem 7. Telemetry Subsystem 8. Data Storage Subsystem	5.5 TCM Test Set	See Part C – Sections 4, 5, 6, 7, and 8
Structure (Including Mechanisms)		3.3.5 – Reqmts & Description 3.3.5.6 – Mechanisms	–	–	4.2 Leveling 5.5 Structural/Mechanical	9. Structure 10. Mechanical	–	See Part C – Sections 9, 10
Pyrotechnic		3.3.6 – Requirements & Description	–	–	Section 5.6	Section 11	5.8 Pyro Initiation Test Set	See Part C – Section 11
Packaging and Cabling		3.3.7 – Description	–	–	Section 5.7	Section 12	–	See Part C – Section 12
Thermal Control		3.3.8 – Description	–	1.5 Major Considerations	Section 5.8	Section 13	5.7 TCS Test Set	See Part C – Section 13
Science		3.3.9 – Sequence & Description 3.3.9.4 – Integration	–	1.3 Major Considerations 2 – Stationary Laboratories 3 – Extended Sample Gathering 4 – Mobile Laboratories 5 – Mobile Systems Performance	4.1 Science Integration 4.5 Independent Data Package Study 5.9.1 Science Data Subsystem 5.9.2 Sample Acquisition & Processing 5.9.3 Science Instruments	14.1 Science Data Subsystem 14.2 Sample Acquisition & Processing Equipment 14.3 Science Instruments	5.6 Science Test Set	See Part C – Section 14

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PART B

ALTERNATIVES, ANALYSIS AND SELECTION

Selection of McDonnell's preferred Capsule Bus design was based on comparative studies of the performance of several candidate concepts. The objective of this selection was to determine a design which meets all of the constraints and which performs the capsule mission successfully. The hazards imposed by the sterilization requirement, long lifetime, and uncertain environment have necessitated a conservative approach utilizing redundancy, design margin, and operating flexibility.

We have sought optimization of the entire Flight Capsule, rather than any individual subsystem. Achievement of capsule landing, performance of entry science experiments, and performance of landed science experiments were treated as the primary mission objectives. Compatibility with growth toward future mission requirements also played a strong part in the selection. Probability of mission success was our most important single optimization criterion. Others were system performance, development risk, versatility, and cost.

Mission analyses determined the range of profiles which satisfy mission objectives and are compatible with the capabilities of the Flight Capsule and other VOYAGER systems.

Functional requirements on the various subsystems were established from the mission profile. The alternatives considered for satisfying these requirements and the analyses leading to selection of a preferred concept are described in this part of the report.

The complexity of system and subsystem interactions necessitates frequent reference to other parts of this volume. The selected configuration is described in Part A; the Subsystem Functional Descriptions are in Part C. Other reports generated during our VOYAGER studies are also referenced for the benefit of specialists who are interested in further detail.

SECTION 1

STUDY APPROACH AND ANALYSIS

The objective of the Phase B study was to select from among the various candidate techniques for performing the VOYAGER mission the combination best achieving the mission objectives within the constraints of Section A2. This was accomplished by a multi-step reduction in the number of alternates, so that only a few needed detailed analysis and evaluation. These high value candidates and the techniques used to select the preferred concepts will be discussed in this part.

1.1 BASELINE IDENTIFICATION DOCUMENT - The basic device for determining the preferred design was the use of a baseline identification document. (See Figure 3-1.) For each mission phase, various functions to be performed and alternative techniques to perform those functions were identified. One of these techniques was selected early as the preferred approach, based on data available at the time. The baseline document was then revised as further analysis revealed the desirability of a change in the selected approach.

The baseline document also served to identify critical trade studies, defined as those strongly affecting the entire design or those requiring extensive interdisciplinary effort. The interdisciplinary studies were conducted under the cognizance of the systems integration groups, but utilized the efforts of virtually all project personnel as a systems analysis resource. Engineering analyses supporting the trade studies defined the operational parameters and design conditions, but did not of themselves lead to selection of preferred approaches. The more important system analyses and some of the major trade studies are presented in Section 4. Trade-offs essentially within a single subsystem or discipline were handled within the affected technology and are reported in Section 5.

1.2 REQUESTS FOR TECHNICAL INFORMATION - Our design approach, development scheduling, and cost estimates for the SLS are supported by the technical capabilities of the industry through the use of Requests for Technical Information (RFTI). These RFTI, sent to appropriate suppliers of subsystems and components, contain basic performance requirements and request that the responder submit a recommended design, data to substantiate his recommendation, and estimated cost and delivery schedules. Figure 1-1 lists the subject matter of these RFTI and the names of companies responding with information.

SURFACE LABORATORY SYSTEM REQUESTS FOR TECHNICAL INFORMATION

ITEM	VENDOR RESPONSE
TV Cameras for Entry Science Package & Surface Lab	Westinghouse
Temperature Sensors for the Entry Science Package & Surface Lab.	Rosemont Engineering West Coast Research
Moisture Sensors for the Surface Lab.	Panametrics
Pressure Transducers for the Entry Science Package and Surface Lab.	Consolidated Controls Rosemont Engineering Consolidated Electrodynamics Servonics Stratham Instrument Transonics
Mass Spectrometer for Entry Science Package	Nuclide Corporation General Electric
Spectro-Radiometer for the Surface Lab.	Block Engineering
Sequencer & Timer	Conductron
Silver Zinc Batteries	Eagle-Picher Electric Storage Battery

Figure 1-1

1.3 FEASIBILITY TESTING - Whenever an extension beyond presently proven techniques was deemed necessary to provide confidence in a design concept selection or to assist in making a selection, a test was initiated. The testing activity, selected to complement NASA and JPL testing programs, is summarized in Figure 1-2. These test programs provided materials compatibility verification, aided the extrapolation of analytical efforts to Martian environment conditions, and guided the establishment of operational procedures. The tests are more fully described in Section VI B1.

1.4 SELECTION CRITERIA - All subsystem selections and interdisciplinary trade study decisions were made on the basis of optimizing the total Capsule System. Five broad criteria were selected to measure optimization. These are considered to be composed of several factors, each of which varies with each subsystem or trade study. This method provides an easily understood picture of the advantages and disadvantages of the candidates.

The selection criteria are:

Criteria	Typical Factors
Probability of Mission Success	Subsystem reliability
	Effect on other subsystems
	Vulnerability to environmental uncertainty
	Probability of violating quarantine
System Performance	Subsystem weight
	Attainability of desired landing site
	Quality of data to be obtained
	Environmental compatibility
Development Risk	Duration of development cycle
	Effect on other subsystems
	Need for state-of-the-art improvement
	Test complexity, confidence in results
Versatility	Ease of accommodating changing requirements
	Growth capability
	Ability to adapt to new environmental data
Cost	Fabrication ease
	Accessibility
	Unusual handling requirements
	Need for redundant development
	Special facilities

SURFACE LABORATORY SYSTEM FEASIBILITY TESTING

SURFACE ENVIRONMENT SIMULATION

- Development of a Martian Environmental Simulation Facility
- Dust Particle Behavior in a Simulated Martian Atmosphere
- Behavior and Characteristics of Simulated Martian Sand and Dust Storms
- Wind Blown Sand and Dust Tests
- Investigation of Martian Surface Phenomena

RELIABILITY

- Environmental Effects on Electronic Parts

STERILIZATION TECHNIQUES EVALUATION

- Microbiological Research
- Sterile Assembly
- Class 100 Facility Operation

TELECOMMUNICATION

- Antenna Breakdown Tests in Simulated Martian Atmospheres
- S–Band Antenna Radiation Pattern Tests

THERMAL CONTROL EVALUATION

- Effect of VOYAGER Mission Requirements on Thermal Control Coatings
- ETO Effects on Thermal Control Coatings
- Heat Pipe Demonstration
- Heat Pipe Control Valve Test

EXPERIMENTAL INSTRUMENTATION STUDIES

- Measurement of Wind Velocity at Low Pressure

Figure 1–2

These criteria were applied in two steps. Initial screening (to reduce a large number of candidates to a few high value prospects for detailed study) generally used a numerical rating system having the following values:

a.	Probability of mission success	.35
b.	System performance	.20
c.	Development risk	.20
d.	Versatility	.15
e.	Cost	.10

The surviving candidates, after additional study, were then entered into a matrix presentation briefly identifying the pros and cons applied for each factor or criterion. Selection of the preferred design approach used the numerical system as a guide, but ultimately depended on our accumulated experience and engineering judgement.

SECTION 2

MISSION ANALYSIS

Analysis of the sequence of experiments has defined the requirements for experiment integration, data transmission, and power. These requirements are also dependent on the time of day for Capsule landing and the initiation of Surface Laboratory operation.

The selection of the landing site and the time of landing is strongly influenced by the scientific interest in specific sites and the capability of the VOYAGER systems to land in particular regions. This selection is discussed in Section 2.2. Recommendations for landing sites, compatible with Spacecraft, Capsule, and Surface Laboratory operation, are contained in Section 2.2.4.

2.1 EXPERIMENT PROGRAM ANALYSES - In addition to the constraints imposed upon the mission profile by landing site and communications considerations, there are very real constraints imposed by the type of experiments we wish to perform and the instruments proposed to accomplish this. Some experiments should operate only during daylight, others during the period of dawn or dusk. The electrical power requirements and the data storage requirements also must be considered in the development of the mission profile and mission sequence.

During the Phase B study, the instruments, electrical power, and data storage requirements were defined for the experiments. These are representative of a typical SL for the 1973 opportunity. A preferred sequence was developed and this is presented in Part A of this volume. The following paragraphs discuss the constraints analyzed during the study.

Throughout the study of the VOYAGER Surface Laboratory, a principal goal has been to maximize the utilization of required equipment and the return of data while keeping the electrical power requirements to a minimum. This establishes the requirement for experiment design integration and the coordinated effort of the principal investigators involved in each mission.

2.1.1 Experiment Operation Constraints - Various experiments impose unique constraints on the Surface Laboratory mission sequence. Visual imaging experiments require daylight with Sun elevation angles 30° to 60° above the horizon. Thus, such experiments must be scheduled when the Sun is in these favorable positions. In turn, the Sun position relative to the landing site is affected by the latitude of the landing site and the character of the local physical features.

Specific life detectors may require a period of daylight for operation. If the landing occurs in the late afternoon, there may not be sufficient time remaining for completion of the experiment; for a morning landing, there obviously is a longer available period of sunlight.

Some experiments may depend upon the successful completion of another experiment. For example, the biochemical detector experiments depend upon the successful completion of the soil sampler experiments.

A set of typical experiments and their operating constraints (selected during the two years we have been studying the VOYAGER) are summarized in Figure 2 -1. These were used to develop the preferred mission sequence presented in Part A of this volume. Note the usage of the gas chromatograph in three different experiments: the subsurface gas analysis, the atmospheric gas analysis, and the soil volatiles analysis. The time references indicate that at least two different clocks are required, one for mission reference and one for solar reference.

2.1.2 Electrical Power Constraints - Detailed mission planning includes a careful analysis of the total power requirements as a function of time. Thermal control system heating and cooling require electrical power as a function of the local conditions at the landing site and the amount of internal heat generated by the SL subsystems. The communication subsystem has high electrical power requirements during periods of transmission. Various proposed experiments have significant electrical power requirements, i.e., the soil volatiles experiment has a 50 Watt oven for heating the soil samples and the soil sampler has a peak demand of 60 Watts.

Wide excursions beyond the average power level impose penalties upon the electrical power system design. The planning of the mission and preparation of the mission profile must consider the electrical power profile. It is composed of the SLS equipment requirements and the thermal control requirements. Since the thermal control requirements are fixed by the landing site environment, the SLS equipment should be sequenced - to the extent practical - to produce a power profile which minimizes sustained excursions from the average power level.

SCIENCE INSTRUMENT OPERATING CONSTRAINTS

EXPERIMENT (INSTRUMENT)	OPERATING MODES	MODE CHARACTERISTICS	OPERATING CONSTRAINTS
Visual Imaging (Facsimile Cameras)	a) Panoramic Stereo	$360^\circ \times 90^\circ$ Total angular coverage by both cameras, 10 minute duration.	Taken morning first day, transmitted first day.
	b) Medium Resolution Selected Area Coverage	Preprogrammed sequence for area coverage of experimentation sites, duration 8 minutes.	Taken afternoon first day, transmitted second day, solar angles 30–60° above local landing site plane desired.
	c) Command Coverage	Coverage of Earth commanded areas, may partially or completely override data of mode b).	Morning of second day reserved for these operations, not necessarily used.
Atmospheric Properties (Atmospheric Package)	a) Sunrise/Sunset	Sensors continuously energized, each instrument sampled every minute.	From one hour prior to sunrise and sunset continuing to one hour after each event.
	b) Day/Night	Sensors energized every 15 minutes, all instruments sampled within one minute.	All times other than mode a).
Insolation (Spectroradiometer)	a) Part 1	Wide field, broad band measurement of isolation using one detector.	One measurement every 15 minutes day and night, <1 minute.
	b) Part 2	Spectral scan, of incident radiation from 4 viewing directions using narrow field, multiple detectors and filters.	Daylight only, total of seven measurements taken at sunrise and sunset with other measurements equally spaced through day. Five minute/measurement (includes 4 views).
Soil Analysis (α Spectrometer)		Instrument takes count of ambient radiation background then analyzes a series of three soil samples prepared by surface sampler and processor.	Background count – 2 hr. soil analysis – 8 hr. each, has a single sample pan which must be dumped and refilled by sample processor prior to each analysis, constrains surface sampler operation.
Gas Analysis (Gas Chromatograph)	a) Subsurface & Atmospheric Gas Analysis	Uses two columns to separate constituents of a gas sample. Subsurface gas sample and atmospheric gas sample analyzed in series (10 min-	Thirty minutes per analysis with flushing operations. Subsurface gas pump cycled at start of mode to observe diurnal variations in composi-

Figure 2–1

		utes each).	tion, analyses at noon, midnight and three times at each sunrise and sunset period.
	b) Soil Volatiles Analysis	Uses four columns to separate constitutents in a gas sample obtained by pyrolyzing a soil sample.	Analyzes each of four soil samples gethered. 110 min. per analysis.
	c) Calibration	Calibration on all columns using standard gas sample.	Performed immediately prior to first analysis and immediately after last analysis, 50 min duration each.
Subsurface Probe	a) Sunrise/ Sunset	Nine thermocouples along length of probe + one reference, energized continuously, sampled every minute.	From one hr prior to sunrise and sunset continuing to one hr after each event.
	b) Day/Night	Nine thermocouples and reference energized and sampled every 15 minutes.	At all times other than a).
	c) Gas Pumping	Gas pump provides subsurface gas sample for gas analysis.	At start of each gas chromatograph subsurface gas analysis.
Life Detectors 1) Metabolism I	a) Normal	Operates to 12 hours.	After biological soil sample is available.
	b) In Situ	Operates to 5 hours.	After deployment, daylight only.
2) Metabolism II		Operates to 12 hours.	After biological soil sample is available.
3) Growth		Operates to 26-2/3 hours.	After biological soil sample is available.
(Soil Sampler & Processor)	a) Soil Sampling	Takes four soil samples at different locations and depths. First sample is for life detection exps and gas chromatograph, last three for α spectrometer and gas chromatograph, collection times in order 15, 28, 33 and 40 minutes.	First sample started immediately after deployment, last three samples phased to α spectrometer operations to have sample in processor 10 minutes prior to soil analysis.
	b) Sample Processing	Process soil samples and delivers to appropriate exps. Four operations at 10 minutes each.	After each soil sampling.

A peak of over 250 Watts occurs in the equipment power profile during the communication periods. These occur when the Earth is visible from Mars. The Earth is seen from Mars as a morning star during the 1973 mission landing dates. Thus the communication periods occur during the Martian morning. The communication equipment power profile is particularly sensitive to the time of landing.

a. Early Morning Landing
   - Long period of high power on landing
   - Short period of high power on second morning

b. Early Afternoon Landing
   - Short period of high power on landing
   - Long period of high power on second morning

c. Evening Landing
   - Long period of high power on first morning
   - Short period of high power on second morning

For extended periods of low power levels, heater power is required to maintain the minimum temperature. For extended periods of high power levels, cooling is required to maintain the equipment below the maximum temperature. Thus, thermal considerations tend to produce an electrical power profile that is essentially constant for the entire mission.

2.1.3 Data Storage Capacity - The telecommunications system has the capability of transmitting a specific number of information bits every second. Some experiments, such as the visual imaging, could produce 2.4×10^6 bits of information per picture and a total of 9×10^6 bits minimum during the mission. The design goal is 30×10^6 bits. This could create a tremendous backlog of data to be transmitted, and this in turn creates a data storage capacity problem.

The mission sequence must consider the data bits generated and the telecommunication system data bit transmission rate so as to minimize the amount of data storage capacity required to insure mission success. The data bit requirements for the selected experiment program were considered in the development of the preferred mission sequence. This sequence is defined in Part A, Section 3.2.

2.2 LANDING SITE SELECTION - Choices of landing sites require careful consideration, based on all available data, of the hypotheses of Mars topography and areas which may be conducive to biological growth. Evaluation of possible landing sites at the present time yields several which are compatible with the kinematic and communication constraints of the VOYAGER. Prior to launch, a particular site will be chosen for the nominal landing area. After Planetary Vehicle orbit insertion, examination of the data obtained from the orbiting spacecraft will be used to confirm the selected site or change to a suitable alternate area.

2.2.1 Experiment Considerations in Landing Site Selection - Landing site selection from the standpoint of achieving experiment objectives has two primary considerations: the landing site must possess adequate time periods with suitable lighting conditions to obtain the desired surface images, and the landing site should represent an area that has a high probability of containing biological material.

2.2.1.1 Surface Imaging Considerations - As a general criterion for visual imaging, the Sun elevation should be greater than 15°, but less than subsolar, in order to cast sufficient shadows for image interpretation. Thirty to sixty degrees has been used as a desirable range. The time available for imaging varies with landing site latitude and the time of year, i.e., on the difference in the latitude of the landing site and the subsolar point latitude. For 17 March 1974, as indicated in Figure 2-2, peaks at high and low latitudes occur because the solar angle does not exceed 60 degrees. The middle latitudes have a significant portion of their daylight hours (centered around local noon) with Sun angles greater than 60 degrees. Ground slopes can shift the time of day when particular Sun angles are obtained.

2.2.1.2 Biological Considerations - Excellent analyses of presently available data regarding the geological and environmental conditions on Mars are contained in Reference 2-1. The results of these analyses from the standpoint of biological implications of landing site selection are presented in Figure 2-3. Three classes of preferred landing areas in order of decreasing interest are:

a. Class A
   - Within zone of ecological growth
   - Within area of maximum migration of groundwater

b. Class S
   - Within survival temperature zone
   - Within area of maximum migration of groundwater

IMAGING TIME VARIATION WITH LANDING LATITUDE

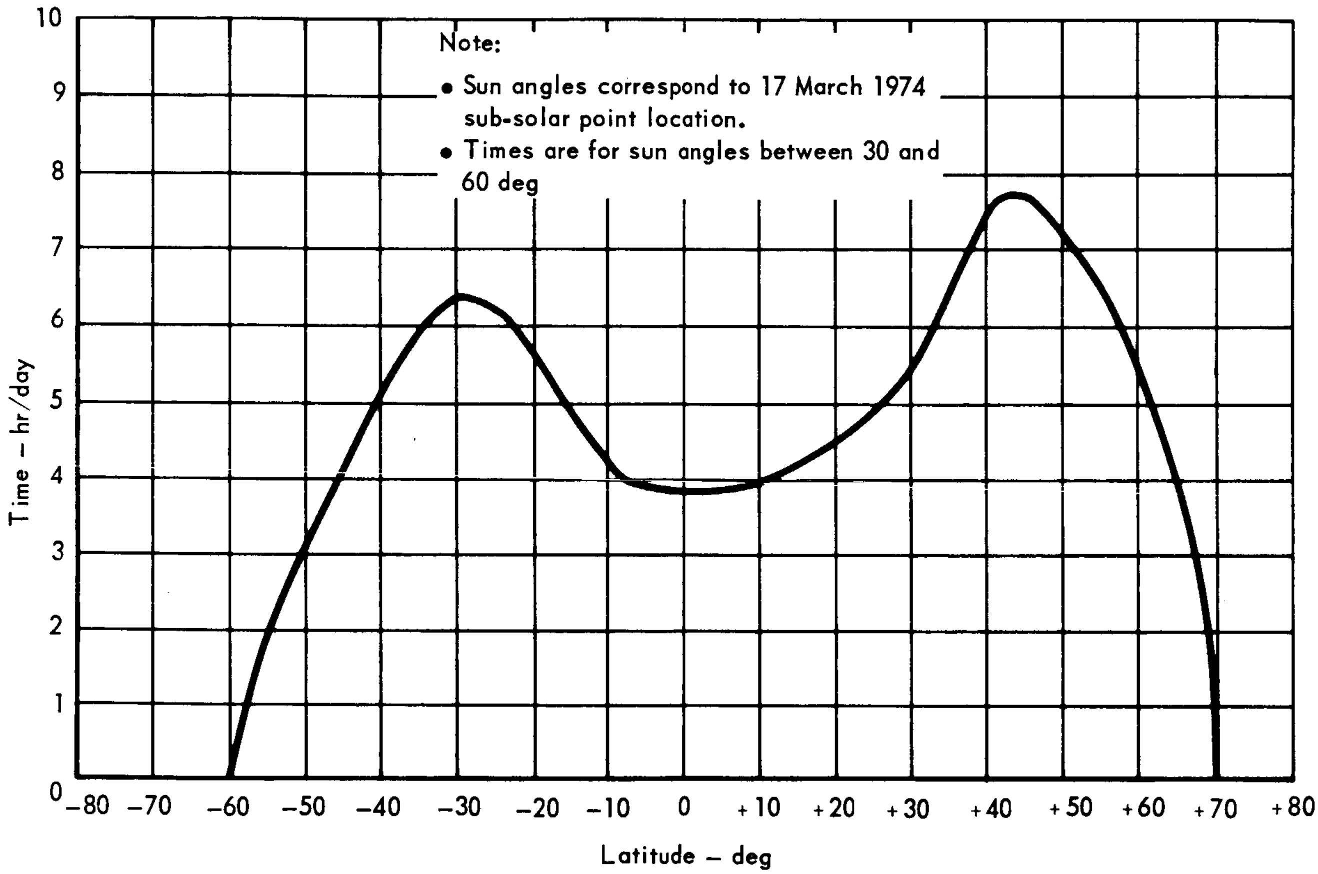
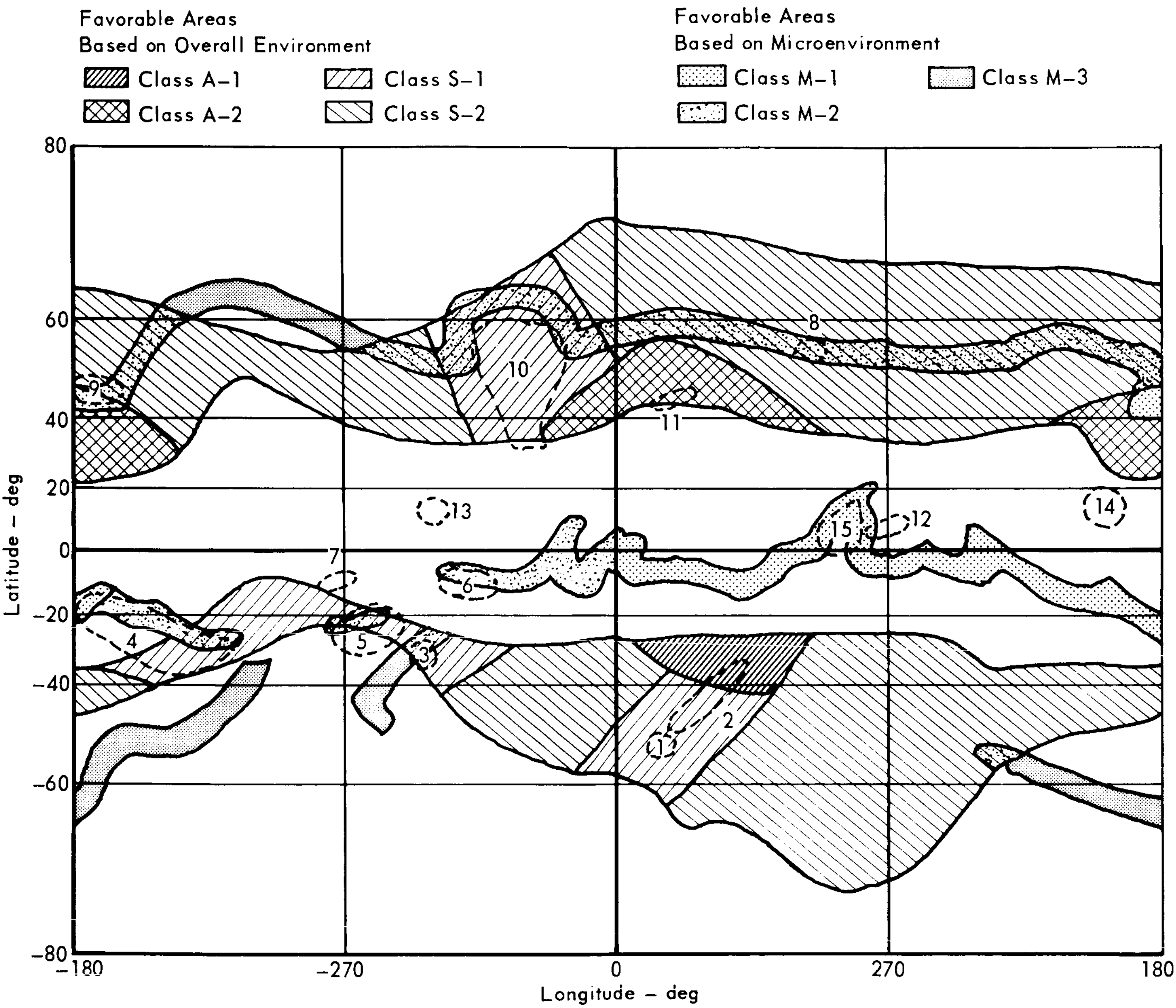


Figure 2–2

PREFERRED LANDING AREAS



Areas that React Strongly to the Wave of Darkening:

Southern Hemisphere	Northern Hemisphere
1. Depresso Hellespontica	8. Boresyrtis
2. Hellespontus	9. Propontis
3. Phrixi Regio	10. Mare Acidalium
4. Mare Sirenum	11. Ismenius Lacus
5. Solis Lacus	12. Thoth–Nepenthes
6. Aurorae Sinus	13. Lunae Palus
7. Tithonius Lacus	14. Trivium Charontis
	15. Syrtis Major

Figure 2–3

c. Class M - microenvironments

- o M-1 (low latitude)
- o M-2 (within survival temperature zone)
- o M-3 (high latitude)

Within these classes, particular areas (A-1, S-1) are affected by the wave of darkening in the Martian spring. For the 1973 opportunity, VOYAGER will arrive in the late winter to early spring for the northern latitudes. As indicated in Figure 2.-3 the area of maximum interest lies between 25 to 45 degrees South latitude and 295 to 355 degrees longitude. However, sites of secondary interest lie within a latitude band of 20°N to 25°S.

2.2.1.3 Experiment Preferred Landing Sites - In order to obtain the maximum amount of information about the Mars environment from the two landings in 1974, it is desirable to land first in a dark area and second in a light area. This provides the greatest amount of data from two landings for correlation of biological material and surface environment.

2.2.2 Kinematic Considerations of Landing Site Selection - Selection of landing sites are limited by the scientific interest in particular locations and the kinematic restrictions imposed by specific mission constraints. These mission constraints limit the choice of mission profiles; each profile has only a limited flexibility of landing areas. The landing area available for a given profile depends on the time during the mission sequence that a selection or decision is made. Launch date, arrival date (including separation), Mars orbit selection, descent trajectory design, and arrival and post-arrival event timing represent various events where landing site selection is influenced. The preferred mission profile was formulated to demonstrate, by way of example, a mission which best meets the majority of mission constraints, especially as related to Capsule landing and efficient operation of the Surface Laboratory.

From a kinematic point of view, landing site availability and flexibility are best described in terms of landing latitude. Event sequencing is an important consideration in the selection of a specific longitude, but of less influence in establishing landing site availability and flexibility.

2.2.2.1 Mission Profile Considerations - The selection of a mission profile must be within the gross bounds of the launch and Planetary Vehicle capability, satisfy the objectives of the Spacecraft, and meet the requirements of the Capsule and Surface Laboratory engineering and experiments. The mission profile which best meets these requirements will influence the availability of landing regions.

Accordingly, several of the constraints used in the selection process for our preferred mission profile do contribute directly to the landing site definition. These include:

- o Suitable surface lighting at the landing site (15 to 30 degrees to the terminator).
- o Landing in regions with seasonal color change (within latitudes between 10°N to 40°S), Reference 2-2.
- o Continuous descent communication between Capsule and Spacecraft.
- o Maximum data transmission prior to the onset of Martian night.

Conformance to these constraints is important in the formulation and subsequent selection of the preferred mission profile. The preferred mission profile meets these constraints.

Other mission constraints affect landing site availability to varying degrees. Noteworthy are those restrictions related to launch-arrival date selection, orbit selection, and descent trajectory design. These constraints impose bounds on the orbit position and descent trajectory, and consequently limit the areas for landing.

2.2.2.2 <u>Landing Site Availability</u> - Mission profile has a strong influence on Capsule landing site. Three prospective profiles,landing at 25° from the morning terminator, 25° from the evening terminator, and at 50° from the morning terminator, have been investigated for a typical case (1000-20,000 km altitude orbit, 40° orbit inclination, -20° entry flight path angle, and 327° landing anomaly. Figures 2-4 to 2-6 present contours of landing site latitude on the 1973 launch opportunity plot for the three profiles and show a more southerly latitude for the forenoon and evening landers. For the morning lander, Figure 2-4, the variation of landing latitude across the available launch opportunity is between 1°N and 22°S. For the forenoon lander, Figure 2-6, the range of latitudes is 8°S to 35°S; for the evening lander, Figure 2-5, the range is 30°S to 39°S.

It should be noted that the landing site is out of view of the Earth for the evening terminator lander (no post-landing direct-link communications with Earth). For the forenoon lander, the landing point is not positioned for good descent television (15° to 30° from terminator). Therefore, the only profile which fully satisfies the landing site constraints is the morning terminator lander. However,

LANDING LATITUDE VARIATION FOR MORNING TERMINATOR LANDER
1973 LAUNCH OPPORTUNITY

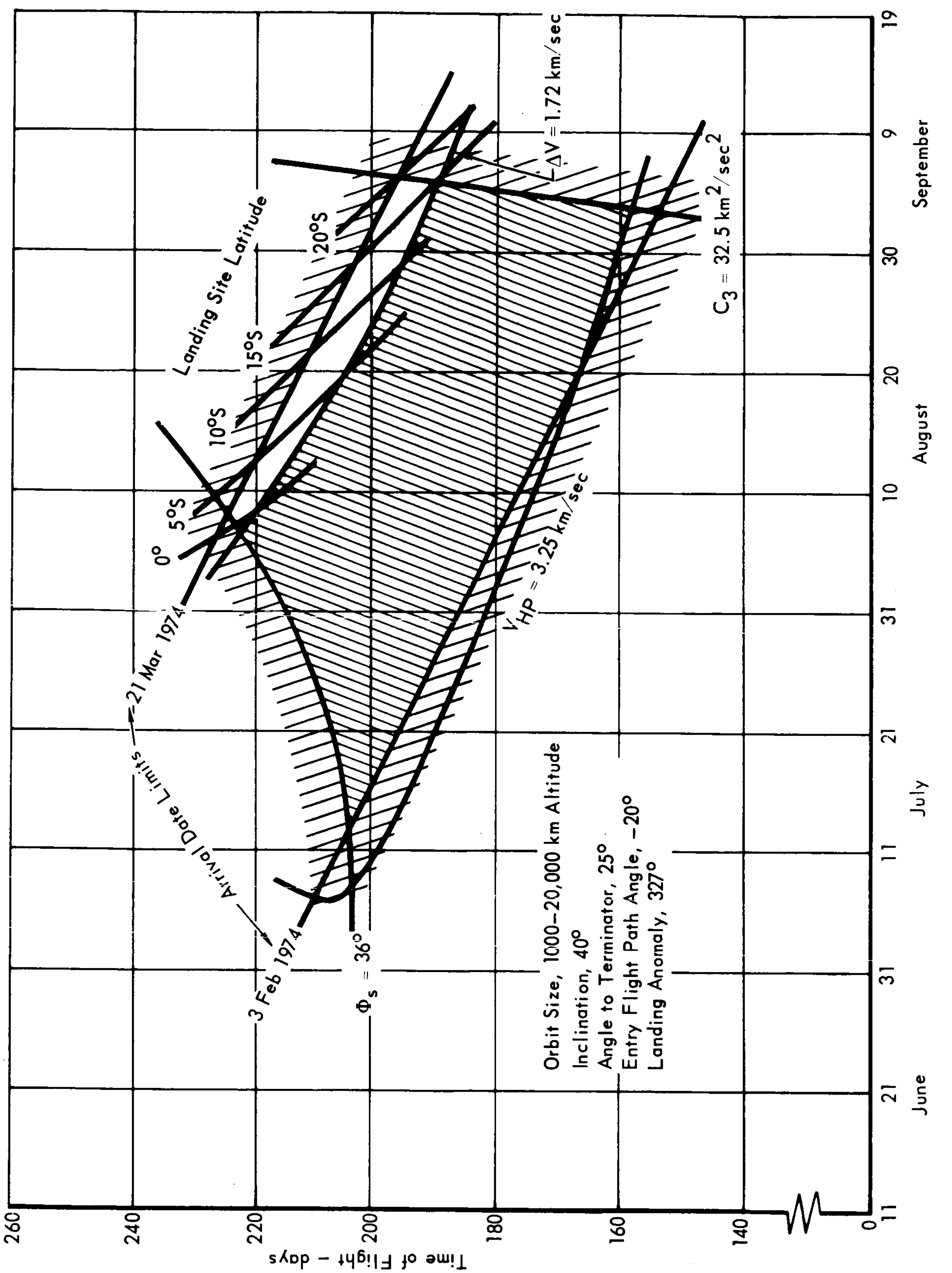


Figure 2-4

LANDING LATITUDE VARIATION FOR EVENING TERMINATOR LANDER
1973 LAUNCH OPPORTUNITY

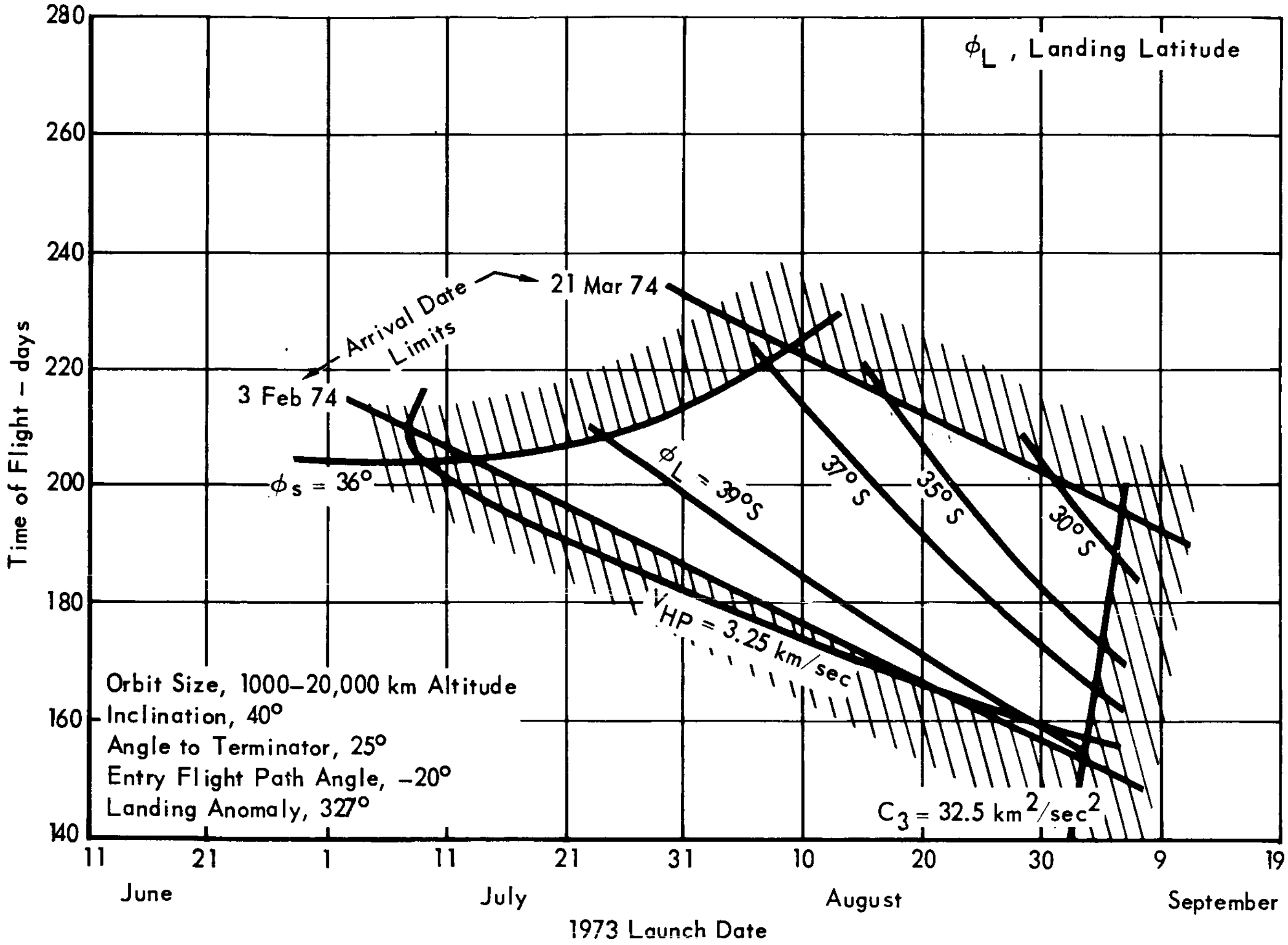


Figure 2-5

LANDING LATITUDE VARIATION FOR FORENOON LANDER
1973 LAUNCH OPPORTUNITY

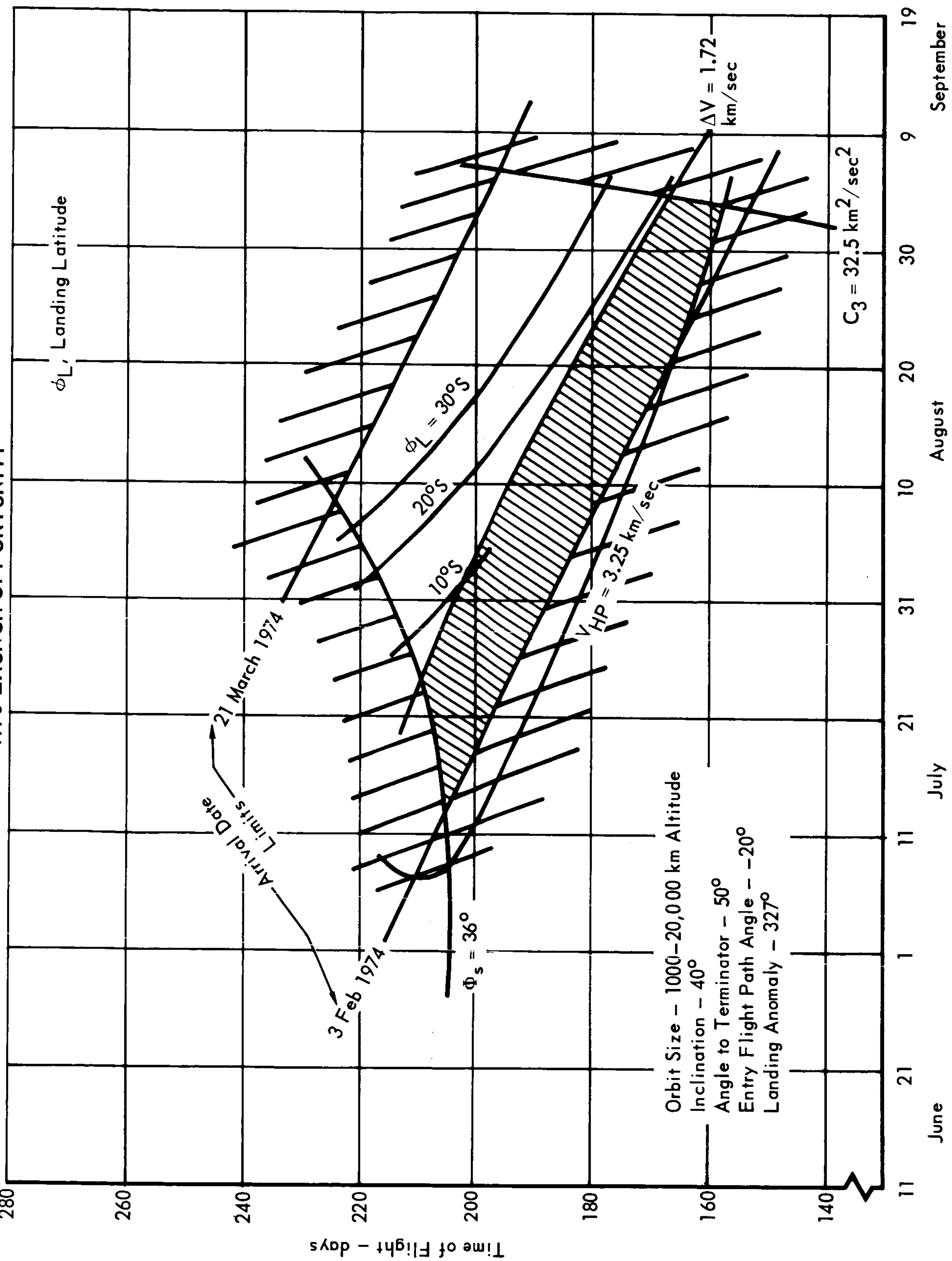


Figure 2-6

the others do provide flexibility, if desired, but at the price of compromise to landing site related constraints.

This compromise is illustrated in Figure 2-7. Landing site availability for all three profiles is shown in terms of landing site latitude and incremental longitude from the subsolar point. A single launch arrival date is used. The regions available to the individual profiles are cross-hatched. The range of orbit inclinations (30° to 50°) is compatible with mission constraints on orbit orientation. Zero incremental longitude corresponds to local noon. The limit of Earth view ($\Gamma_E = 34°$) is indicated. As may be seen, almost any landing occurring later than mid-afternoon will not be visible from the Earth. However, nearly the entire desired latitude band from 10°N to 40°S is achievable for the morning or forenoon landers.

In addition to the effects of mission profile selection, landing site availability and flexibility can be analyzed in stages. These generally represent the principal periods where selection decisions are possible. They are:

- o Availability through selection of launch-arrival date.
- o Flexibility through selection of Mars orbit position.
- o Availability due to design of descent trajectory.

Launch date is not a useful tool to obtain landing site flexibility. The landing latitude does vary with launch date, Figure 2-8, but the flexibility must be achieved by sacrificing some of the total launch period.

A greater flexibility in landing sites can be achieved through orbit selection, either by position adjustment in terms of inclination and line of apsides or by adjustment in orbit size. This flexibility is illustrated in Figure 2-9 for the preferred mission profile. Landing latitude is shown as a function of orbital inclination and Capsule landing location (central angle to the terminator). A launch date later than 7 August 1973 would result in a general shift toward more southerly latitudes. Obviously, greater flexibility in landing latitude selection is possible with a slight relaxation of surface lighting angle. Orbital repositioning (apsides adjustments) resulting from changes in the design descent trajectory does not affect landing latitudes.

Descent trajectory design, however, does have an influence on landing location. For the preferred mission profile, more shallow entry angles, Figure 2-10, will move the landing site to more southerly latitudes, but farther from the morning terminator. Deorbit anomaly also provides flexibility in landing location. The deorbit deflection angle provides another variable. The greater

LANDING SITE AVAILABILITY

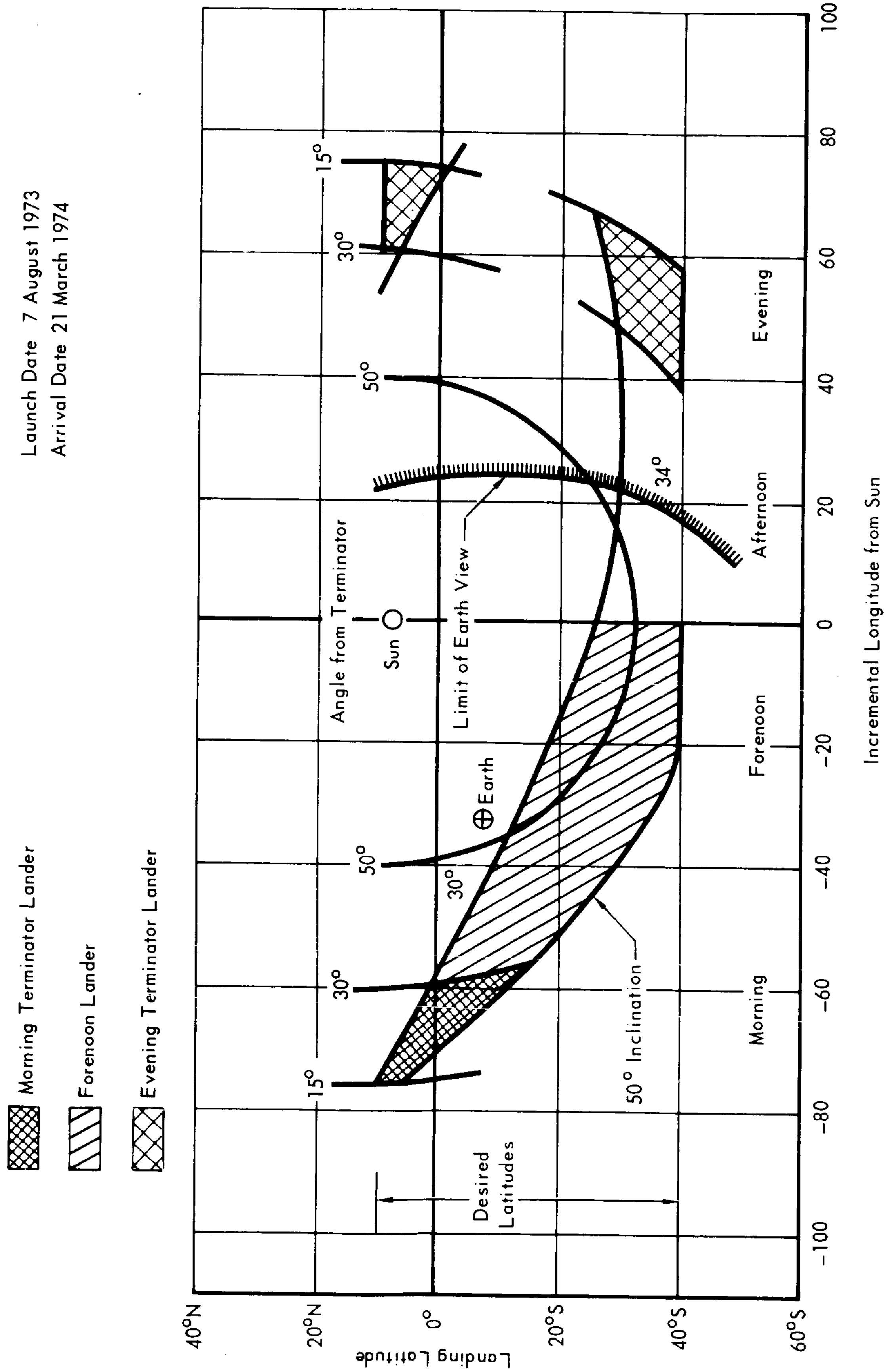


Figure 2-7

EFFECT OF LAUNCH DATE

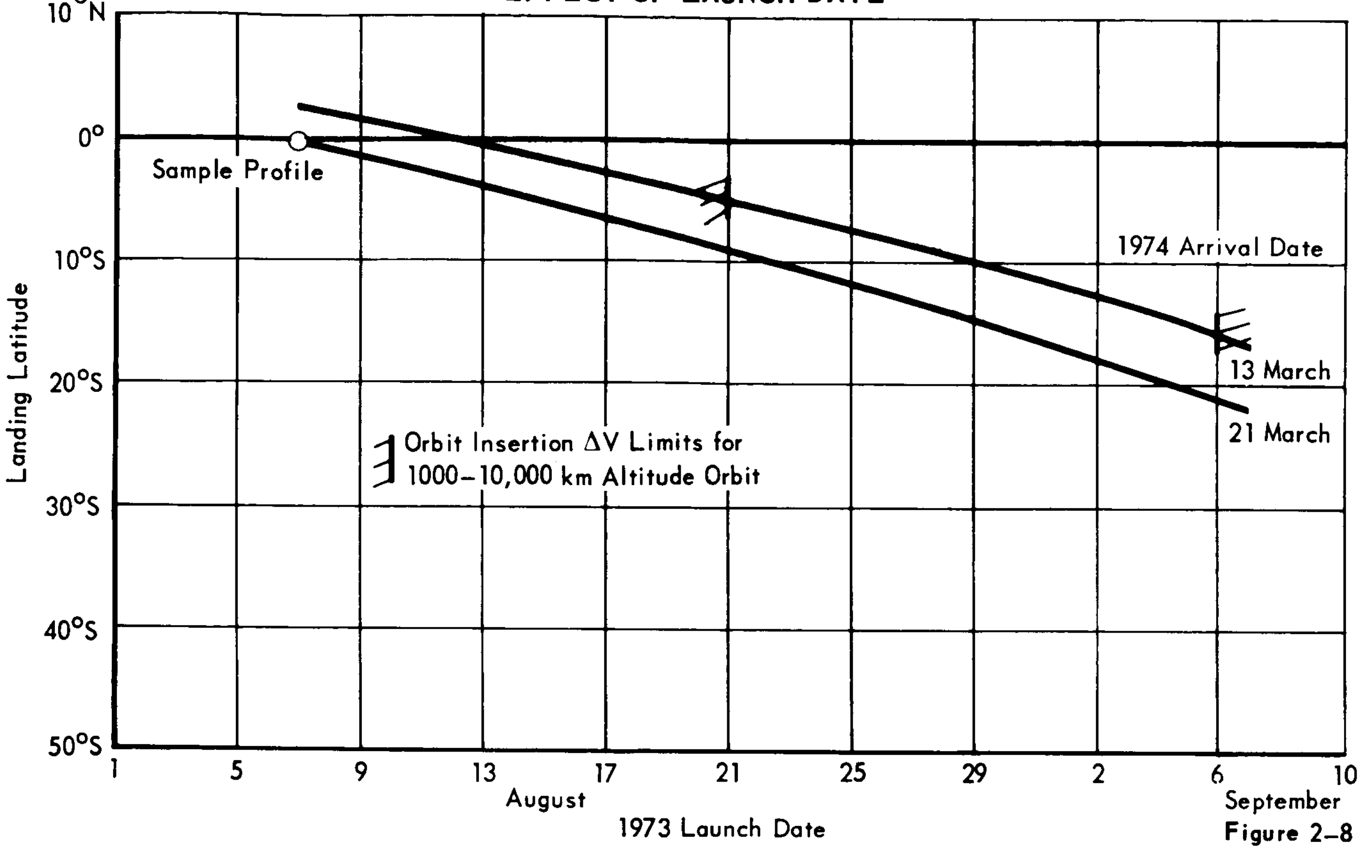


Figure 2-8

EFFECT OF ORBIT INCLINATION

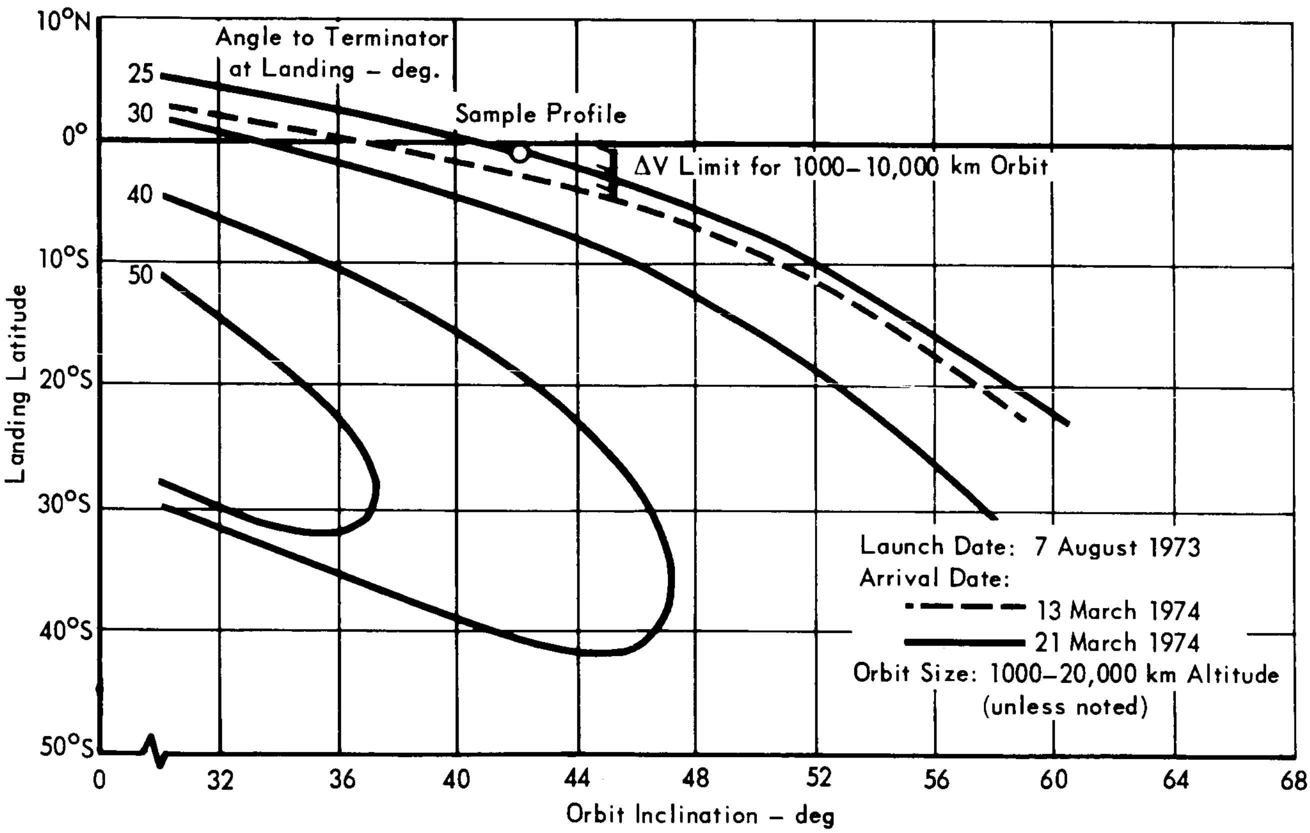
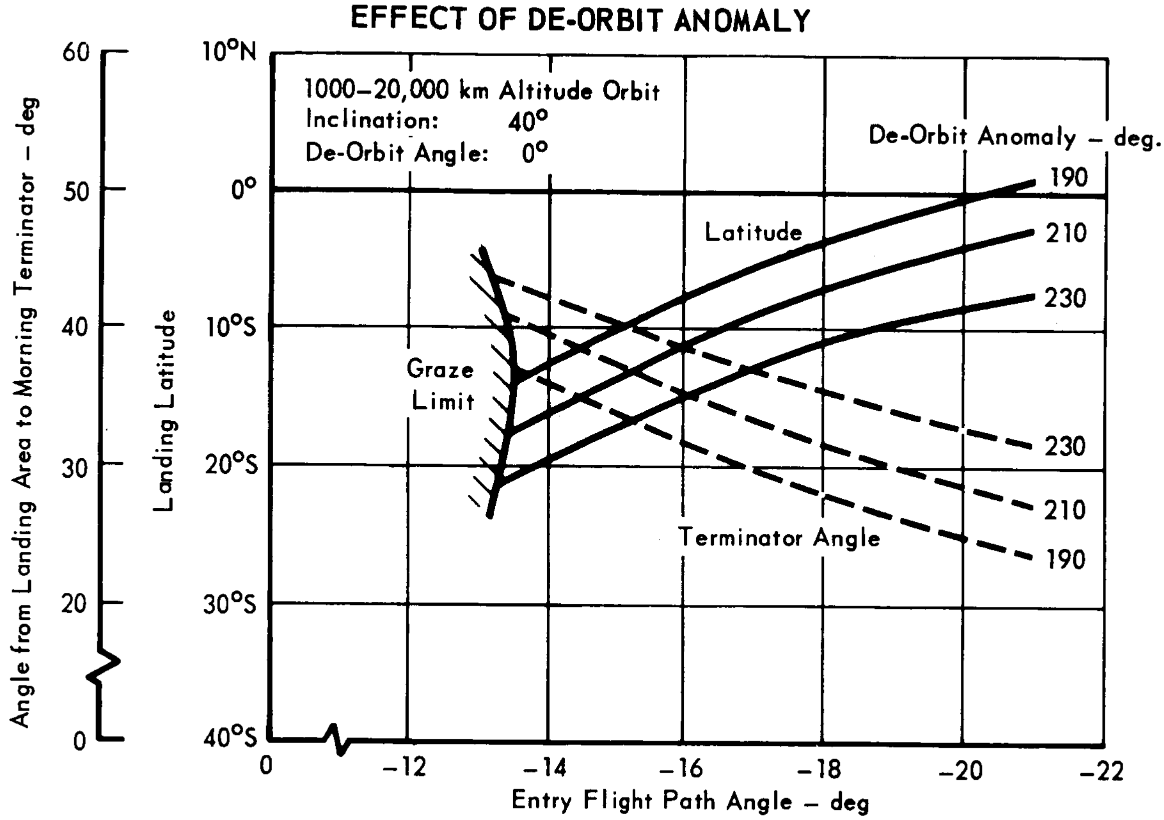


Figure 2-9


EFFECT OF DE-ORBIT ANOMALY
1000–20,000 km Altitude Orbit
Inclination: 40°
De-Orbit Angle: 0°
De-Orbit Anomaly – deg.
190
210
230
230
210
190
Latitude
Graze Limit
Terminator Angle
Angle from Landing Area to Morning Terminator – deg
Landing Latitude
10°N
0°
10°S
20°S
30°S
40°S
60
50
40
30
20
0
0
–12
–14
–16
–18
–20
–22
Entry Flight Path Angle – deg

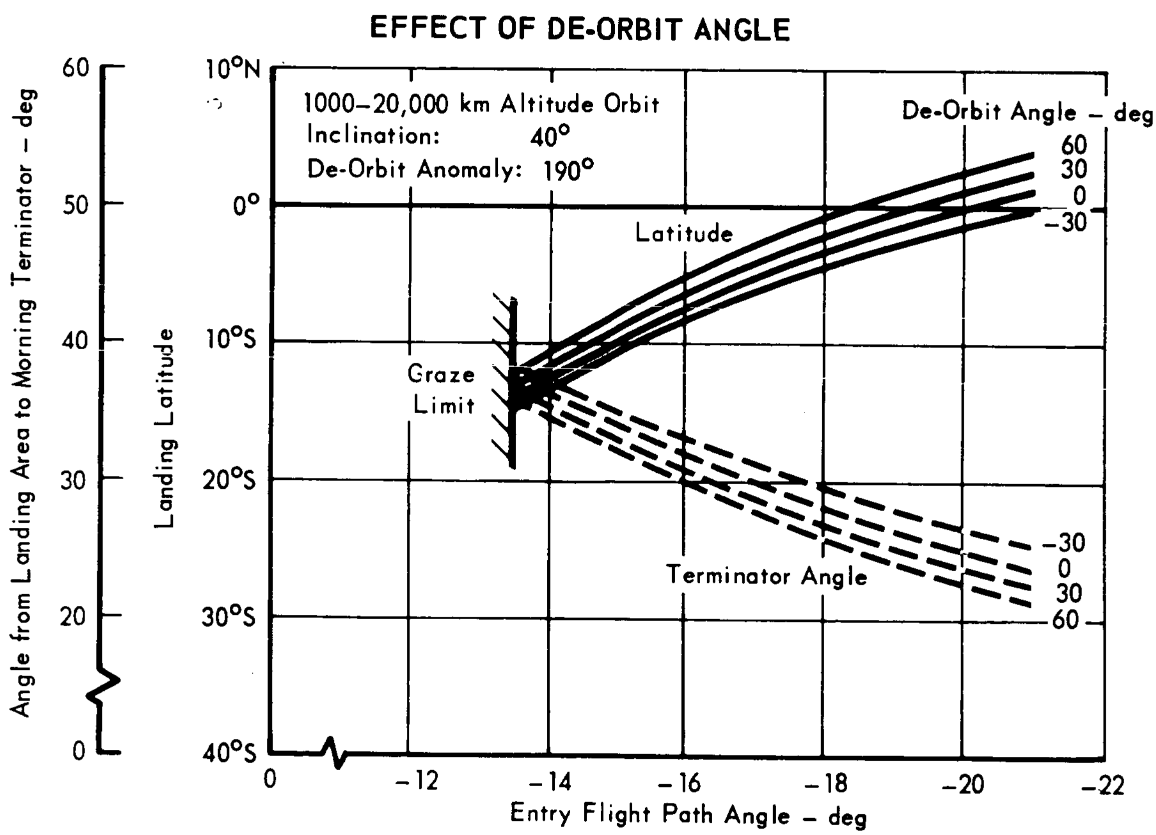

EFFECT OF DE-ORBIT ANGLE
1000–20,000 km Altitude Orbit
Inclination: 40°
De-Orbit Anomaly: 190°
De-Orbit Angle – deg.
60
30
0
–30
–30
0
30
60
Latitude
Graze Limit
Terminator Angle
Angle from Landing Area to Morning Terminator – deg
Landing Latitude
10°N
0°
10°S
20°S
30°S
40°S
60
50
40
30
20
0
0
–12
–14
–16
–18
–20
–22
Entry Flight Path Angle – deg

Figure 2–10

effectiveness of entry flight path angle to vary the landing latitude is significant.

2.2.2.3 Summary - Sufficient flexibility exists to select landing sites for the preferred or closely related mission profiles. This flexibility is generally achievable with little relaxation of VOYAGER constraints or mission objectives. As may be noted, aside from mission profile selection, orbit selection appears to be the most attractive way to achieve landing site flexibility without relaxing constraints. Descent trajectory adjustment offers flexibility in landing site selection following achievement of the Martian orbit. The actual band or range of latitudes is established, however, by the launch date and arrival date.

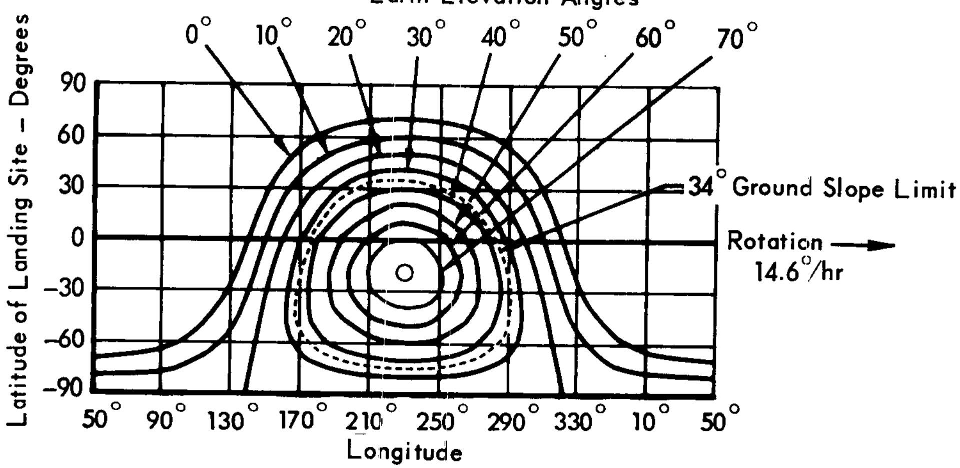
2.2.3 Post-Landed Communication Considerations - The latitude of the landing site, combined with the calendar date of landing, determines the basic operating conditions for communications to Earth. For the short duration Surface Laboratory missions of the 1973 mission, the time of landing, morning or evening, determines the amount of communications time available immediately after landing. The communications time and pointing accuracy of the high gain antenna for the high rate radio link are affected by the dispersions (errors) of the descent trajectory from orbit. From a communications viewpoint, landing sites near the morning terminator and the Martian equator are preferred because they allow a near maximum of communications time.

Communications time is important to determine the amount of data that can be transmitted or, in conjunction with communication distance, the data rates and radiated power requirements to transmit a fixed quantity of data. Elevation angles set the boundaries for the tracking program for a high gain antenna and the beamwidth of a fixed antenna. The latitude of the site and the date of landing, combined with the rotational rate of Mars, determine the time histories of the elevation angle of Earth. The time histories of the elevation angle, as limited by the ground slopes at the site, are used to determine the communications time available at a given latitude. The gross effects of site latitude and landing date are shown in Figure 2-11. The sites of maximum communications time move northward as the arrival date becomes later. Also of interest is that certain latitudes, with the 34° ground slope condition, do not permit communications to Earth.

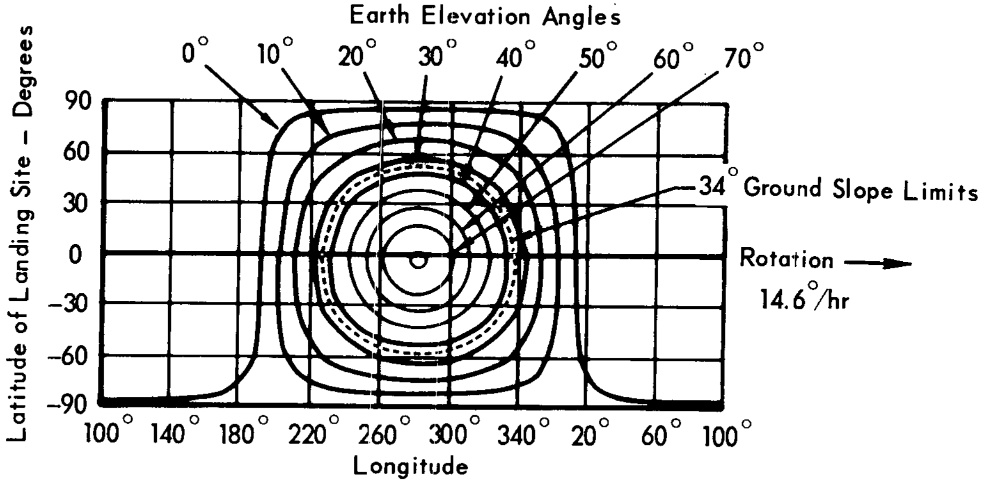
EFFECTS OF LANDING SITE LATITUDE & LANDING DATE UPON COMMUNICATIONS OPERATION

ELEVATION ANGLE OF EARTH

LANDING DATE – 16 JANUARY 1974
0 HOURS U.T.



LANDING DATE – 6 APRIL 1974
0 HOURS U.T.



LANDING SITE LATITUDE	COMMUNICATION TIME (Hours/Day)	
	LAND 16 Jan. 1974	LAND 6 April 1974
+90°	0	0
+80°	0	0
+70°	0	0
+60°	0	0
+50°	0	2.0
+40°	0	5.1
+30°	3.4	6.1
+20°	5.1	6.9
+10°	6.3	7.3
0°	7.1	7.6
-10°	7.7	7.7
-20°	8.1	7.4
-30°	8.4	6.9
-40°	8.5	6.0
-50°	8.3	4.2
-60°	7.8	0
-70°	5.7	0
-80°	0	0
-90°	0	0

1. No Dispersions
2. 34° Ground Slopes

Figure 2–11

The local Martian time of landing and the landing latitude set the limits for the first day communications opportunity. Normally, we classify the landing sites into two categories, near the morning or evening terminator. As shown in Figure 2-12, landings near the evening terminator, within 15° to 30° of the actual terminator for descent imaging, do not allow communications to Earth after landing. Therefore, for evening terminator landings, the Surface Laboratory must undergo the Martian night before communications to Earth can be established.

Morning terminator landings allow communications immediately after landing. However, the time required for Earth view immediately after landing is always greater than the length of actual data transmission. After landing, the Surface Laboratory has to perform many engineering tasks, such as antenna erection and gyrocompassing, deployment of sample gatherers, and checkout. We currently estimate that upwards of 1-1/4 hours will be required before high rate data can be transmitted. Although engineering data is transmitted continuously throughout this period, it is desirable to have landing sites which allow several hours of high rate data transmission after the high gain antenna is set up. Regions which allow a minimum of one hour of high rate data transmission before Earth-set are shown in Figure 2-12.

Launch date has a slight effect on the period of Earth view for a morning lander, Figure 2-13. For near equatorial landings which are 25° from the morning terminator, the period of Earth view decreases from approximately 6 hours to 5.5 hours from early to late launch date. The available sunlight after landing is of course considerably longer for the morning landing, Figure 2-14, than for an evening landing.

Dispersions in the landing site create time uncertainties for low rate radio link, and time and pointing uncertainties for the high rate radio link. The low rate link, which uses a low gain, virtually hemispherical coverage antenna, is affected only by the time in which the Earth is above the Martian horizon - the absolute value of the Earth's elevation angle is relatively unimportant. Unless the Surface Laboratory has the capability to determine its longitude, its programming must account for worst case landing site dispersions. The effects of dispersions on the low rate link, shown in Figure 2-15, cause a nominal loss of .5 hours in the available communications time. The high rate link, like the low rate, must account for dispersions by the reduction of transmitting time to the limits set by the dispersions. Since the high rate is dependent upon

FIRST DAY COMMUNICATIONS OPPORTUNITY

LANDING SITE LATITUDE	TIME TO EARTH SET (HOURS)	
	LAND 15° FROM MORNING TERMINATOR	LAND 30° FROM MORNING TERMINATOR
+50°	2.0 *	2.0 *
+40°	5.1 *	4.3
+30°	6.0	4.9
+20°	6.2	5.2
+10°	6.4	5.3
0°	6.4	5.3
-10°	6.3	5.1
-20°	6.0	4.7
-30°	5.4	4.2
-40°	4.3	2.7
-50°	2.7	0

1. No Dispersions
2. Earth Set at 34° Ground Slopes
3. All Evening Terminator Landings After Earth Set
* Landings Occur Before Earth Rise

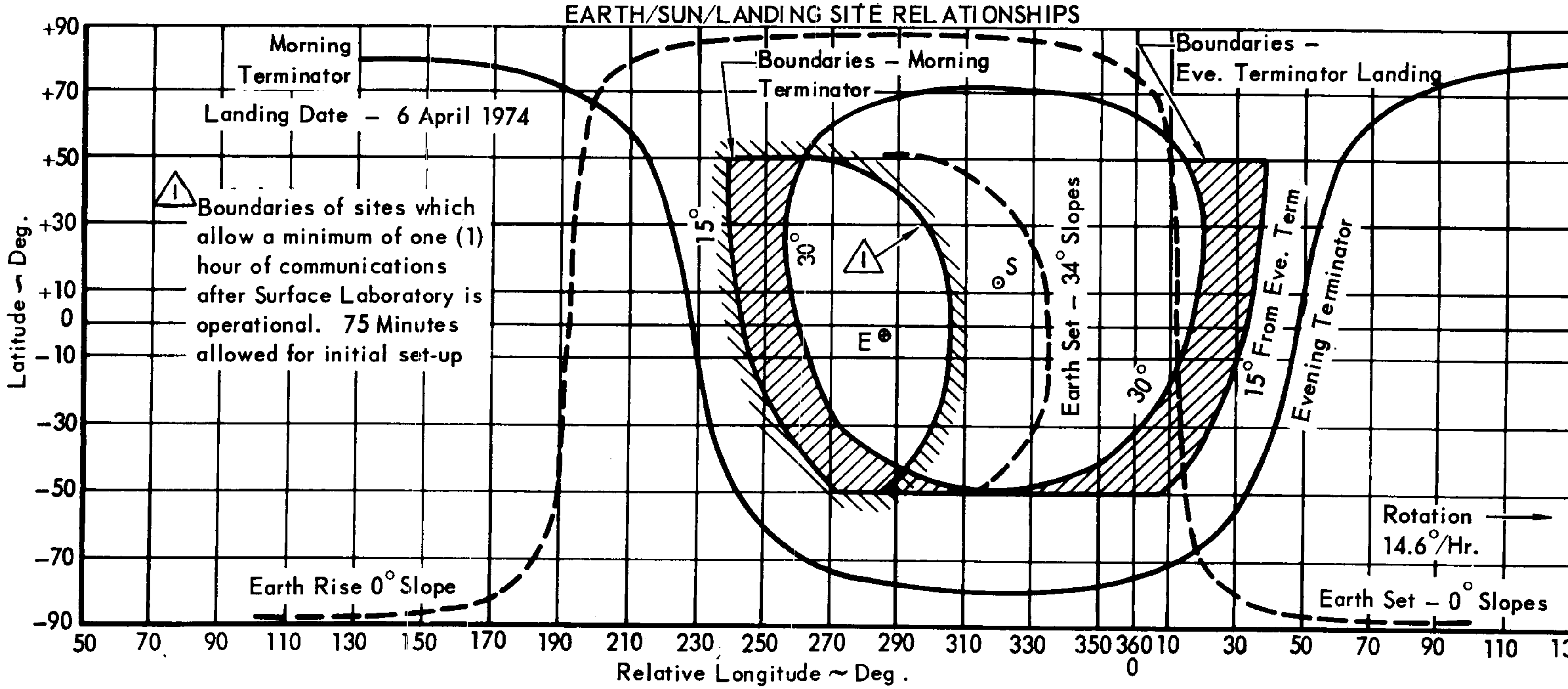


Figure 2-12

EARTH-SURFACE LABORATORY COMMUNICATIONS TIME
FIRST-DAY, DIRECT LINK
MORNING TERMINATOR LANDER

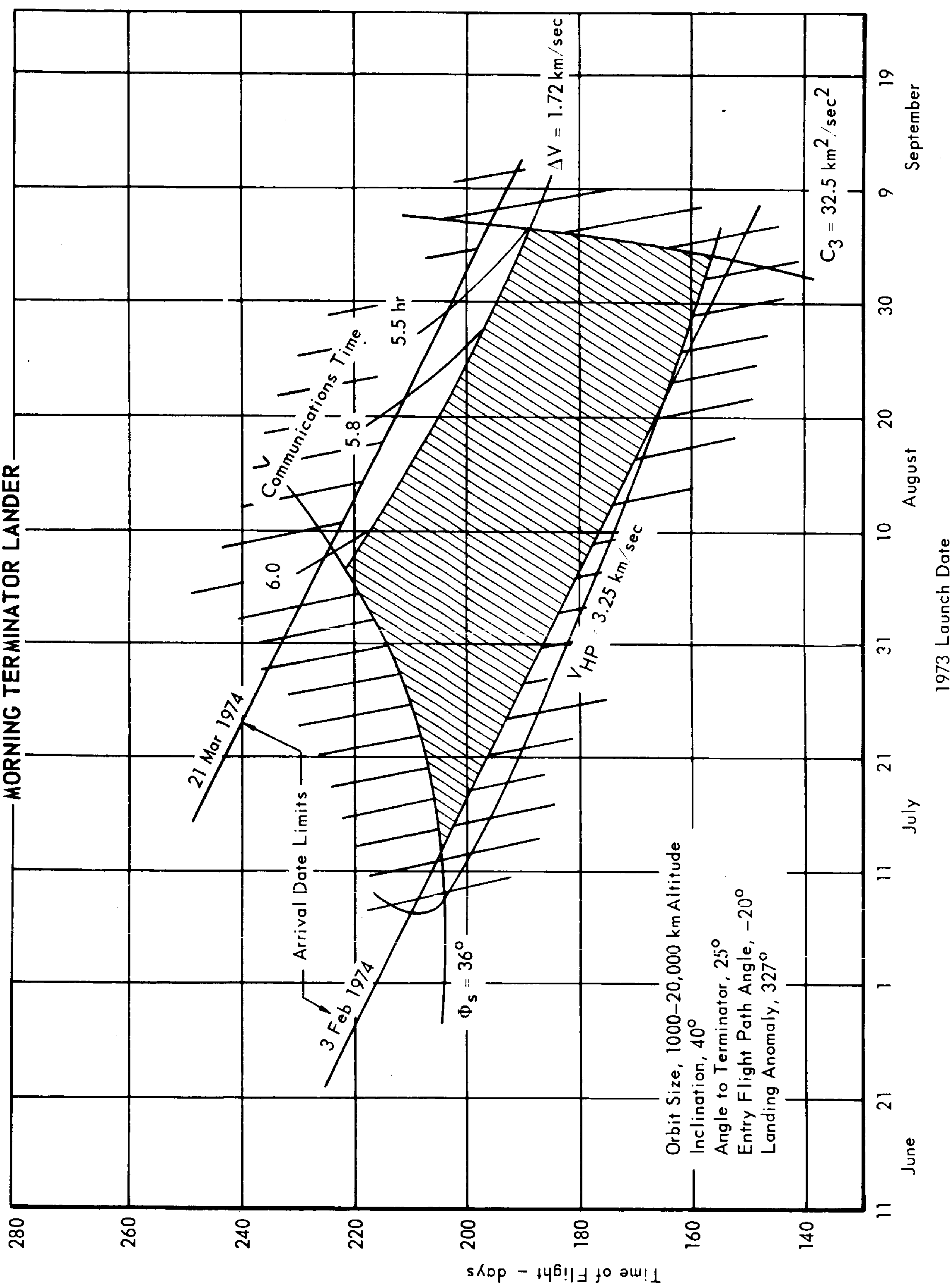
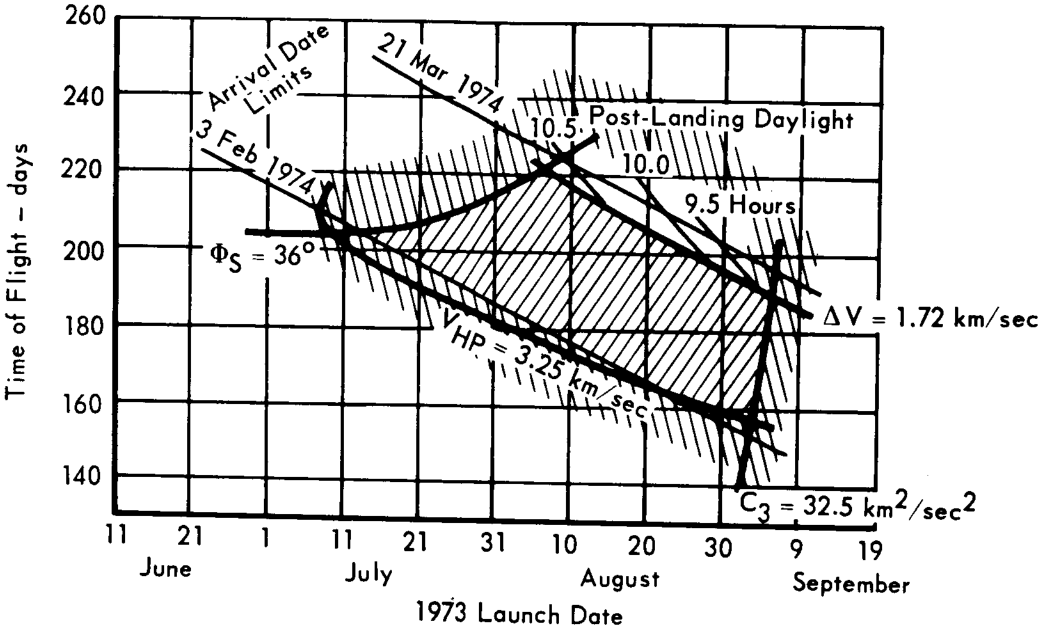


Figure 2-13

EFFECT OF MISSION PROFILE ON POST-LANDING DAYLIGHT

MORNING TERMINATOR LANDER



Orbit Size, 1000–20,000 km Altitude
Inclination, 40°
Angle to Terminator, 25°
Entry Flight Path Angle, –20°
Landing Anomaly, 327°

EVENING TERMINATOR LANDER

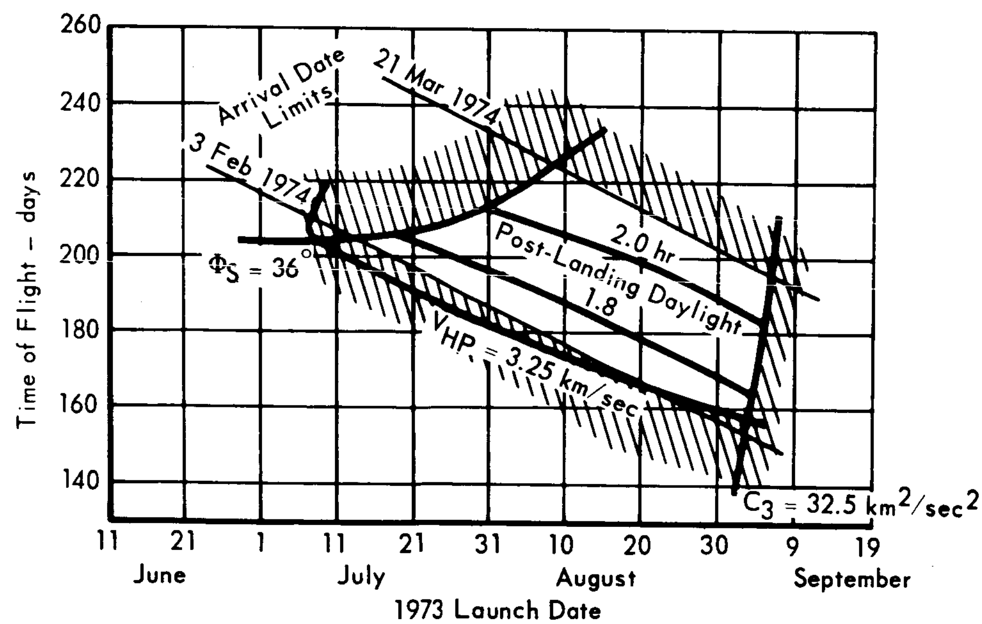


Figure 2–14

EFFECTS OF LANDING SITE DISPERSION IN LONGITUDE UPON COMMUNICATIONS

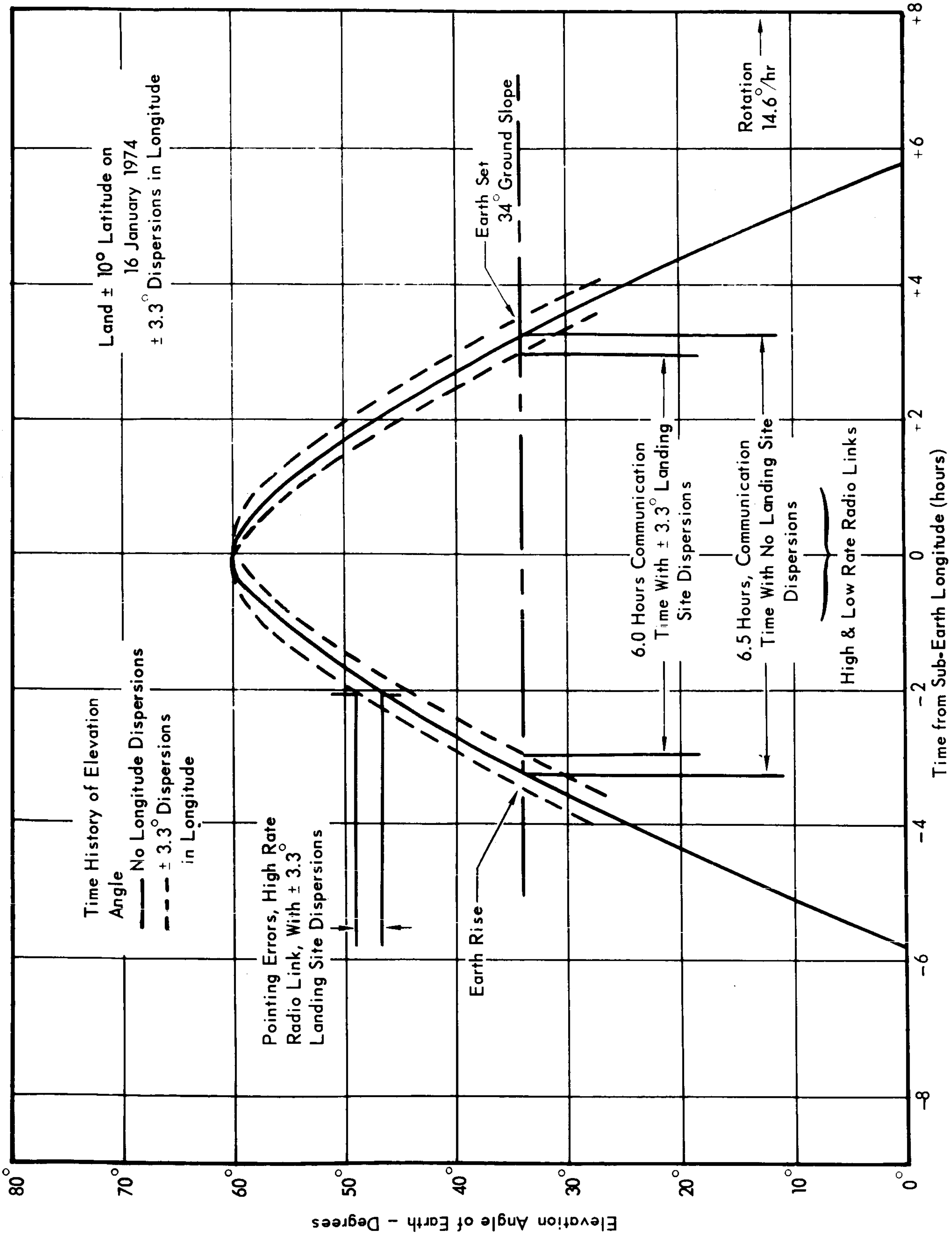


Figure 2–15

the position of the Earth, dispersions cause antenna pointing errors. For the dispersions shown in Figure 2-15, the worst case pointing errors are approximately ± 3.0°.

2.2.4 Recommendations for Landing Site Selection - For the 1973 VOYAGER, the selection of specific Martian sites for landing will be based on all available data at the time of final decision during flight. At the present time, particular sites are considered scientifically interesting; certain latitude bands are accessible from kinematic considerations of the approach, orbit, and descent trajectories; communications from the SL to Earth and descent TV influence the choice of time of day for the landing. Consequently, recommended sites must be further evaluated, up to and including the Planetary Vehicle overflight, prior to Capsule descent and landing.

Since the time of day for landing has a significant influence on the sites available, as well as on particular aspects of the Capsule Bus descent, the Surface Laboratory lifetime, and Spacecraft operation, this aspect will be discussed first. The descent experiments are considered highly desirable for the 1973 mission. Consequently the primary emphasis on selection has been on landing between 15° to 30° to the terminator for good descent TV. Primary aspects of the desired location relative to the terminator are discussed in Section II B2.3. Briefly summarized:

a. Spacecraft prefers evening terminator orientation, because of the rapid regression of the line of nodes with the small orbit desired for mapping.
b. Capsule prefers morning terminator landing, because it permits solar heating of the heat shield during descent, while maintaining a suitable attitude for descent communications.
c. Surface Laboratory prefers a morning landing, because it provides maximum data relay before the first nightfall and reduces total SL operating time.

It must be realized that although the preferences are contradictory, compromise solutions are feasible. With a larger orbit, the nodal regression is less rapid and an acceptable three month Spacecraft operation can be made with initial orbital orientation for a morning landing. For an evening lander, the heat shield of the Capsule can be protected from cold by the addition of a thermal blanket, although the landing will occur out of view of Earth. No data will be transmitted from the Surface Laboratory until the following morning. For data transmission of one complete diurnal cycle, landing near the evening terminator can require survival of two nights before transmission of data obtained in the late afternoon of the first complete day. The longer operating time increases the battery weight.

Another compromise is to land farther from the terminator toward the subsolar point. Such a forenoon lander provides a compromise orbit orientation for Spacecraft and Capsule operation, some first day transmission, and does not require survival of the SL for two nights. However, the shadows during descent will be shorter with a degradation of the descent images.

Landing areas which can be obtained for the approach, orbit, and descent conditions were discussed in Section 2.2.2. The longitude is adjustable by timing of the arrival and orbit stay time. The latitude for the morning terminator ranges from 10°N to 15°S with slight variations through the launch opportunity. The morning terminator landing has been selected (Section II B2.4) because of its favorable characteristics for the Capsule Bus and Surface Laboratory missions. Examination of scientifically interesting areas (Section 2.2.1) within this latitude band indicates that three sites (Syrtis Major, Auroae Sinus, and Tithonius Locus) react strongly to the wave of darkening.

At the present time, Syrtis Major is considered to be a suitable preferred landing site. This region covers a reasonably large area, reacts strongly to the wave of darkening, is within a microenvironmental locale, conforms to the kinematic considerations of landing locations, and is in the vicinity of a light area. Since the dark and light areas are the most obvious topographic features on Mars, they are the most likely candidates for initial exploration. Assuming that initial analysis of the Spacecraft data or the first Surface Laboratory data indicated a change in desirability from the dark area to a light area, the change in de-orbit would be minimized with the initial selection of Syrtis Major.

2.3 MISSION EVALUATION AND SELECTION - Within the framework of the individual considerations of science, communication, kinematics,and experiment sequence, a single preferred mission has been selected. The sequence of experiments, instrumentation, timing of data acquisition, storage and transmission, and power requirements were defined for the Surface Laboratory in Section 2.1. The selection of the landing site of the SL must be a compromise between desirable scientific areas and suitable communications, within the limits of the kinematic available sites.

Selection of the morning terminator landing was based primarily on the following: 1) Capsule Bus constraints on descent thermal control and descent communications, 2) the desire for no Sun or star occultation, and 3) the desire for Earth visibility of landing. Moreover, landing near the morning terminator will provide maximum experiment time before the first Martian night for SL data transmission.

Surface image data are effectively eliminated for the evening terminator landing until the following morning. This is a result of the higher (30° to 60°) lighting angle for the SL images as compared to the desired 15° to 30° lighting angle for the descent. Also, the power requirements (Section 5.3) of the SL are increased by an evening landing because of the longer operation time.

With the morning landing, the range of primary landing latitudes obtainable by pre-orbit insertion selections and de-orbit conditions is limited to approximately 10°N to 15°S latitude (Section 2.2.2). Within this band Syrtis Major was selected as a preferred landing site (Section 2.2.3) with Auroae Sinus and Tithonius Locus as alternates.

After orbit insertion, the band of landing latitudes is reduced,but stay time in orbit and de-orbit conditions permit longitude variation. To make use of this trajectory flexibility and also maximize utilization of available knowledge, the following technique for final landing site selection is considered. During the first few days after orbit insertion, the Planetary Vehicle is tracked for accurate determination of the orbit. During this time, the Spacecraft will be transmitting pictures of the Mars surface which will be evaluated on Earth for suitable SL landing sites. In addition, other Spacecraft experiments on surface emissivity, ultraviolet spectrometry, etc., will be made. Rapid analysis of these data should indicate gross terrain features - altitude differentials and general surface roughness. Assuming no startling difference from the hypothesized landing area, the pre-selected site would be utilized. If the initial analysis indicated the pre-selected site is unsuitable, a slight change in the descent trajectory and/or orbit stay time will move the landing location to a different site. The capability to update the program of the Capsule Bus sequence and timer is available at any time prior to Capsule-Spacecraft separation.

Additional flexibility in landing site selection is obtained through the use of the second (late arrival) Planetary Vehicle. Variables are arrival date, orbit size, location and inclination, orbit stay time, and descent trajectory.

The most difficult combination to obtain would be for the second Capsule to land at essentially the same site in a minimum time after the first landing. This combination has been determined as feasible for the preferred Capsule mission (Volume II A 4). This close repetition permits comparison of the data obtained from the two Capsules during descent and from the two Surface Laboratories with a minimum effect of landing site variation and time differential. A desirable alternative is to retain the second Capsule in orbit for several days, analyze the data from the first Capsule and Surface Laboratory and then select the landing site compatible with the kinematics of the specific orbit.

With the preferred landing site as Syrtis Major for both Capsules, with a landing near the morning terminator, the mission provides reasonable compromises for the primary considerations and retains sufficient flexibility for a wide range of changes based on all available data up to the point of Capsule-Spacecraft separation.

REFERENCES

2-1 Final Report, Study of the Automated Biological Laboratory Project Definition, Volume III, NASA CR-81307, 10 September 1965

2-2 CIT/SPL SE002BB001-1B21, Performance and Design Requirements for the 1973 VOYAGER Mission, General Specifications, 1 January 1967, Page 8

SECTION 3

SYSTEM FUNCTIONAL REQUIREMENTS

Functional requirements are imposed on the Surface Laboratory by the mission objectives. Additional requirements, such as providing a suitable environment and adequate electrical power, are derived from the Surface Laboratory itself. Because the latter interrelate with the laboratory design and operation, the determination of requirements is an iterative process.

Functional requirements can be separated into two principal categories, associated with the various phases of the mission profile. During the period from launch through landing, the Surface Laboratory must be kept in a condition which will allow initiation of the subsequent, more active phases. After landing the laboratory must prepare itself for operation. Finally, the operational, surface experiment, phase occurs.

The Surface Laboratory is carried aboard the Capsule Bus from launch until landing. Maintenance of suitable thermal control is the most important aspect of protecting the vehicle from the interplanetary environment. In addition, continuous monitoring to determine laboratory status is necessary. Electrical power to support these functions is required.

After landing, the initial requirement is preparation for the operation of the laboratory. Equipment which has been stowed, for its own protection or because of volume constraints, must be deployed. Some equipment may require leveling. Some subsystems, dormant before landing, must be turned on. Communication must be established and data transmitted. Thermal control must be maintained. Surface operations consist of conduct of the experiments and communication of data. The laboratory must provide the necessary power and sequencing. A listing of the most significant functional requirements and the candidate methods for satisfying each presented in Figure 3-1.

PREFERRED SURFACE LABORATORY SYSTEM

PERIOD	FUNCTIONAL REQUIREMENT	CANDIDATES	REFERENCE: SELECTION	REFERENCE: SUBSYSTEM CHARAC-TERISTICS	REFERENCE: FUNCTION DESCRIP-TION	REFERENCE: TEST PERFORMED
Launch Through Landing	Monitor System	None *Continuous–Passive Monitor Periodic – Active Monitor Periodic – Passive Monitor	B4.4	A3.1	C7.1	
	Initiate Battery Charge	*Flight Spacecraft Sequencer Surface Laboratory Timer	B5.2	A3.3.2	C2.1	
	Maintain Thermal Control	None Surface Lab. Passive *Heaters, Coatings, Insulation	B5.8	A3.3.8	C 13	VI B3.1.1
	Survive Entry and Landing	*Capsule Bus Energy Attenuation Surface Laboratory Energy Attenuation Equipment Energy Attenuators	II B4.2	II A3.2.1	II C1.4	
Surface Operations	Sense Local Vertical	*Local Vertical Sensor Stellar Sensor Solar Sensor Gyro Compass None	B4.2	A3.2.4	C 10	
	Erect Equipment	*Mechanical Extensors Pneumatic Extensors Fluid Suspension Mechanical Gimballing None	B5.9.3	A3.3.5	C 10	

*Preferred Design

Figure 3-1

PREFERRED SURFACE LABORATORY SYSTEM (Cont.)

PERIOD	FUNCTIONAL REQUIREMENT	CANDIDATES	REFERENCE			
			SELECTION	SUBSYSTEM CHARAC-TERISTICS	FUNCTION DESCRIP-TION	TEST PERFORMED
Surface Operations (Cont.)	Maintain Communications Communication Concept	S-Band Transmitter 1–Way UHF Relay Transmitter *S-Band Transceiver	B5.4.7	A 3.3.4	C 4	VI B 1.2.5 VI B 1.2.8
	Antenna Mounting High Gain S-Band	Fixed Azimuth-Elevation Directable *Directable-Equatorial Mount	B5.4.6	A 3.3.5	C 5.1	VI B 1.2.5
	Low Gain S-Band	*Surface Laboratory – Mast Mounted Surface Laboratory – Surface Mounted None	B5.4.6	A 3.3.5	C 5.2	VI B 1.2.5
	Erect Antennas Low and High Gain	Local Vertical *Lander Axis None	B5.4.6	A 3.3.5	C 5	
	Transmit Data	*1.0 Bits/Sec Low Gain Antenna 10 Bits/Sec 50 Bits/Sec	B5.9.1	A3.3.4	C7.1	VI B 1.2.5
		*445.2 Bits/Sec High Gain Antenna 1000 Bits/Sec	B5.9.1	A3.3.4	C7.1	VI B 1.2.5
	Maintain Thermal Control Heating	*Electrical Heaters, Insulation Chemical Heaters, Insulation Isotopic Heaters, Insulation	B5.8	A3.3.8	C13	

*Preferred Design

Figure 3–1 (Continued)

PREFERRED SURFACE LABORATORY SYSTEM (Cont.)

PERIOD	FUNCTIONAL REQUIREMENT	CANDIDATES	REFERENCE			
			SELECTION	SUBSYSTEM CHARAC-TERISTICS	FUNCTION DESCRIP-TION	TEST PERFORMED
Surface Operations (Cont.)	Cooling	Hinged Insulation Panels Heat Sink Water Evaporator *Heat Pipes, Radiators None Pump and Coolant, Radiators	B5.8	A3.3.8	C13	VIB 3.1.1
	Perform Experiments	*Yes, Preferred Instrument Payload No Yes, Alternate Instrument Payload	B5.9	A3.3.9	C14	
	Operate on Surface	36 Hours (min); 50 hours (with 3σ Dispersions) *28 Hr. for Morning Terminator Landing or 40 Hr. for Evening Terminator Landing Other	B23	A4.1	C4.3.4	
	Acquire Samples	Hoe Shovel *Auger Brush	B5.9.2	A3.3.9	C14.2	
	Provide Power	*Surface Laboratory Batteries Fuel Cells RTG Array of Solar Cells RTG and Batteries	B5.1	A3.3.1	C1	

*Preferred Design

FIGURE 3-1 (Continued)

SECTION 4

MAJOR TRADE STUDIES AND SYSTEMS ANALYSES

In the development of the Surface Laboratory System design major trade studies and systems analyses were a direct evolutionary step following the establishment of system functional requirements.

Having determined these system requirements from the mission profile and the basic approach to achieving the mission goal, the logical process required the establishment of major system boundaries and the optimization of decisions affecting the complete system.

These studies involved both trade-offs in the bounding parameters including (weight, power and communications capability) and the refinement of basic information leading to system and subsystem development. Both areas of study were equally important in defining an optimum system.

A number of the studies ran concurrently. Therefore, the process of developing data and making decisions was largely iterative. Constant reexamination was required to insure an updated basis for making decisions. Similarly, as the major studies progressed they affected and imposed constraints on the subsystems. The resulting subsystem efforts, in turn, provided data which usually had an effect on the establishment of system parameters (payload is a typical example). A continuously updated baseline was maintained and was a most valuable tool in implementing this iterative process.

An important factor in the system development was the interrelationship of the Surface Laboratory with the Capsule Bus. Much of the analysis for the two systems was done with the two as inseparable entities.

One system effort, the Extended Mission Study, is special purpose in nature. This study was undertaken, not so much to help define the baseline system, but rather to examine the benefits to be derived and the attendant cost, of a longer mission for the Surface Laboratory.

4.1 SCIENCE INTEGRATION - Performance of the Surface Laboratory experiments requires a careful analysis of the design and operational interfaces of the science experiments with other capsule subsystems. Integration of the science subsystem has been treated in our Phase B studies in the selection of the Capsule subsystems and by system studies of several potential science integration problem areas. These science integration studies have allowed us to define and evaluate problem areas and, where possible, determine solutions consistent with the entire system objectives. The analyses and results of these studies are reported in this section.

For our Phase B studies, the following major problem areas were defined and evaluated: 1) landing site surface contamination; 2) nuclear radiation generated by Surface Laboratory subsystems; 3) mechanical integration of the science instruments into the Surface Laboratory; 4) integration of the instruments with supporting electronic subsystems and 5) interfacing the sequencing of science operations with total mission operations.

Surface heating, erosion, and chemical deposition by the terminal propulsion subsystem are the primary sources of landing site contaminations. Our evaluation indicates the water deposited by the engines could negate a major scientific objective - the measurement of natural water concentration in the Martian surface.

The possible use of radioisotopes in the Surface Laboratory required an evaluation of possible degrading effects on science instruments using radiation counting techniques, such as the metabolic life detector. While radioisotopes are not included in the preferred 1973 approach, careful design and shielding will be required when large radioisotopic sources, such as an RTG, or used.

Mechanical integration studies treated the conflicting requirements of the science instruments, their support subsystems, and the total vehicle limitations. Of particular concern were field of view, placement in the Surface Laboratory, and thermal control. The preferred Surface Laboratory design reflects the results of these studies.

Integration of the science instruments with the supporting electronic subsystems is complicated by the desire to standardize the subsystems and yet provide the flexibility to accept changes in the science instruments. Analysis has shown that a Science Data Subsystem (SDS) composed of discrete modules (Remote Interface Units) - one for each science instrument - permits standardization of other electronic subsystems and provides the necessary flexibility to adapt to instrument changes.

The sequencing of the science operations must be consistent with the total mission operations. This sequencing has been resolved by provisions for adaptive control of the experiments and landing site flexibility.

4.1.1 Surface Contamination by Terminal Descent Engines - The use of a propulsive terminal descent mode presents a possible compromise to some scientific objectives of the VOYAGER mission by perturbing the natural surface environment at the landing site. This is of particular concern to surface composition measurements and life detection experiments. A preliminary analysis of the interaction with the surface of the exhaust plume from a single engine of the preferred propulsion subsystem design has identified some problem areas and indicated the need for more detailed analysis and testing to derive those constraints on both the science experiments and terminal propulsion subsystem which minimize the problems.

4.1.1.1 Plume Model - The most severe interaction of the exhaust plume with the surface occurs during the final 50 feet of descent with no surface wind. This descent was assumed to occur at a constant velocity of five feet/second normal to the surface, with the terminal descent propulsion engines at deep throttle and the exit pressure at the nozzle 1.67 psia (115 mb). These values are applicable to the preferred Capsule Bus design except that terminal descent is at a constant velocity of 10 feet/second. Therefore, the following analysis is conservative for the preferred design.

For the range of VM atmospheres, the surface pressure varies from 5 to 20 mb. Therefore, during the final 50 feet of descent the engine exhaust plume is underexpanded. Downstream of the nozzle exit the exhaust gases continue to expand radially from the nozzle centerline. The exhaust plume from a single engine exhausting into a vacuum was used as an approximation to the actual case of exhaust into the Martian atmosphere. This model, illustrated in Figure 4.1-1, is used in the following analysis of surface contamination.

4.1.1.2 Chemical Contamination - The mass flux from the engine to the surface was integrated for the final 50 feet of descent, assuming an engine cutoff altitude of 10 feet. The result is shown in Figure 4.1-2 with surface contamination in terms of mass per unit area plotted as a function of radial distance from the nozzle centerline. This distribution assumes unity absorption of all plume exhaust products and ignores nozzle ablation products, physical absorption effects on the surface, and reactions in the atmosphere.

SURFACE CONTAMINATION - PLUME MODEL
JET EXHAUST EXPANDING INTO A VACUUM - FAR FIELD

Chamber Pressure = 30 PSIA
Chamber Temperature = 5366 °R
Ratio of Specific Heats = 1.25
Molecular Weight = 22.15
Nozzle Exit Radius Re = 4.42 in.

Exhaust Gas Constituents

SPECIE	MOLE FRACTION
CO	.04897
CO_2	.08515
H	.00001
H_2	.12581
H_2O	.39394
N_2	.34612

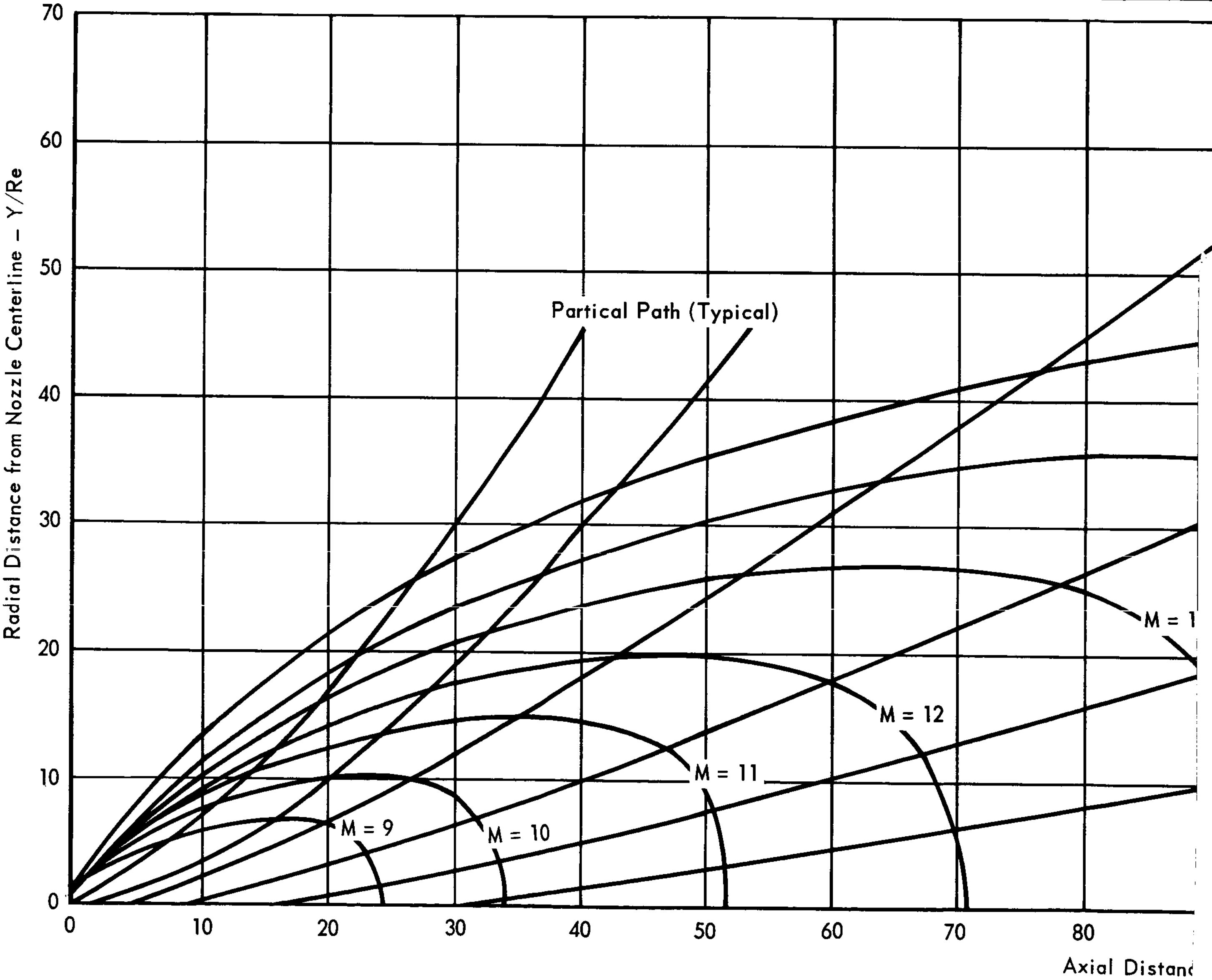
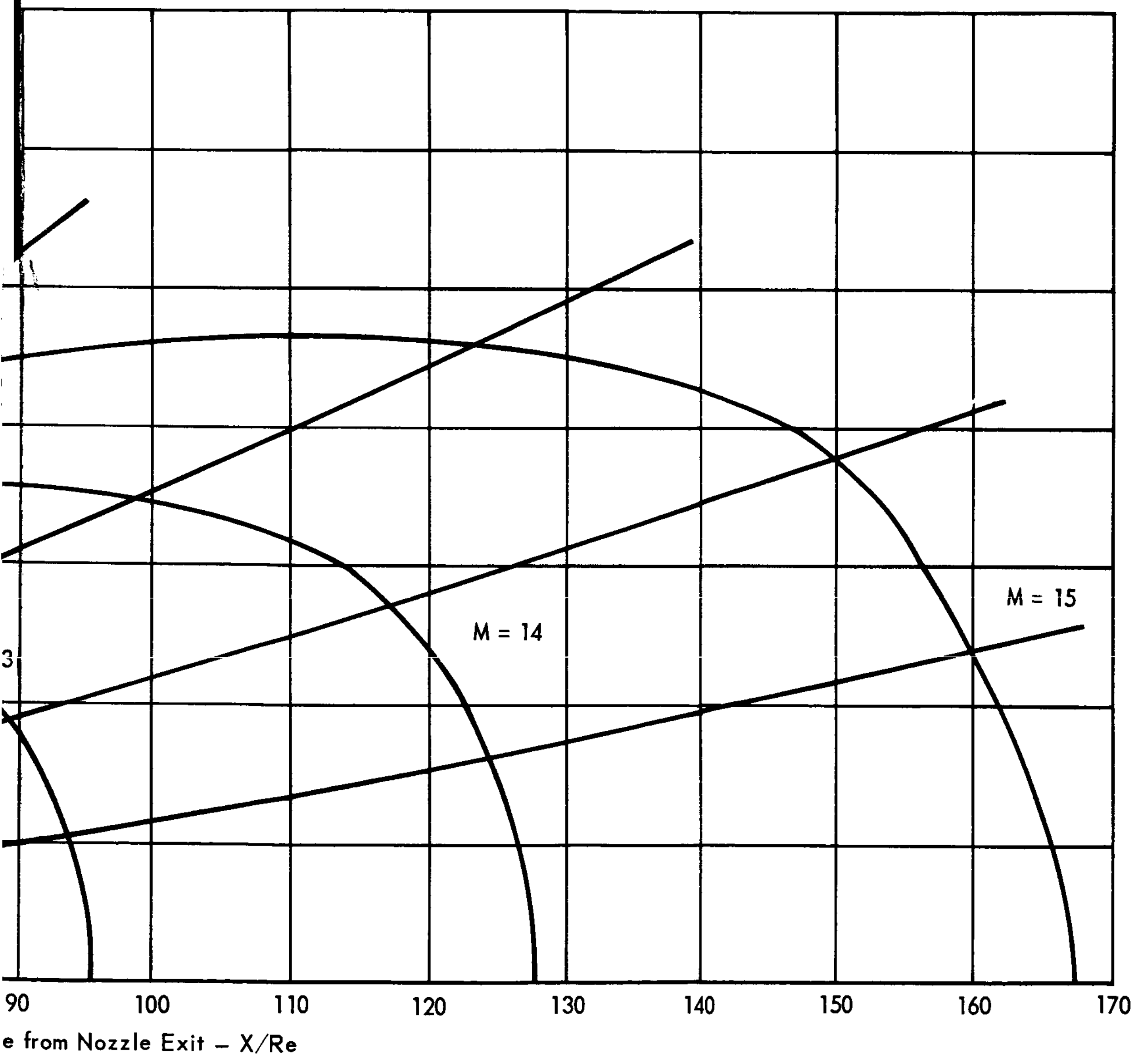


Figure 4.1–1

Static Properties Corresponding to ISO-Mach Contours

M	T - °R	P - psia	ρ - lbm/ft^3
6	977	6.00×10^{-3}	1.26×10^{-5}
7	754	1.63×10^{-3}	4.48×10^{-6}
8	597	5.10×10^{-4}	1.76×10^{-6}
9	483	1.77×10^{-4}	7.55×10^{-7}
10	398	6.69×10^{-5}	3.48×10^{-7}
11	333	2.74×10^{-5}	1.70×10^{-7}
12	283	1.21×10^{-5}	8.86×10^{-8}
13	243	5.67×10^{-6}	4.82×10^{-8}
14	205	2.77×10^{-6}	2.71×10^{-8}
15	185	1.43×10^{-6}	1.62×10^{-8}



SURFACE CONTAMINATION AS A FUNCTION OF RADIAL DISTANCE FROM THE NOZZLE CENTERLINE. – ENGINE CUT OFF ALTITUDE 10'

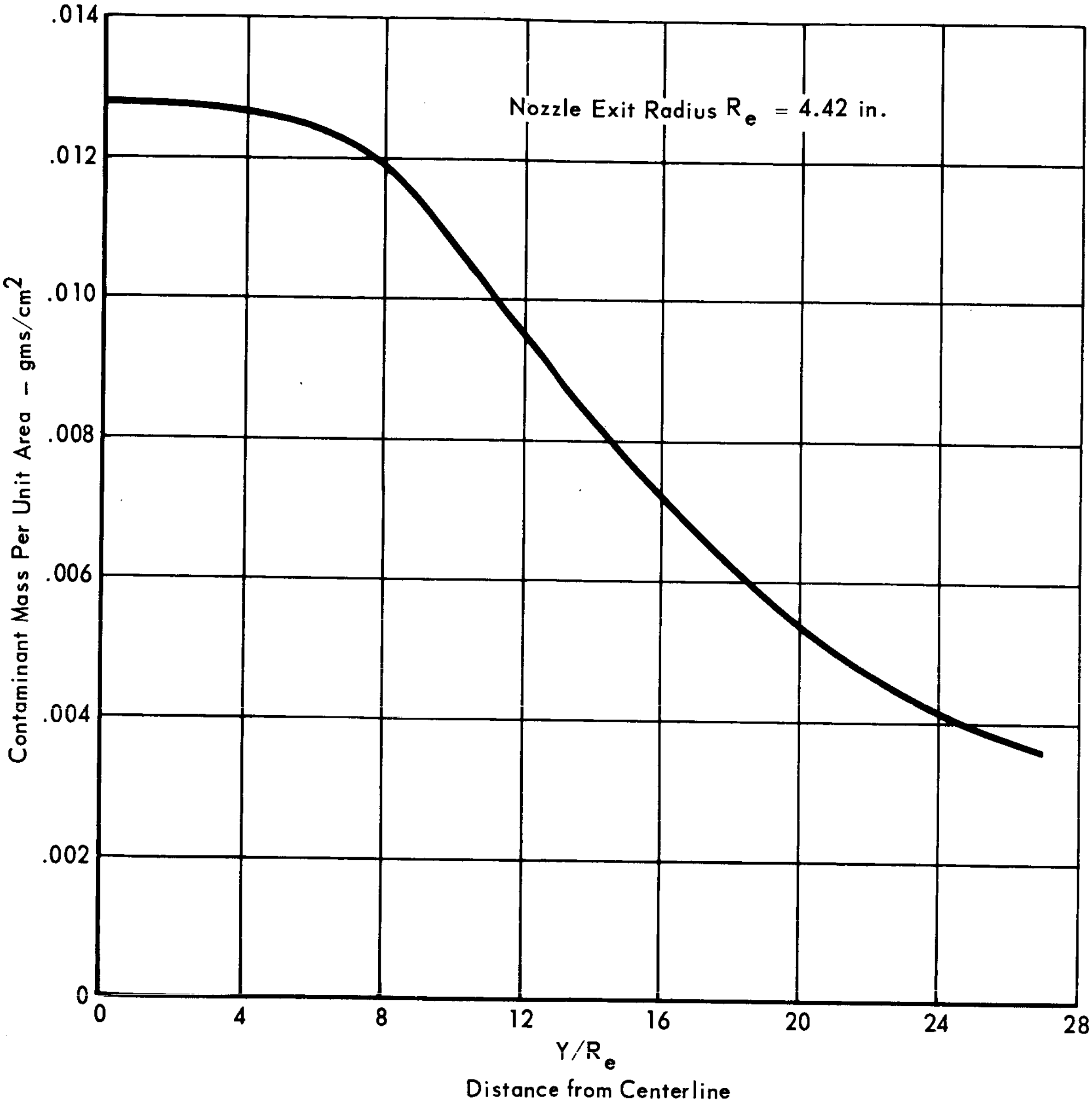


Figure 4.1–2

A more detailed examination of the interaction with the surface assumed a VM-8 atmosphere and frozen chemistry in the plume. In this case the composition of the plume for the nitrogen tetroxide (N_2O_4) - monomethylhydrazine (MMH) engine is given in the figure below.

EXHAUST GAS CONSTITUENTS

Mol	Fraction	Molecular Wt.	Product	Weight Percent
CO	.04897	28	1.37	6.19
CO_2	.08515	44	3.75	16.93
H	.00001	1	0.00	0.00
H_2	.12581	2	0.25	1.13
H_2O	.39394	18	7.09	32.01
N_2	.34612	28	9.69	43.74
			22.15	100.00

The reaction of combustion products with an atmosphere must consider three factors:

- An atmospheric dilution effect.
- Lowered temperatures caused by the dilution.
- True chemical changes induced by the atmosphere.

The initial factor can be normalized out; the others are interdependent and more difficult to portray. For the VM-8 model atmosphere and assuming thermo-chemical equilibrium is attained, the atmospheric environment does not become saturated with CO_2. Since the vapor pressures of CO, H_2, and N_2 are zero, these species will be dispersed in the gaseous phase. Minor amounts, negligible for the purposes of this study, may be trapped as discontinuous monolayers absorbed on silicate surfaces. Therefore, the surface contaminant will be water alone in this case.

Some water vapor will condense from the plume and be deposited on the surface of Mars as the result of the transient pressure wave impinging on the surface, permitting localized atmospheric saturation. The condensation phase should be followed by a light blanket of snow. The deposition of water will be closely related to the character of the surface in terms of roughness, porosity, grain size, and mineralogical composition. The character of permanently abosrbed layers on selected grain surfaces will also enter the condensation and gas absorption picture. Three simplified surface models were used to demonstrate the relation of the contaminants to the Mars environment and to the experiments and sensors onboard the Surface Laboratory. The figure below designates these models and their significant physical characteristics (Reference 4.1-6).

SOIL MODELS

Model	Mean Grain Size, μ	Porosity	Cold Trap Factor, K,	Surface Area, cm^2/cm^3
Dense bedrock	NA	0	0	1
Dune Sand	330	48	.947	182
Loess (silt)	20	28	.999	1370

For communications, biological experimentation, and observation reasons, early VOYAGER missions should be timed and directed to observe the following geometry:

- o Landing latitude within 25° at the subsolar latitude.
- o Dark area landing site.
- o Summer season (plus 0°C for part of day - no clouds).
- o Landing timed for early morning.

These factors are important to the following analysis (Reference 4.1-7).

Water vapor molecules impinging on a surface either (1) are reflected to escape, (2) reimpact, or (3) are trapped by condensation according to an accommodation coefficient which may vary from zero to one but has been experimentally verified at a value of 0.94. Under reduced pressures, condensation occurs almost instantaneously (10^{-4} to 10^{-5} seconds). As the temperature rises above the value for an equilibrium vapor pressure, sublimation sets in. For a smooth uniform surface, such as the bedrock model, the maximum sublimation rate is given by:

$$m_s = P\sqrt{\frac{\mu}{2\pi RT}}$$

Where: m_s = surface mass loss rate by sublimation, $gm/cm^2/second$

P = vapor pressure, $dynes/cm^2$

μ = gram molecular weight, 18

R = gas constant

T = temperature, °K

For a porous system, a series of microscopic cold traps are present and the solar thermal flux is less effective. Thus, the net mass loss becomes:

$$m_n = K\,(m_s - m_c)$$

Where: m_n = net subsurface mass loss rate, gm/cm^2/second.

K = cold trap factor, fraction of surface area shaded.

$(m_s - m_c)$ = difference between sublimation and condensation rates, gm/cm^2/second.

The condensation rate, m_c, is related to three factors: the accommodation coefficient, the probability of escaping a cold trap for a single molecule, and the cold trap factor, as follows:

$$m_c = \gamma\,\alpha m_s$$

Where: $$\gamma = \frac{Ka(1-\alpha)}{Ka(1-\alpha) + \alpha}$$

and m_c = condensation rate, gm/cm^2/second

a = fraction of molecules trapped versus those escaping

α = probability of a molecule escaping cold trap in a single jump, 4×10^{-3}

A summary of the mass loss rate data is depicted in Figure 4.1-3. In order to construct these tables it was necessary to briefly investigate thermal flux parameters for the engine jet (see Section 4.1.1.2) and the solar constant within the assumed latitude range of 25° of the subsolar point. The following estimates are pertinent:

- Solar flux during pre-noon hours:

Solar flux	78 cal/cm^2
Latent heat of sublimation	-2
	76 cal/cm^2

- Vernier engine flux (single engine) 3.4

Latent heat of vaporization	7.0
Latent heat of fussion	1.0
	11.4 cal/cm^2

SUMMARY OF TOTAL CONTAMINATE DISSIPATION TIME

MASS OF CONDENSED WATER, (gm/cm^2)

MODEL	SURFACE LAYER	COLD TRAP FILMS
Rock	0.0042	0.0
Sand	0.0022	0.002
Silt	0.003	0.0012

DISSIPATION TIME

MODEL	T, °K	LOSS RATES m_s	m_n	DISSIPATION TIMES t^s	t^h	t^d
Rock	300	0.38		0.01		
	225	6.19×10^{-4}		6.8		
	168	1.006×10^{-7}				0.5
Sand	300	0.38		0.01		
	225	6.19×10^{-4}		3.5		
	168	1.006×10^{-7}				0.32
	Cold Trap		4.26×10^{-10}			53.0
Silt	300	0.38		0.01		
	225	6.19×10^{-4}		4.85		
	168	1.006×10^{-7}			8.4	
	Cold Trap		3.996×10^{-10}			34.0

Explanation:

m_s = surface sublimation mass loss rate, $gm/cm^2/sec$

m_n = net subsurface mass loss rate, $gm/cm^2/sec$

t^s = time, sec

t^h = time, hours

t^d = time, days (Mars)

Figure 4.1-3

When these thermal flux values were applied to the sand and silt models, the maximum diurnal variation was damped out at 5 cm depth and the vernier engine flux at 1 cm depth. Hence, the concept of porosity-created cold traps appears to be valid for the Mars case. In turn, outgassing of condensation products trapped in these voids would proceed slowly.

For the purposes of this preliminary analysis, cold traps were defined as shadowed intergranular voids where condensation products were trapped as the result of transient jet flow overpressure. Figure 4.1-3 summarizes the rate at which trapped molecules are released. The physical parameters utilized to develop the data in this figure are outlined in the figure above titled "Soil Models." Based on the porosity and the surface area per cubic centimeter shown in the table, the depth to which a 100 mole thickness of ice can condense is approximately 14 cm for the sand model and 1.0 cm for silt. These differences in vapor penetration depth are reflected in the dispersal times noted in Figure 4.1-3.

These preliminary data are applicable to worst case conditions at the nozzle centerline. To approximate conditions at a radial distance of ten feet from the nozzle centerline, the contamination can be reduced by a factor of four.

Future efforts should be directed toward sharpening the application of these concepts, better approximating the temperature distribution with atmospheric dilution, estimating the absorption of gases on silicate surfaces, and introducing additional atmospheric and surface models, e.g., soluble salts duricrust model.

In addition, the degree and type of contamination due to nozzle ablative lining erosion should be determined. Also, the true chemistry of the plume should be examined to determine the possibility of formation of other than equilibrium products. However, it is felt that further analysis should be combined with a test program to aid in directing the analytical effort.

4.1.1.3 Thermal Effects - The heat flux imparted to the surface by the descent engines can affect the viability of the uppermost layer of the surface or induce chemical changes (e.g., dehydration). To evaluate this potential problem a simple, conservative analysis of surface heating using the single engine model was made.

The maximum incident heat flux was estimated assuming the total enthalpy of the jet is imparted to the surface. Since the jet total temperature is large compared to the Martian surface temperature, an average initial surface temperature of 0°F was assumed (nominal for normal diurnal variation and early morning landing). The incident heat flux to the surface as a function of radial distance from the nozzle centerline and altitude above the surface is presented in Figure 4.1-4. The heat flux values were integrated with the terminal descent velocity profile (5 fps constant descent velocity, normal to surface) to determine the resulting temperature profile during the final 50 feet of descent. This profile is depicted in Figure 4.1-5 where the Martian soil has been characterized by:

$$\rho = \text{density} = 124.8\ \text{lbm/foot}^3$$

$$C_p = \text{specific heat} = 0.15\ \text{Btu/lbm°R}$$

$$k = \text{thermal conductivity} = 0.048\ \frac{\text{Btu-foot}}{\text{hour-foot}^2\text{-°F}}$$

The temperature profile of Figure 4.1-5 does not appear to be a problem from the scientific mission standpoint. At the nominal ranges at which surface samples are to be collected (four to six feet from an engine centerline), the temperature perturbation does not exceed the maximum due to diurnal solar heating. Therefore, plume heating is not likely to affect adversely the nature of these soil samples.

It is emphasized that the above analysis was preliminary and needs refinement considering the following factors: multiple engine configuration; plume-in-atmosphere model; plume-surface flow field; heat transfer methods; and a range of soil models.

4.1.1.4 Surface Contamination - Particle Transport - A strong interaction of the plume with the surface is expected as the deep throttled engine nears the surface. The exhaust gases will accelerate radially from a stagnation region on the nozzle centerline to supersonic velocity tangent to the surface. Preliminary estimates indicate that the dynamic pressure of the tangential flow will exceed the shear strength of the surface, resulting in erosion. The extent of the erosion is dependent on the type of surface, i.e., size, cohesion, and depth of particle layers. Such erosion will affect experiments contingent on obtaining representative samples of the top surface. The design of in-situ instrument deployment and sample gathering equipment depends on how far from the touchdown point it is necessary to go to collect surface samples unaffected by erosion fallout coatings. Terminal descent viewing may also be affected by erosion debris.

SURFACE CONTAMINATION – THERMAL EFFECTS

Incident Heat Flux to Surface as a Function of Radial Distance from the Nozzle Centerline

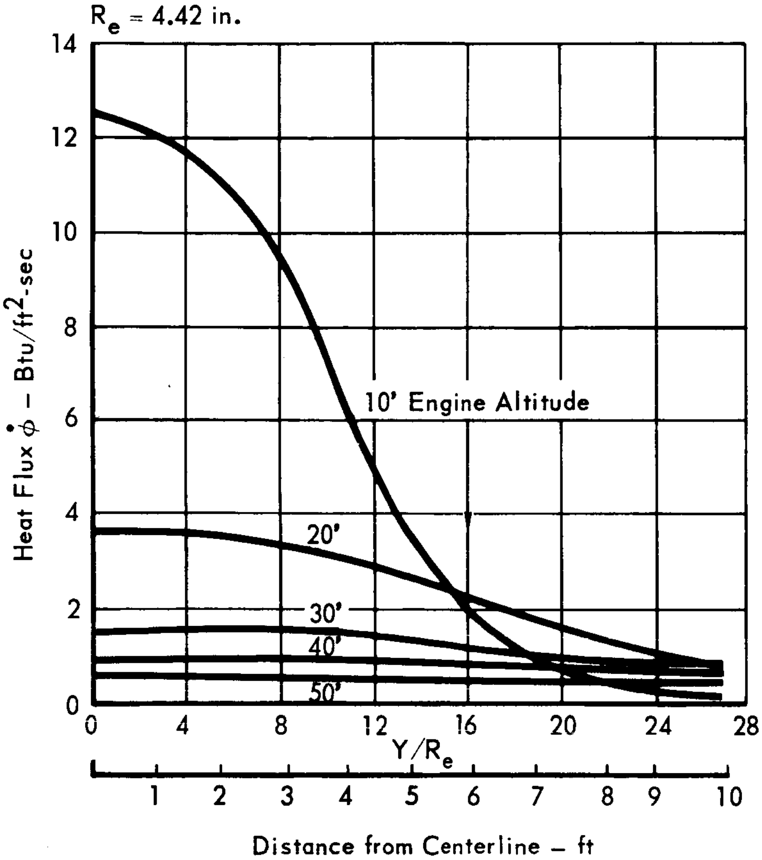


Figure 4.1–4

SURFACE CONTAMINATION – THERMAL EFFECTS

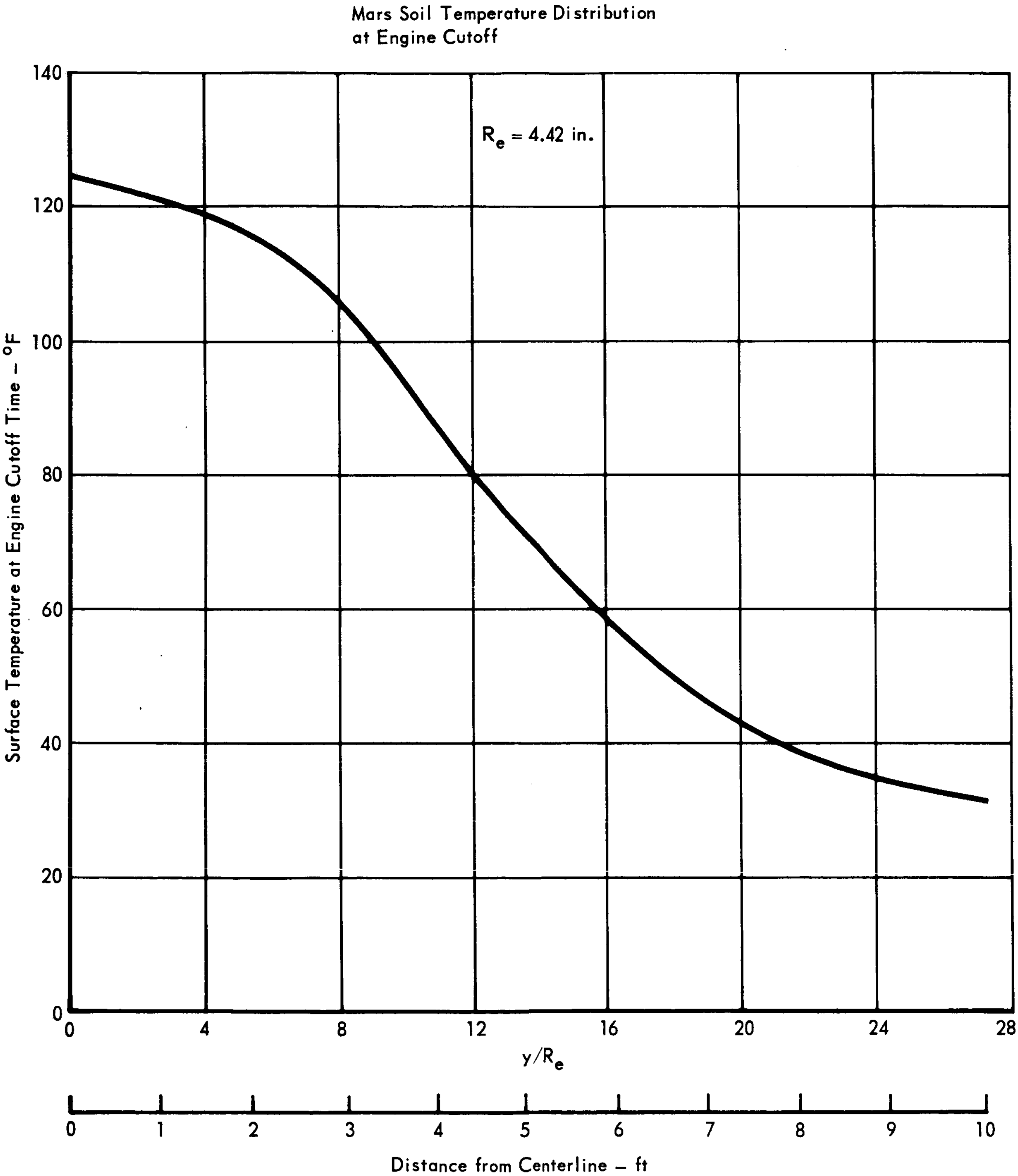


Figure 4.1–5

An extremely simplified estimation of the eroded particle transport is made using the characteristics of the engine jet at the cutoff height. The maximum range depends on particle size that the particles are accelerated almost linearly with distance from an area near the nozzle centerline to some maximum velocity which is a fraction of the gas velocity.

Particles larger than 1000μ diameter will essentially follow ballistic trajectories from the jet. Particles less than 1000μ diameter will experience local atmosphere drag forces which limit their ballistic range. These particles are subject to entrainment in local winds.

Figure 4.1-6 shows the estimated maximum horizontal transport of surface material from the landing point, as a function of size. The initial particle-to-gas velocity ratios were calculated from those given by the reference indicated on the figure. A gas velocity of 3000 feet/second was assumed.

4.1.1.5 Integration Assessment - Contamination is a problem relative to the scientific objectives and operational methods of the VOYAGER system. With the use of a roving Surface Laboratory, any experimentation that might be affected by contamination could be performed outside of the contaminated area. However, for the 1973 mission (with a stationary Surface Laboratory), the possibility of water deposition on the surface represents a serious problem since one of its main objectives will be to determine the amount of water in the Martian surface environment. This question is of utmost importance to evaluate the surface as a potential abode for life and should be answered early to adequately plan the post-1973 mission objectives. The analysis carried out in this section indicates that the amount of water condensed from the plume can exceed the amount estimated for the Martian surface and that the dissipation time for this contamination can exceed the lifetime of the 1973 Surface Laboratory.

The possibilities for avoiding the contamination are remote sampling, subsurface sampling, and alternate terminal descent modes. A trade study of these alternatives is not warranted at this time due to the preliminary nature of the contamination analysis. The problem has been recognized, however, in the preferred capsule system design and first order solutions have been incorporated in the design. In the trade study of the Capsule Bus terminal descent engine configuration (Volume II, Part B, Paragraph 4.3), centerline contamination has been used as a parameter. The Surface Laboratory sample acquisition equipment is a shallow subsurface auger which rejects the top few millimeters of soil.

MAXIMUM RANGE OF SURFACE PARTICLES TRANSPORTED BY ROCKET PLUME INTERACTION

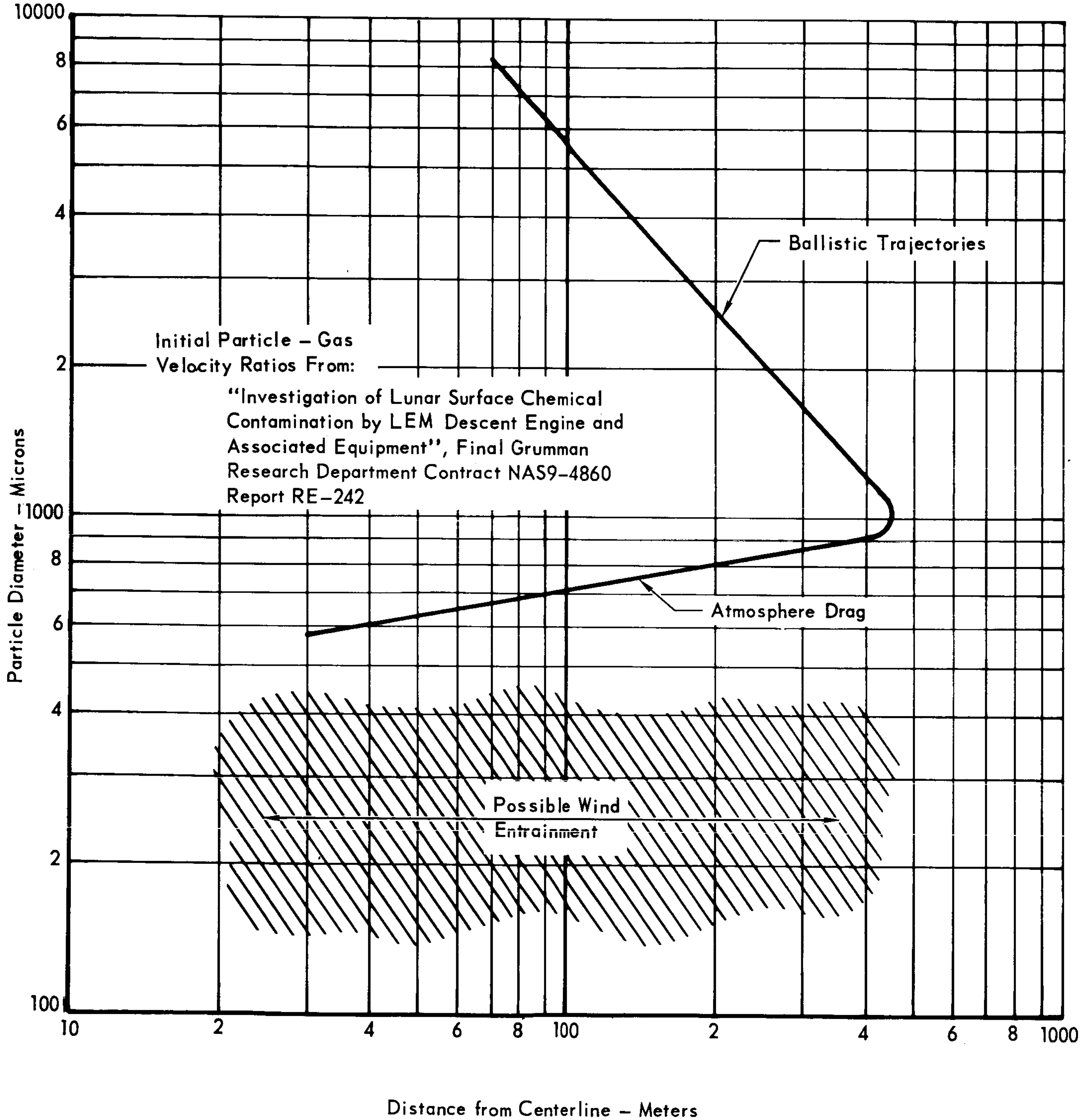


Figure 4.1-6

4.1.9 Radiation - The use of nuclear materials in Surface Laboratory subsystems can effect science instrument operation through the induced nuclear radiation environment. This radiation can degrade or damage materials used in the instruments and can directly interfere with those instruments utilizing radiation measurement techniques. Two such nuclear sources are the Radioisotopic Thermoelectric Generator (RTG) used for electrical power generation and the radioisotope heater, a candidate for thermal control heat generation.

The RTG is the prime candidate as an alternate electrical power source for long term Surface Laboratory missions in 1973 and as the primary power source for post-1973 missions. The factors involved in the selection of an RTG fuel and RTG configuration are discussed in Section 5.1. The following analyses are limited to RTG's fueled with Pu^{238} in the form of $Pu\ O_2$.

Radioisotope heaters are attractive heat sources for localized thermal control. A brief discussion of the nuclear environment from these devices follows the RTG discussion.

4.1.2.1 RTG Nuclear Environment - Neutron Radiation - Neutrons are emitted from Pu^{238}-fueled RTG's from spontaneous and induced fissions and from (α , n) reactions.

Spontaneous Fissions - The spontaneous fission half-life of Pu^{238} is 5×10^{10} years, giving a yield of 3.4×10^3 neutrons per second per gram of plutonium product (Reference 4.1-1). This number is, of course, independent of the fuel mass present and of its geometry.

Induced Fissions - Plutonium-238 is fissionable by neutrons with energies ranging from over 10 Mev down to thermal (20°C) neutrons. This aspect makes the neutron yield dependent upon the mass of fuel present, and upon its effective density. This variation is illustrated for a single RTG in Figure 4.1-7. The energy distribution of neutrons released in induced fissions is essentially the same as for those produced in spontaneous fissions.

Alpha-Neutron Reactions - The most significant, and currently most uncertain, source of neutrons from Pu^{238} RTG's comes from the interaction between high energy alpha particles from natural decay of Pu^{238} and nuclei of light elements. Because of the short range of alpha particles, these nuclei must be present either in the fuel molecule or as homogeneously distributed contaminants introduced during the manufacturing process. In the case of oxygen, the reaction $^{17}O\ (\alpha\ ,\ n)\ ^{20}Ne$ and $^{18}O(\alpha\ ,\ n)^{21}NE$ becomes the major neutron source term with a yield of approximately

SOURCE NEUTRON FISSION MULTIPLICATION

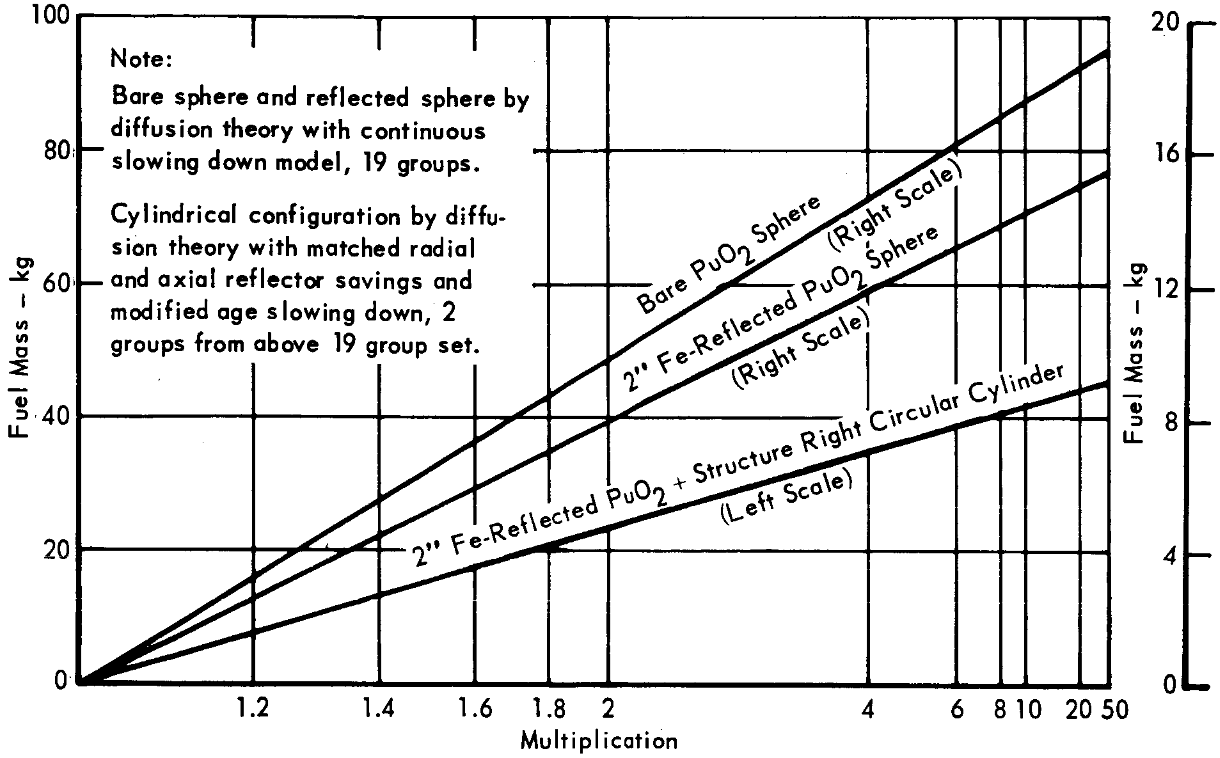


Figure 4.1-7

5×10^4 neutrons per second per gram of oxide. The spectrum of neutrons from (α, n) reactions is difficult to predict or to measure. Estimates of the neutron energy distribution from plutonium oxide fuel form have been made utilizing the fission spectrum reported by Savannah River Laboratory (Reference 4.1-1). An estimated (α, n) energy distribution from oxygen was added, without neutron multiplication, to the fission spectrum. The (α, n) distribution was obtained from experimental work with alpha particles from polonium-210 with natural oxygen (Reference 4.1-2). The resulting neutron spectrum, Figure 4.1-8, shows good agreement with recent preliminary measurements with a Stilbene crystal detector, reported for the SNAP-19 RTG (Reference 4.1-3). Disregarding the small variations near the peak of the distribution, an average curve shows a broad peak at 1.0 to 2.5 Mev. The use of the oxygen 16 isotope in a Pu O_2 fuel form would appreciably reduce the total neutron dose level.

Neutron Yield - The neutron yield from spontaneous fission alone has been fairly reliably established. However, this is only a small fraction of the source of neutrons from plutonium-238. The dominant contributor is the (α, n) reaction. Thus, the yield of neutrons is intimately related to the concentration of light element contaminants; even if the fuel were in the form of the oxide, small traces of certain of the light elements, such as boron or fluorine, could significantly increase the neutron yield. The presence of such contaminants in turn depends on the care observed during fuel processing and on the plutonium history, among other factors. An accurate assessment of the neutron yield from plutonium-238 therefore cannot be made analytically but must await actual measurement of any given batch of fuel.

4.1.2.2 RTG Nuclear Environment - Gamma Radiation - Gamma Sources - Gamma photons from Pu^{238}-fueled RTG's may be contributed by several sources. The characteristic decay gammas of Pu^{238} and its daughters yield a preponderance of gammas in the lower energies. The fission process, either spontaneous or induced, is accompanied by gamma photons emitted essentially at the time of fission, the "prompt gammas." Most fission product nuclei decay with the emission of at least one gamma photon. The majority of these gammas have energies above 1 Mev. Neutron inelastic scattering is generally accompanied by the emission of a gamma photon. Finally, beta particles from the decay of fission product particles collide with surrounding matter and lose

NEUTRON ENERGY SPECTRUM FROM ^{238}Pu FUEL

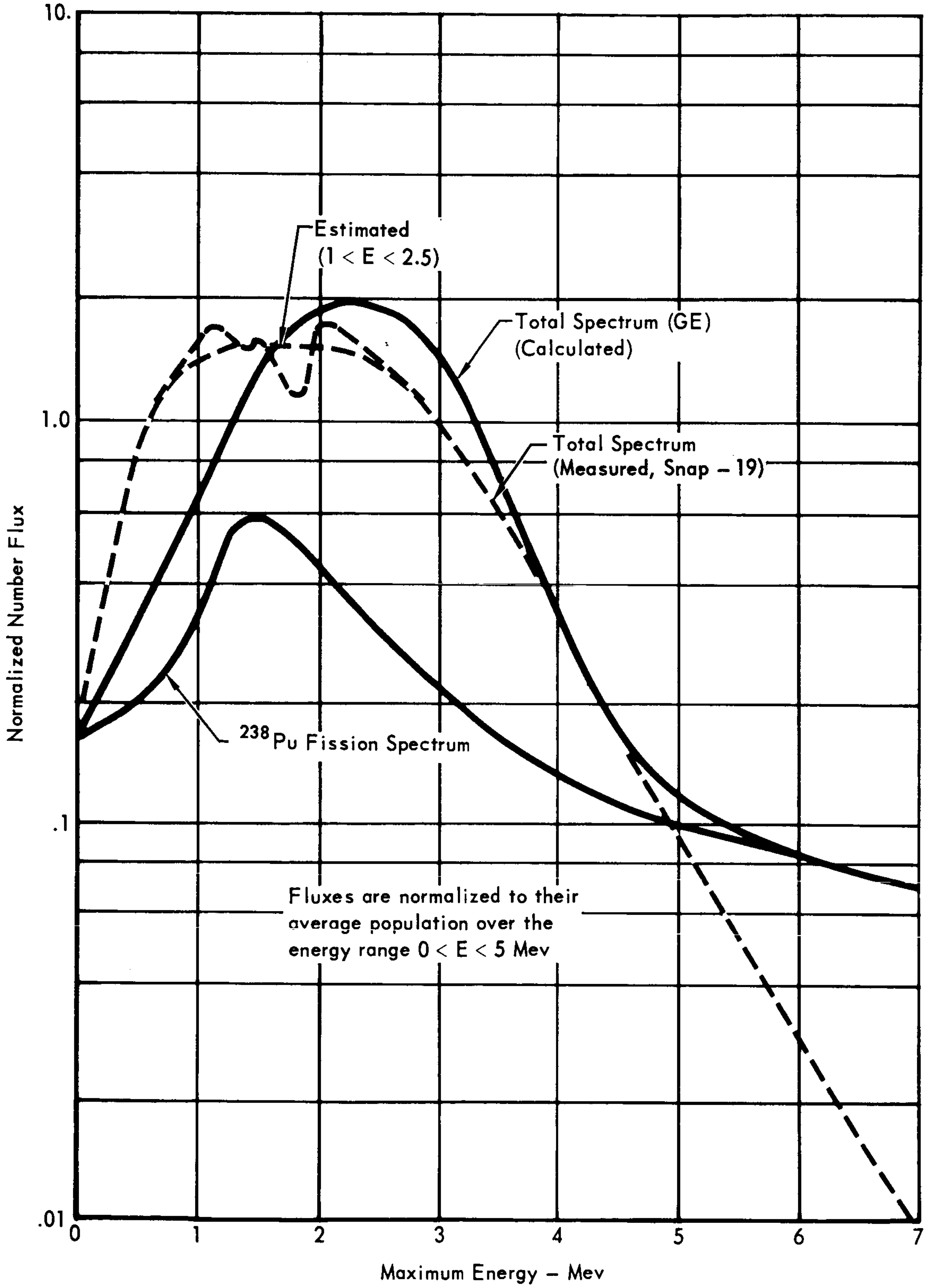


Figure 4.1–8

a part of their energy in the collision. This energy in turn shows up as a so-called bremsstralung photon. The predominant gamma photon population emitted from an RTG has energies between .1 and 1 Mev. The dominant decay gamma at .043 Mev is practically completely absorbed in the fuel.

Gamma Yield Variation with Time - The Pu^{238} product contains only about 80% of the Pu^{238} isotope. Among the several actinide impurities is Pu^{236} which is said to be present (Reference 4.1-1) in concentrations of less than 1.2×10^{-6} grams Pu^{236} per gram Pu^{238} product. The decay chain of Pu^{236} includes thallium-208 with a 34% yield. The decay scheme of thallium-208 produces a 2.614 Mev gamma at 100% yield. The half-lives of the precursers of thallium-208 are such that a concentration builds up with time until it reaches a primary peak value in about 17 years from initial fuel separation. In relation to the other gamma sources from the decay and fission of Pu^{238}, this thallium-208 high energy gamma photon produces as many photons as all of the decay gammas combined after approximately 10 years. For long-term missions, or extended storage prior to launch, this source of gamma radiation must be considered. The presence of Pu^{236} might be controlled by alternate production methods during the irradiation cycle of the neptunium-237 feed material. However, this method is not presently employed.

4.1.2.3 RTG Nuclear Environment - Flux and Dose Rates - The neutron flux and gamma dose rates in the Surface Laboratory is a direct function of the size, number, and configuration of the RTG sources. A parametric investigation of the radiation levels in the Surface Laboratory versus RTG size and configuration is an exceedingly complex task and is unwarranted at this phase of the program.

To demonstrate the analytical technique and provide design point data for the alternate design 1973 RTG-powered Surface Laboratory, the neutron flux and gamma dose rate for a 1500 W(thermal) RTG with SNAP-27 configuration have been determined. The neutron flux and gamma dose rate near one RTG was calculated along the axis and perpendicular to the axis along the fuel capsule midplane, as a function of distance. The calculations were performed for an isolated RTG in vacuum with QAD, a Los Alamos neutron and gamma shielding code, which uses the Albert-Welton kernel for neutrons and the point isotropic kernel plus buildup for gamma calculations. The results are shown in Figures 4.1-9 and 4.1-10. Gamma dose rates are for fresh fuel. Previous, more simplified evaluations (Reference 4.1-4) of neutron and gamma dose rates are

TOTAL GAMMA DOSE RATE AS A FUNCTION OF DISTANCE FROM CENTER AND MIDPLANE OF A 1500 w(th) RTG

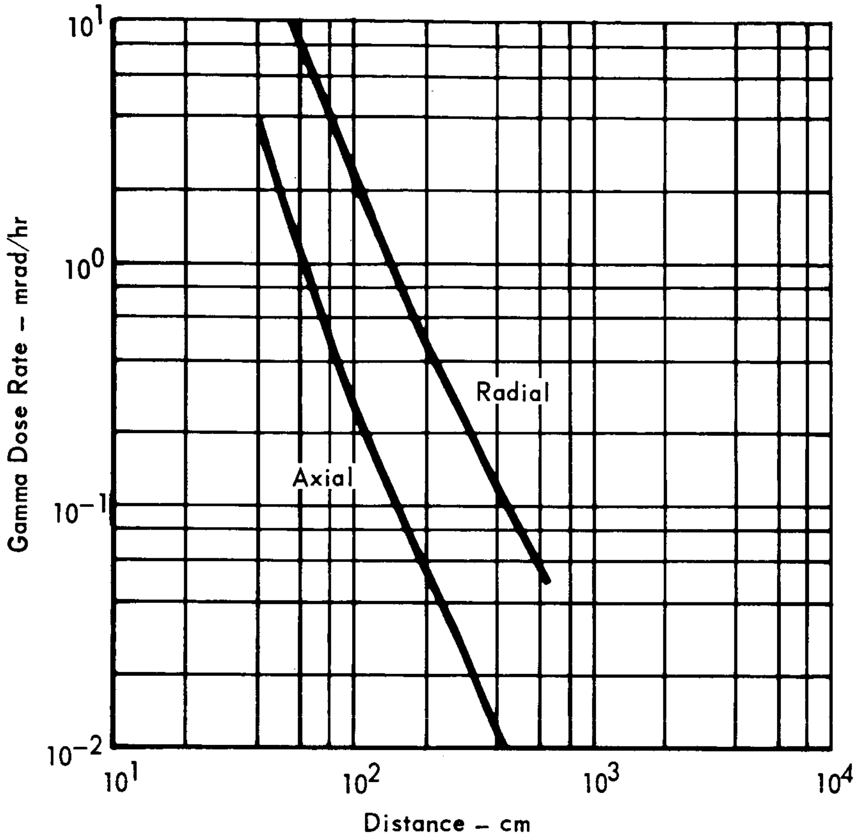


Figure 4.1-9

TOTAL NEUTRON FLUX AS A FUNCTION OF DISTANCE FROM CENTER AND MIDPLANE OF A 1500 w(th) RTG

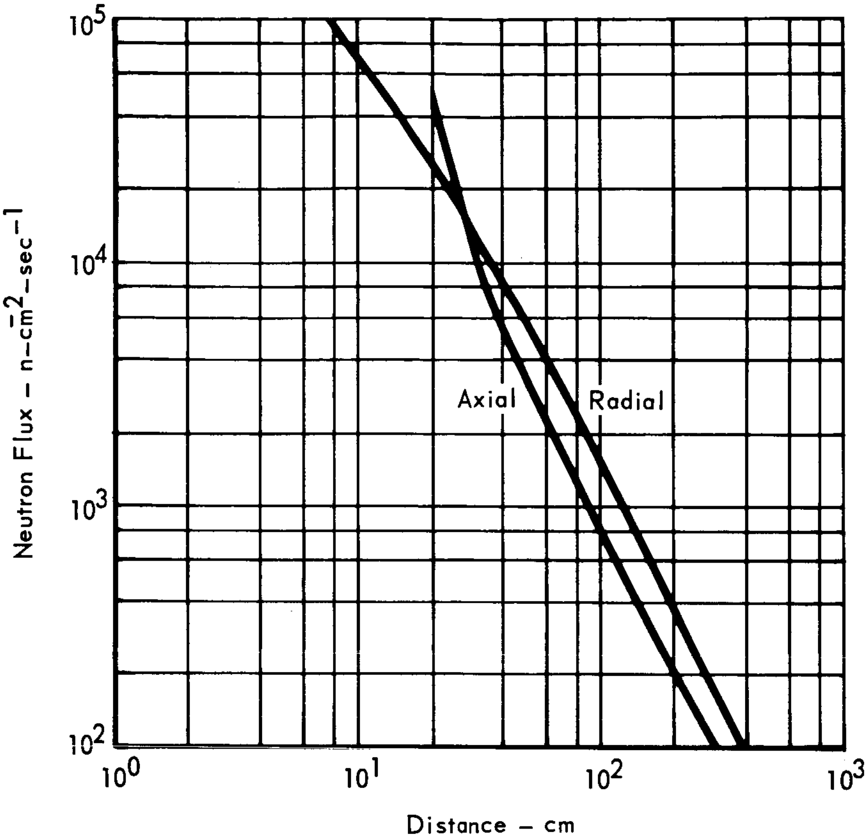


Figure 4.1-10

within reasonable agreement with the above design point values and will be used in the present work to examine the dose rate variation with RTG size. It should be noted, however, that no analytical treatment is complete without attention to such factors as neutron multiplication, gamma buildup, self-shielding, and yield variations with time and that no analytical result can yield the confidence of a good set of experimental measurements.

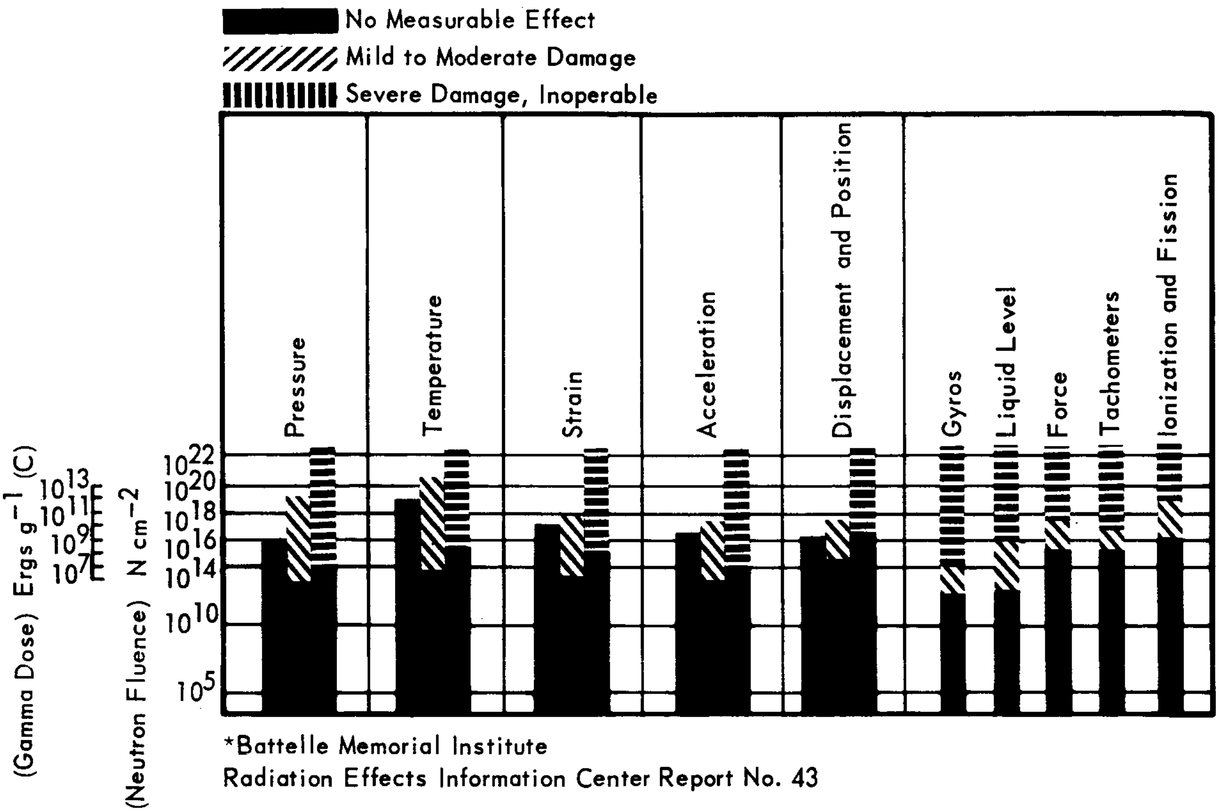
4.1.2.4 Integration Assessment - General - Our studies have considered RTG-equipped Surface Laboratories with RTG sizes from one 1500 W(thermal) unit to three 3300 W(thermal) units. Examination of worst case radiation effects on science instruments has been conducted for a Surface Laboratory with three 3300 W(thermal) units. For this case, the neutron dose rate - mainly from the $^{16}O(\alpha,n)$ reaction - is about 2×10^{11} n/cm^2-year over most of the Laboratory rising to 2×10^{12} within 10 cm of an RTG. At the end of one year, the accumulated gamma ray dose (for photon energies above 0.5 Mev) ranges from 10^4 to 10^5 erg/gm (Al), again depending on the position in the Surface Laboratory. The change with time of the isotope mix in the RTG doubles the accumulated dose every year. The gamma dose rate is 0.02 erg/gm-min at the end of one year; it doubles every two years.

The above dose rates are many orders of magnitude below those for which damage occurs in transducers that might be used in the Laboratory, as shown in Figure 4.1-11. Here "damage" means a measurable change in the operating performance of the transducer. If the RTG dose rates are integrated over a year, the accumulated doses are only lower by a factor of 10 than the accumulated damage doses. However, measured damage was at a high rate for a short time. The integrated dose of Figure 4.1-11 should be used with care, since at low dose rates some damages may anneal itself away.

4.1.2.5 Integration Assessment - Sensitive Instruments - An assessment of the effects of the RTG radiation on some of the more radiation-sensitive science instruments which are typical for 1973 mission and which may be carried on later missions follows.

Metabolic Life Detection by Carbon 14 Counting - One metabolic life detection instrument used in the preferred design group of Surface Laboratory science instruments utilizes the detection of beta radiation from metabolically evolved $C^{14}O_2$. This instrument uses a thin windowed geiger tube with a Ba OH gas collector to

TRANSDUCER
CAPABILITIES FOR RESISTING NUCLEAR-RADIATION DAMAGE*



*Battelle Memorial Institute
Radiation Effects Information Center Report No. 43

Figure 4.1-11

detect the soft beta (endpoint energy 156 Kev) radiation from C^{14} although other detectors such as ionization chambers or proportional counters could be used.

The calculation of the background noise level and signal to noise degradation in the radiation field of an RTG will depend on the details of the detector. An illustrative calculation is given here for an ionization chamber with a gas path of 0.03 gm/cm^2. Such a mass dimension is approximately equal to the path length of the maximum energy beta particle from C^{14}. A cross sectional area of 4 cm^2 is assumed for the detector. The arguments for other forms of detectors based on gas ionization will lead to similar conclusions.

If RTG-generated gamma radiation with energies less than 300 Kev is neglected, the energy of the average photon from the RTG is 812 Kev or 1.205×10^{-6} erg. The mass absorption coefficient for photons of energy greater than about 200 Kev is about 0.03 cm^2/g for all elements. Thus, the efficiency of the ionization chamber is $0.03 \times 0.03 = 10^{-3}$ for counting RTG radiation. Using 10^4 erg/gm-year absorbed dose in aluminum as a reasonable radiation dose, yields an average count rate of

$$\frac{10^4\text{(erg/gm-yr)} \times 60 \text{ seconds/minute} \times 4 \text{ cm}^2\text{(area)} \times 10^{-3}\text{(efficiency)}}{.03 \text{ cm}^2\text{/gm} \times n \times 10^7 \text{ sec/yr} \times 1.305 \times 10^{-6} \text{ erg/photon}} = 1800 \text{ cpm}$$

Discussions of proposed experiments using counting of the B decay of ^{14}C use net count rates of 25 cpm as a basis for sensitivity calculations. Since the total rate must be at least twice background for good statistics, the above number must assume a background of about 25 cpm (from natural radionuclides in construction materials, etc.). According to the above calculations the sensitivity of such an experiment is reduced by a factor of almost 100.

The effects of possible design changes are facilitated by expressing the above calculations in the approximate formula.

$$B = 2000 \ (L/.03)^3 \ (\frac{K}{10})$$

Here B is the background rate from the RTG in counts/minute; L is the characteristic dimension of the chamber in g/cm^2 of counting gas; and K is $4\pi r^2$ times the dose rate (ergs cm^2/gm second) at distance r from the RTG (taken to be K = 10 for original calculations). The β rays from the decay of ^{14}C are emitted in a continuous spectrum from 0 to 156 Kev with a maximum at about 50 Kev. If the ionization chamber were sized for 80 Kev electrons instead of 156 Kev, little β counting efficiency would be lost. On the other hand, the allowed reduction of L to

$0.01\ g/cm^2$ would reduce the background rate by a factor of 30. Furthermore, if the counter were surrounded by $20\ g/m^2$ of lead, the r value is reduced to 4 so that the new count rate would be about 30 cpm, which would only halve the presently estimated sensitivity of ^{14}C counting.

Solid State Detectors

Solid state detectors are used mainly to detect alphas and protons with energies above 1 Mev. The detectors used in the preferred design α spectrometer are lithium-drifted silicon detectors. If all of the alpha or proton energy is absorbed in the detector, only that RTG radiation that can deposit more than 1 Mev in the detector in the form of charged particles is of interest.

Since charged particles have short ranges in solid matter, only charged particles produced in the detector by the long-range gammas and neutrons need be considered. For gammas, the predominant mode of interaction is the Compton effect. The maximum Compton electron energy is

$$E_C = 2(h\nu)^2/(mc^2 + 2h\nu)$$

where mc^2 = .511 Mev and $h\nu$ must exceed 1.2 Mev. About 4% of the gammas that exceed 0.5 Mev also exceed 1.2 Mev. A typical detector made of silicon is 50 microns thick. If $0.03\ cm^2/g$ is used as the absorption coefficient, and 2.4 as the density of silicon, the detector efficiency for photons is

$$0.03 \times 2.4 \times 50 \times 10^{-4}.$$

Detector areas range from 0.2 to 2 cm^2. Using the larger number, and comparing with the first equation of this section, yields a background count rate of 350 cpm for all photons or 16 cpm for photons above 1.2 Mev if the electron lost all of its energy in the active volume of the chamber. However, the range of a 1 Mev electron in silicon is about 0.1 cm, so only those electrons moving in the plane of the detector (about 3×10^{-4} of the 1 Mev electrons produced) would cause a spurious count. Thus gamma interference in the solid state detector can be neglected.

Neutrons cause spurious counts in an alpha or proton detector only via secondary interactions, the most important of which is scattering. The pulse comes from the recoil energy of the silicon necleus. The maximum energy E_m of the residual necleus is given by

$$E_M = 4\ E_m\ mM/(m + M)^2 = .133\ E_m \text{ (for silicon)},$$

where E_m is the neutron energy, m is the neutron mass, and M is the silicon mass. Thus, only neutrons above 7.5 Mev - about 10^{-3} of the total neutron spectrum - can cause a spuriour count. The scattering cross section for 1-10 Mev neutrons

neutrons is 3 barns or .15 cm^{-1}; the detector efficiency is then 7.5×10^{-4} for a 50 micron thick active layer. The expected neutron count is

$$\frac{2 \times 10^{11} \text{ (neutrons/cm}^2 \text{ - yr)} \times 2 \text{ cm}^2 \text{ (area)} \times 60 \text{ sec/min} \times 7.5 \times 10^{-4}}{3.16 \times 10^{7} \text{ sec/yr}}$$

$$= 75 \text{ counts/min}$$

for all neutrons or .075 cpm for neutrons above 7 Mev, so neturons effects are negligible.

For counting β particles, the higher backgrounds must be used: 75 cpm for neutrons and 350 cpm for gammas. Here energy discrimination is not feasible, and the active volume of the detector is large enough to trap most Compton electrons.

4.1.2.6 Radioisotopic Heaters - The potential fuels for use in radiosotopic heaters for VOYAGER are listed below along with some of their characteristics:

Potential Radiostopic Heater Fuels

Isotope	Pu^{238}	Co^{60}	Cs^{137}	Pm^{147}
Half life	86.4 y	5.2 y	30 y	2.6 y
Principle decay modes (energy - Mev)	α(5.49) γ(0.044)	β(0.31) γ(1.17)	β(1.17) γ(0.67)	β(0.223) γ(0.121)
Fuel form	PuO_2	Metal	CsCl	Pm203
Specific Power (watts/gm)	0.39	1.03	0.12	0.29
Shielding (personnel)	Light	Heavy	Heavy	Light

From the figure, it is apparent that Pu^{238} is also a prime candidate for heater application. The discussion of nuclear radiation environment for the RTG is then also valid for heaters.

Since radioisotopic heaters are under consideration for localized as opposed to bulk thermal control, the unit size is small; in the 2-5 watt (thermal) range. For minimally shielded Pm^{147} units in this power range, the dose rate varies from 9 to 20 mrad/hr. respectively at the meter. This is on the order of the radiation levels produced by the RTG and that discussion is valid as a first approximation, to consideration of these heaters.

4.1.2.7 Conclusions - The nuclear radiation environment produced by RTG's and radioisotopic heaters is negligible for the bulk of scientific instrumentation. Some specific instruments such as the C^{14} life detector will require heavy shielding against the radiation background. Shield weight, however, should never be a system problem.

4.1.3 Mechanical Integration - The installation of the science instruments and all supporting subsystems into the Surface Laboratory must satisfy the basic operational requirements of the instruments and yet result in minimum interference with supporting subsystem equipment. In order to resolve conflicts a priority ordering of the considerations for mechanical integration was developed as follows:

a) The operational requirements of deployable Surface Laboratory science or science support equipment (i.e. fields of view, access to surface, etc.) must be provided.

b) The constraints on Surface Laboratory form due to enclosure in the Capsule Bus must be met.

c) Instrument thermal control requirements should be resolved in a manner to minimize thermal control subsystem weight.

d) Installation should be such that mechanically interfacing instruments and support equipment are in close proximity.

The Phase B mechanical integration studies were concerned only with the preferred complement of science instruments.

4.1.3.1 Installation Problem Areas - The problem areas encountered in installation of the science instruments are summarized below:

Deployables - The deployable science instruments and equipment include: imaging cameras; surface sampler; subsurface probe; in situ life detectors; atmospehric package and spectro-radiometer.

The panoramic imaging cameras requires a 360° field of view which maximizes the surface area viewed. A conflicting requirement is the 360° unobstructed field of view required by the high gain antenna.

The surface sampler requires a large angular (120°) field of coverage and has a minimum length of 5 feet, extendable to 9 feet. The sampler must be installed to maximize soil surface area available and must be folded for stowage in the Flight Capsule.

The subsurface probe, also folded for storage is deployed in a pickax manner. This deployment requires an unobstructed arc of 5 feet radius.

The in situ life detector modules are deployed by mortars to distances in excess of 100 feet from the Surface Laboratory. The power and signal cable from the modules to the laboratory must avoid entanglement with other deployables.

The atmospheric package must be deployed to avoid thermal radiation from the laboratory and to minimize wind flow disturbances from other structures.

The spector-radiometer is in two parts, one requiring a nearly 2π steradian unobstructed view of the sky, the other having a narrow pointable field of view. The wide field of view portion is the major problem since the angles subtended at the device by other structure must be minimized.

Surface Laboratory Design Constraints - The configuration of the Surface Laboratory and flexibility for installation is constrained by the overall design and dimension of the Flight Capsule. The height of the SL is limited to 20 inches by the combined requirements of the Capsule Bus's de-orbit motor and parachute and width is restricted to 55 inches by the tankage and plumbing of the Capsule Bus's terminal propulsion subsystem. Additional restrictions are: (1) the thermal control subsystem which requires 4 inches of insulation around the entire SL (this reduces the useable interior dimensions of the SL to 12 inches in height, 47 inches in width and 56 inches in length), (2) the telecommunications subsystem, which uses a large (3 ft. diameter) antenna mounted atop the Surface Laboratory (3) the batteries of the power subsystem, which use over 16% of the volume inside of the insulated portion of the SL, and (4) two sides of the Surface Laboratory are thermal control radiation panels which for proper operation must not be obstructed.

Instrument Thermal Control - The science instrument thermal control problem is divided into two parts: 1) thermal control of instruments internal to the insulated portion of the Surface Laboratory and 2) thermal control of those instruments deployed outside the insulated enclosure. The science instrument thermal control requirements are summarized in Figure 4.1-12.

The average ambient temperature maintained inside the insulated enclosure of the Surface Laboratory ranges from 5°C to 38°C. This ambient is compatible with the bulk of equipment but is too high for the gas chromatograph and life detectors. The gas chromatograph requires maintenance of two columns at 10°C while the life detector culture chambers may require closely regulated temperatures near 2°C. These will require some form of active cooling which, if not carefully applied, can result in a large system weight penalty.

The ambient temperature outside the laboratory enclosure ranges from -123°C to 50°C. Temperatures below -20°C are generally incompatible with operating electronics and some form of active heating will be necessary. The energy required for active heating is directly resolvable into battery weight, therefore the use of active heaters must be carefully applied.

SURFACE LABORATORY SCIENCE EXPERIMENT THERMAL CONTROL REQUIREMENTS

INSTRUMENT COMPONENT	NON OP. TEMP. LIMITS (°C)	OP. TEMP. LIMITS (°C)	TEMP. REGULATION	TOTAL WEIGHT VOLUME	AVERAGE POWER	SENSITIVE ELEMENTS	INSTALLATION
Facsimile Camera							
Camera Head (2)	−55° to +75°	0° to 35°	± 5°	60 IN^3, 5#	15w	Motors, Gears, Detector & Preamp−Detector Response	● Exposed to Atmosphere Elevated Above Lab
Electronics	−55° to +75°	−20° to 75°					● In Lab
Atmos. Package							● All Instruments on Roor to 10' Above Surface
Pressure Trans.	−55° to 75°	−20° to 75°		6.3 IN^3, 0.7#	1.4w	Electronics	● On Boom
Temp, Trans.	−	−		0.5 IN^3, 0.6#	0.01w		● On Boom
Humidity Sensor							
Sensor	−	−110° to 30°		20 IN^3, 0.5#	1w		● On
Electronics	−55° to 75°	−20° to 75°		122 IN^3, 2.5#	4w		● In Lab
Anemometers							
Hot Wire	−	−					● On Boom
Pressure	−	−					● On Boom
Electronics	−55° to 75°	−20° to 75°					
Alpha Spectrometer				600 IN^3, 10 #	2w		
Head	−50° to 75°C	−30° to 50°				Detectors	● In Lab
Electronics	−55° to 75°C	−20° to 75°					
Gas Chromatograph	−55° to 75°	−20° to 50°		400 IN^3, 15#	15 w	Column Heating Recovery, Electronics	● In Lab
Subsurface Probe							
Thermocouples	−	−			0.1w		
Pump	−	−			1w		● In Lab
Spectroradiometer Head & Electronics	−55° to 75°	−20° to 75°		200 IN^3, 5#	2w		● Exposed to Atmosphere
Life Detectors							
Metab Pts 1 & 2	0° to 50°	2° to 30°	±2°	1800 IN^3, 18#	4.5w	Culture Chambers	● In Lab
In Situ Module (4)	0° to 50°	*		8 IN^3, 0.2# Each	2w	Culture	● On Surface
Electronics	−55° to 75°	−20° to 75°			3w		● In Lab
Growth	0° to 50°	2° to 30°	± 2°	500 IN^3 5#	1.5w	Culture Chambers	● In Lab
Soil Sampler & Processor							
Soil Sampler	−	−		750 IN^3, 14 #	30w		● Outside Lab
Sample Processor	−	2° to 30°		1500 IN^3, 8#	10w	Maintain Sample Viability	● In Lab

*Operates on the surface for 5 hours, daytime only. To maintain liquid culture and heat surface, each unit contains a 2w heater which can recover from −30°C.

Figure 4.1−12

Mechanical Interfaces - Mechanical interfaces between science instruments and support equipment are for sample transport. The instruments requiring such interfaces are the gas chromatograph, life detectors and alpha spectrometer.

The preferred sample processor utilizes pneumatic transport of soil samples to all instruments except the alpha spectrometer. The alpha spectrometer requires a smooth, large surface area sample which, in the preferred design, is provided for by a sample pan which is emptied and refilled between measurements.

It is naturally desirable to limit the distance between processor and using instrument in order to reduce the pneumatic tubing required, minimize the required transport gas supply and keep the Surface Laboratory interior uncluttered for easy accessability.

4.1.3.2 Integration Preferred Approaches - The solutions to the mechanical integration problems have been incorporated in the preferred Surface Laboratory design. The installation is illustrated in Figure 4.1-13 and the associated rationale is discussed below.

Deployables - The requirement to provide 360 degree field of view with maximized surface area view for the facsimile cameras conflicts directly with the unobstructed 360 degree field of view requirement for the high gain antenna. The antenna is constrained by Capsule Bus design to be mounted on top the Surface Laobratory. Various mast mounting arrangements for the cameras were considered but rejected because of antenna pattern interference. A final solution was to mount the cameras one on each top edge of the laboratory, below the view of the antenna. This arrangement provides for full panoramic viewing and good viewing of the surface experimentation sites (surface sampler and subsurface probe sites).

Various mounting arrangements for the surface sampler were considered. A corner mounting was chosen for the following reasons:

a) Because the minimum length of the sampler is 5 feet, it can be folded from the corner to obtain minimum overlap of the Surface Lab top surface.
b) This minimizes the heat short through the insulation.
c) This also maximizes the area available for sampling since the distance to the footpad edge is a minimum.

The installation of the subsurface probe was resolved through considerations similar to the surface sampler.

The in situ probe mortars were located at the top edges of the thermal radiation panels since no other deployables are located in front of the panels. Entanglement of the module cables with moving equipment is therefore not possible

SCIENCE SUBSYSTEM EQUIPMENT LOCATION

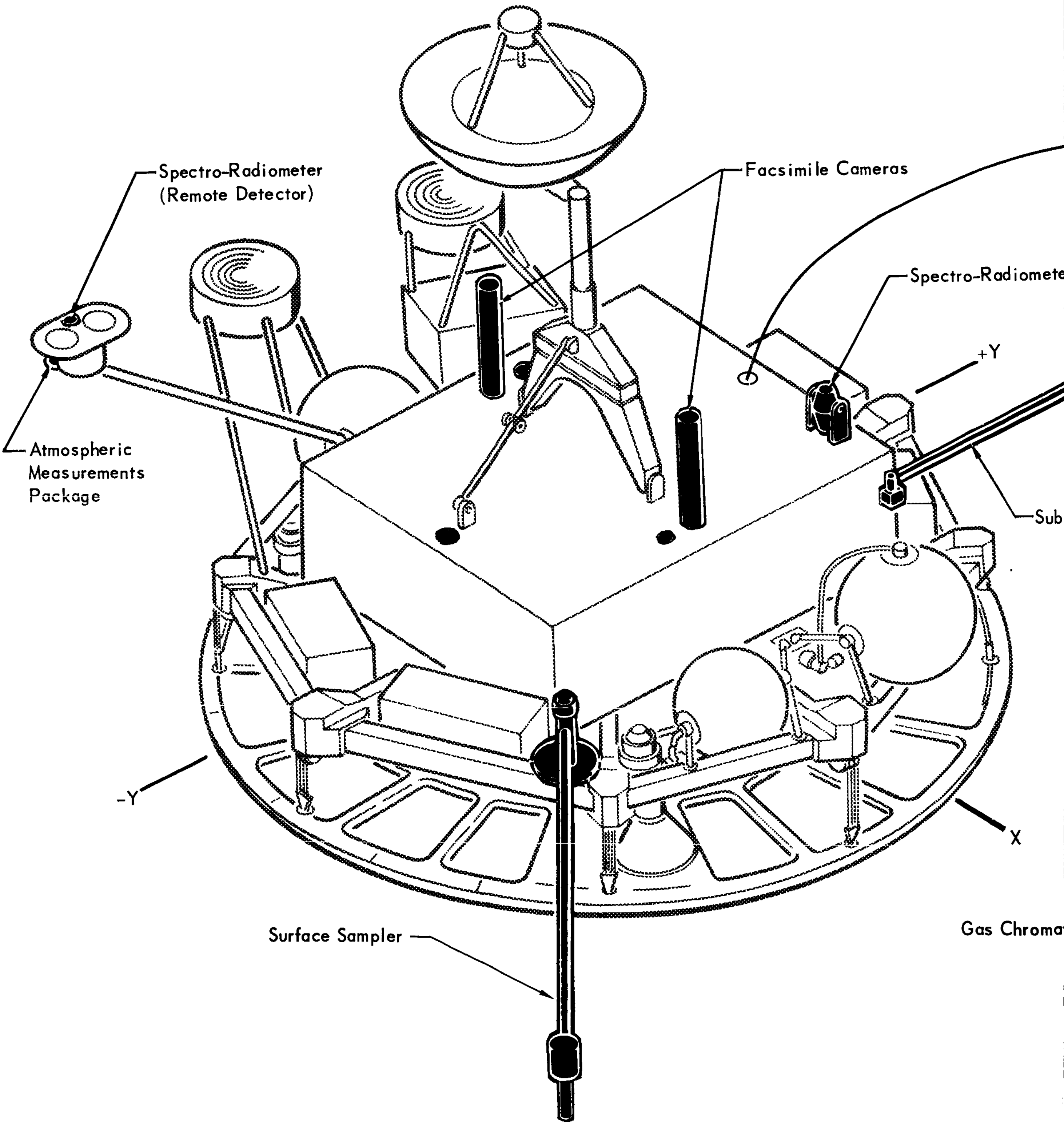
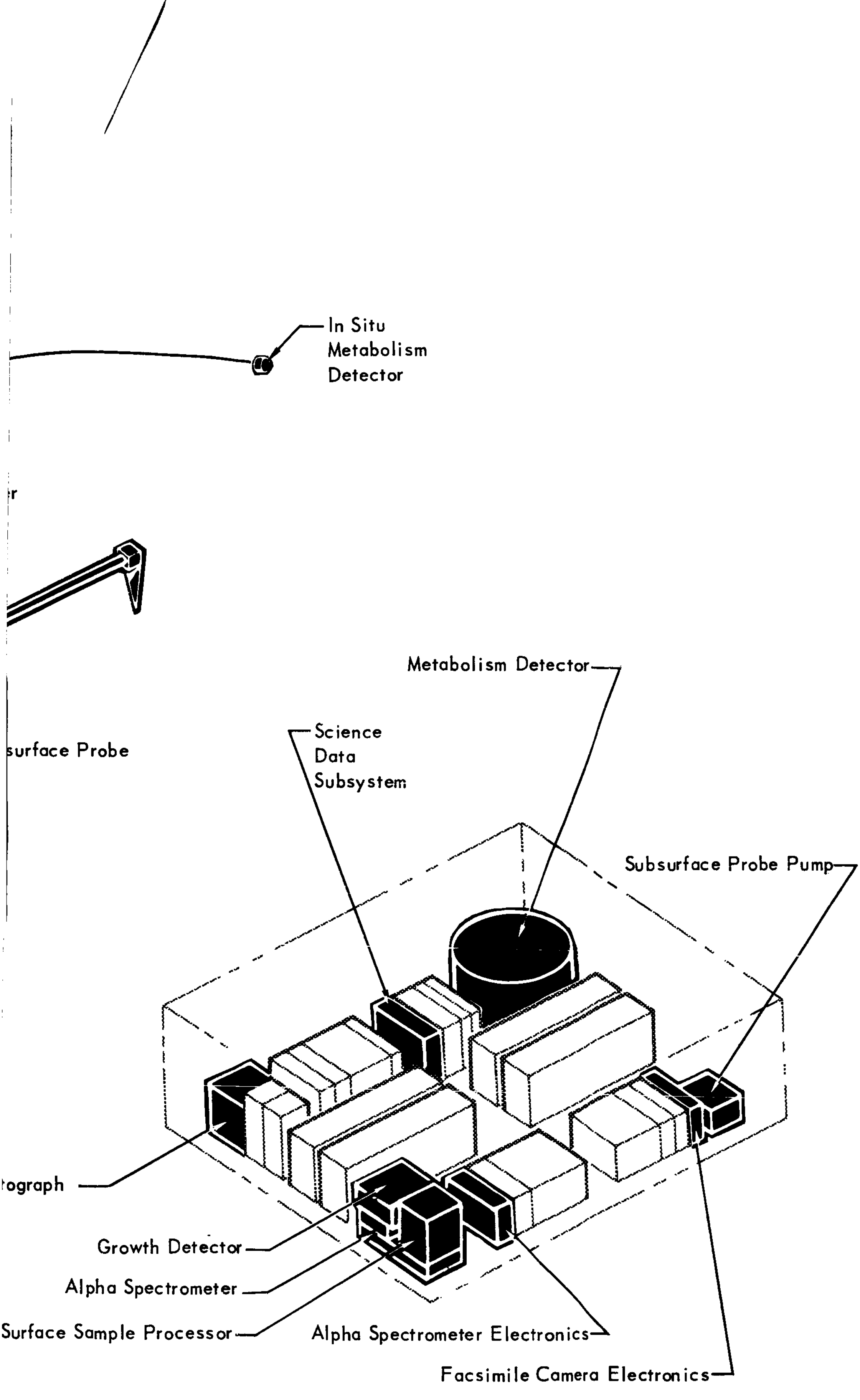


Figure 4.1-13


In Situ
Metabolism
Detector
ɪr
surface Probe
Metabolism Detector
Science
Data
Subsystem
Subsurface Probe Pump
tograph
Growth Detector
Alpha Spectrometer
Surface Sample Processor
Alpha Spectrometer Electronics
Facsimile Camera Electronics

4.1-31-2

and the thin cables do not interfere with the panel radiation pattern.

The atmospheric package was initially mounted on a vertical mast from the top of the Surface Laboratory but this mounting interferred with the high gain antenna pattern. It was found that mounting the package on the boom supporting the low gain antennas was satisfactory for atmospheric measurements purposes and in addition eliminated the extra boom weight.

The wide field of view portion of the spectro-radiometer is also mounted on the low gain antenna boom since this position minimizes the angle subtended by the high gain antenna. The pointable portion is mounted near the laboratory edge to permit near surface viewing.

Thermal Control - To minimize the requirement for cooling the gas chromatograph and life detectors these instruments were arranged so that one was at each corner of the lab. This provides the most remote placement from the high power dissipating subsystem (radio and telemetry) and reduces the instrument surface area viewing the laboratory interior.

Thermal control of the externally deployed instrumentation is achieved by insulating sensitive components and applying active thermal control in such a manner as to minimize system weight.

Mechanical Interfacing - Mechanical interfacing problems have been minimized as follows:

a) installation of the sample processor adjacent to the soil sampler root.

b) installation of the alpha spectrometer adjacent to the soil processor.

Since the sample processor is in one corner of the laboratory enclosure and the gas chromatograph and metabolism life detector are at the other corners the pneumatic transport system is not optimzied. This results from the priority rating system developed in the introduction of this section.

4.1.4 Electronics Integration - Integration studies have shown that potential instruments of the Science Subsystem require a multitude of control and detailed sequencing signals and produce data in a variety of signal forms. In addition it is anticipated that the instrument definition, and hence electronic interfaces will change through Phase C and D of the VOYAGER program. Post 1973 missions will see new and changed instruments in the Science Subsystems. All these factors contradict the requirement for standardized electronics subsystems for the Surface Laboratory.

Through our design and integration studies, however, the requirement for standardized electronics subsystems has been met through the design of the Science Data Subsystem (SDS) which provides the required interface flexibility.

4.1.4.1 Science Instrument Electronic Interface Summary - Through studies of science instruments proposed for VOYAGER, instruments developed for Surveyor, through vendor contacts, and through our own experiment studies, a set of the possible electrical/electronic interfaces for the preferred science instruments have been defined. These interfaces are summarized in Figures 4.1-14 through 4.1-21 for the eight science instruments.

An explanation of the coding and format used in the figures is given below:

Operating Mode - The operating modes of each experiment instrument are listed and briefly described. Modes are here defined on the basis of different required data sampling rates or operating time. The variation of science data sampling rate with mode is indicated.

Science Data Characteristics - The science parameters to be sampled, along with the form of output from the instrument and the required encoding accuracy (in terms of digital bits), is given. Each parameter name represents a separate output line; numbers in parantheses behind a parameter indicate a corresponding number of output lines. Sampling rates for these parameters are given under Operating Mode. Unique sample rates and other comments are given under the Remarks column.

Engineering Data Characteristics - Self explanatory.

Command and Sequencing Summary - Individual commands to the experiment are listed along with their description and basis, where available. Codes under Type have the following meanings:

D - Discrete

P - Proportional

RT - Real Time

NRT - Non Real Time

R - Radio

NR - Non Radio

Status Commands - These are signals generated by the experiment to the Science Data Subsystem (SDS) for overall science subsystem sequencing or for data synchronization.

Power Summary - Self explanatory.

Although these interfaces are largely hypothetical, they have been used in our

ELECTRONIC INTERFACE SUMMARY
EXPERIMENT: VISUAL IMAGING
(FACSIMILE CAMERAS)

1. Operating Modes

MODE	MODE CHARACTERISTICS	SCIENCE PARAMETER SAMPLE RATE
Low Resolution Panoramic Stereo	2 frames, one each camera, to be completed first morning, each frame 300 elements x 1200 lines	Video – 5 lines/sec
Medium Resolution	5 frames (both cameras used) to be completed first afternoon, each frame 400 elements x 400 lines	Video – 5 lines/sec

2. Science Data Characteristics

PARAMETER	FORM	ACCURACY	REMARKS
Camera 1 Video	Digital	6 bits/element	L.R. Pan-Stereo Mode – 9000 bits/sec – 216×10^6 bits/frame M.R. Mode – 12,000 bits/sec – 0.96×10^6 bits/frame
Camera 2 Video	Digital	6 bits/element	Same as Camera 1
Camera No.	B.L.	1 bit	1/frame
Frame No.	Digital	3 bits	1/frame
Azimuth	Digital	7 bits	1/10 lines (L.R.) 1/50 lines (M.R.)
Field Stop	B.L.	1 bit	1/frame
Lens 1 Position	B.L.	1 bit	1/frame
Lens 2 Position	B.L.	1 bit	1/frame
Camera No.	B.L.	1 bit	1/frame

3. Engineering Data Characteristics

PARAMETER	FORM	ACCURACY	SAMPLE RATE	REMARKS
Motor 1 Temp.	0–40 mv	7 bits	One/15 min.	
Motor 2 Temp.	0–40 mv	7 bits	One/15 min.	
Tube Temp.	0–40 mv	7 bits	One/15 min.	

Figure 4.1–14

4. Command and Sequencing Summary

COMMAND	TYPE	MAXIMUM NRT DELAY & TOLERANCE	MAXIMUM PROPORTIONAL VALUE & TOLERANCE	MODE/TIME OF OCCURRENCE
Power On	D–NRT–R&NR	12 h ± 1 m		
Power Off	D–NRT–R&NR	1 m ± 1 m		Upon mode complete
Initiate	D–NRT–R&NR	5 m ± 1 m		From power on
Azimuth	P–RT–R		120 Values	
Camera No.	D–kT–R			
Clock (L.R. Mode)	D–RT–NR			For readout
Data Readout	D–RT–NR			
Field Stop	D–RT–R			
Lens 1 Pos.	D–RT–R			
Lens 2 Pos.	D–RT–R			
Clock (M.R. Mode)				For readout
Heater On-Off	D–RT–R			
Elevation	D–RT–R			

5. Status Commands

COMMAND	FUNCTION
Mode Complete	Generated at completion each mode
End of Line	Data sync

6. Power Summary

OPERATION	AVERAGE POWER	VOLTAGE & REGULATION	PEAK POWER	DURATION OF OPERATION
Imaging	15W	28 ± 5Vdc		30 min. nominal

ELECTRONIC INTERFACE SUMMARY
EXPERIMENT: ATMOSPHERIC PROPERTIES
(PRESSURE, TEMPERATURE, HUMIDITY AND WIND SENSORS)

1. Operating Modes

MODE	MODE CHARACTERISTICS	SCIENCE PARAMETER SAMPLE RATE
Day/Night	Instruments energized every 15 minutes, operates at all times other than during sunrise/sunset mode	One/15 minutes*
Sunrise/Sunset	Instruments continuously energized – from one hour prior to sunrise and sunset to one hour after	One/minute*

*All parameters sampled within one minute

2. Science Data Characteristics

PARAMETER	FORM	ACCURACY	REMARKS
Pressure	0–5V	8 bits	
Temperature	0–40 mv	8 bits	
Humidity	0–5V	8 bits	
H.W. Anemometer 1	0–120 mv	8 bits	
H.W. Anemometer 2	0–120 mv	8 bits	
H.W. Anemometer 3	0–120 mv	8 bits	
H.W. Anemometer 4	0–120 mv	8 bits	
H.W. Anemometer 5	0–120 mv	8 bits	
Pressure Anemometer 1	Digital	8 bits	Sampled once every 5 minutes – all modes
Pressure Anemometer 2	Digital	8 bits	Samples once every 5 minutes – all modes

4.1-35-1

Figure 4.

3. Engineering Data Characteristics

PARAMETER	FORM	ACCURACY	SAMPLE RATE	REMARKS
Press. Trans. Temp.	0–40 mv	7 bits	1/15 min.	Sampling synched to science sample
Anemom. Temp. 1	0–40 mv	7 bits	1/hour	"
Anemom. Temp. 2	0–40 mv	7 bits	1/hour	"
Anemom. Temp. 3	0–40 mv	7 bits	1/hour	"
Anemom. Voltage 1	0–40 mv	7 bits	1/hour	"
Anemom. Voltage 2	0–40 mv	7 bits	1/hour	"

4. Command and Sequencing Summary

COMMAND	TYPE	MAXIMUM NRT DELAY & TOLERANCE	MAXIMUM PROPORTIONAL VALUE & TOLERANCE	MODE/TIME OF OCCURRENCE
Power On	D–NRT–R&NR	27h ± 1s		Prior to sampling –all modes
Power Off	D–NRT–R&NR	1m ± 1s, 2h ± 1s		Power on + mode delay
Anem. Readout	D–NRT–R&NR	5m ± 1s		Every 5 minutes
Clock	D–RT–NR			For Press. Anem Readout
Press. Anem. Pwr. On	D–NRT–R&NR			
Press. Anem. Pwr. Off	D–NRT–R&NR			

5. Status Commands: None

6. Power Summary

OPERATION	AVERAGE POWER	VOLTAGE & REGULATION	PEAK POWER	DURATION OF OPERATION
Press., Humidity Sensors	2.4W	28 ± 5Vdc		
Temp., Wind Sensors	4.4W	5 ± 0.05Vdc		

ELECTRONIC INTERFACE SUMMARY
EXPERIMENT: SOIL ANALYSIS
(ALPHA SPECTROMETER)

1. Operating Modes

MODE	MODE CHARACTERISTICS	SCIENCE PARAMETER SAMPLE RATE
Prepared Sample	Eight hour count each sample, 3 samples total, after surface sampler operation	One/8 hours
Background Count	Two hour operation before prepared sample mode	One sample

2. Science Data Characteristics

PARAMETER	FORM	ACCURACY	REMARKS
Alpha Spectrum	Digital	8 bit	256 – 8 bit Words/Sample
Proton Spectrum	Digital	8 bit	256 – 8 bit Words/Sample

*Each channel 8 bits – all channels readout at end of 8 hour accumulation

3. Engineering Data Characteristics

PARAMETER	FORM	ACCURACY	SAMPLE RATE	REMARKS
Head Temp. 1	0–40 mv	7 bits	1/15 min.	
Head Temp. 2	0–40 mv	7 bits	1/15 min.	
Voltage 1	L.L.	7 bits	1/15 min.	
Voltage 2	H.L.	7 bits	1/15 min.	
Det. Data Rates (6)	0–5V	7 bits	6/hr	Each of six sample one/hour
Det. States (6)	Bilevel	1 bit	6/hr	

Figure

4.1-36-1

4. Command and Sequencing Summary

COMMAND	TYPE	MAXIMUM NRT DELAY & TOLERANCE	MAXIMUM PROPORTIONAL VALUE & TOLERANCE	OCCURRENCE MODE/TIME OF
Power On	D–NRT–R & NR	24 h ± 1 m		Upon sample ready signal
Power Off	D–NRT–R & NR			
Clock	D–RT–NR			
Readout Steering	P–RT–NR		512 Values Channel Address	After data ready*
Heater On	D–RT–R			
Heater Off	D–RT–R			

*See status commands

5. Status Commands

COMMAND	FUNCTION
Data Ready	Alerts data system after prescribed count period or upon accumulator overflow.

6. Power Summary

OPERATION	AVERAGE POWER	VOLTAGE + REGULATION	PEAK POWER	DURATION OF OPERATION
Spectrum Analysis	2W	28 ± 5Vdc		26 hours

ELECTRONIC INTERFACE – SUMMARY
EXPERIMENT: GAS ANALYSIS
(GAS CHROMATOGRAPH)

1. Operating Modes

MODE	MODE CHARACTERISTICS	SCIENCE PARAMETER SAMPLE RATE
Atmos – Subsurface Gas Analysis	• Eight – 20 minute analyses, three each at sunrise and sunset, one at noon and one at midnight. Subsurface 10 minute Cols. 1 and 2 parallel operation then atmos 10 minute Cols. 1 and 2 parallel operation	One/3 seconds
Soil Volatiles Analysis	• 4 – 10 minute analyses phased with soil sampler operation. Temp 1 – 10 min Cols. 1 and 2 in parallel then 40 min Cols 3 and 4 in parallel then cool 10 min. Repeat for Temp 2.	One/6 secs (Cols. 3 & 4) One/3 secs (Cols. 1 & 2)
Calibration	• Two operations (Before and after use) 50 minutes each – Cols. 1 and 2 in parallel (10 min) and Cols. 3 & 4 in parallel (40 min).	

2. Science Data Characteristics

PARAMETER	FORM	ACCURACY	REMARKS	
Col 1 Hi Sens	0–5V	8 Bit	Used for Atmos & Subsurface Gas – 6400 Bits per 10 min Analysis	
Col 1 Lo Sens	0–5V	8 Bit		
Col 2 Hi Sens	0–5V	8 Bit		Used for Soil Analysis 38,400 Bits per 110 min Analysis (Two Temps).
Col 2 Lo Sens	0–5V	8 Bit		
Col 3 Hi Sens	0–5V	8 Bit		
Col 3 Lo Sens	0–5V	8 Bit		
Col 4 Hi Sens	0–5V	8 Bit		
Col 4 Lo Sens	0–5V	8 Bit		
Col 1 Temp	0–40 MV	8 Bit	One Sample/Two Minutes	During Use Only
Col 2 Temp	0–40 MV	8 Bit	One Sample/Two Minutes	
Col 3 Temp	0–40 MV	8 Bit	One Sample/10 Minutes	
Col 4 Temp	0–40 MV	8 Bit	One Sample/10 Minutes	
Oven Temp	0–40 MV	8 Bit	One Sample/50 Minutes	

4.1-37-1

Figure 4.1-17

3. Engineering Data Characteristics

PARAMETER	FORM	ACCURACY	SAMPLE RATE	REMARKS
Carrier Gas Press	0–40 MV	7 Bits	One/Hour	
Calib Gas Press	0–40 MV	7 Bits	One/Hour	

4. Command and Sequencing Summary

COMMAND	TYPE	MAXIMUM NRT DELAY & TOLERANCE	MAXIMUM PROPORTIONAL VALUE & TOLERANCE	MODE/TIME OF OCCURRENCE
Power On	D–NRT–R & NR	27h ± 1m		Upon Sample Ready
Power Off	D–NRT–R & NR	1 m ± 1m		After Completion Signal*
Mode	D–NRT–R & NR	27h ± 1m	5 Values	

*See Status Commands

5. Status Commands

COMMAND	FUNCTION
Analysis Complete	For Sequencing

6. Power Summary

OPERATION	AVERAGE POWER	VOLTAGE REGULATION	PEAK POWER	DURATION OF OPERATION
Gas Analysis	15 W	28 ± 5 Vdc		15.7 Hours
Oven Heating	50 W	28 ± 5 Vdc		40 Min.

4.1-37-2

ELECTRONIC INTERFACE SUMMARY
EXPERIMENT: SPECTRORADIOMETER

1. Operating Modes

MODE	MODE CHARACTERISTICS	SCIENCE PARAMETER SAMPLE RATE
Part 1 (Insolation)	Energized every 15 minutes, day and night	One sample/sec for 5 sec
Part 2	Operates 7 times/day; evenly spaced during daylight hours – 4 viewing directions/operation	One sample/sec for 72 sec. per viewing direction

2. Science Data Characteristics

PARAMETER	FORM	ACCURACY	REMARKS
Part 1			
Insulation	0–5V	6 bits	60 bits/15 minutes
Temperature	0–40 mv	6 bits	
Part 2			
Radiation Intensity	0–5V	6 bits	1010 bits/view
View Direction	0–5V	2 bits	4 values – sample one/view direction
Temperature	0–40 mv	8 bits	

3. Engineering Data: None

4. Command and Sequencing Summary

COMMAND	TYPE	MAXIMUM NRT DELAY & TOLERANCE	MAXIMUM PROPORTIONAL VALUE & TOLERANCE	MODE/TIME OF OCCURRENCE
Power On (Pt. 1)	D–NRT–R&NR	27h ± 1m		
Power Off (Pt. 1)	D–NRT–R&NR	1m ± 1s		Timed from power on
Power On (Pt. 2)	D–NRT–R&NR	27h ± 1m		
Power Off (Pt. 2)	D–NRT–R&NR			
View Position	P–NRT–R&NR		4 values	
Clock (Pt. 2)	D–RT–NR			1/sec for filter step

5. Status Commands

COMMAND	FUNCTION
Positioning Complete	For sequencing

6. Power Summary

OPERATION	AVERAGE POWER	VOLTAGE & REGULATION	PEAK POWER	DURATION OF OPERATION
Part 1	0.1W	28 ± 5Vdc		109 min.
Part 2	1.9W	28 ± 5Vdc		40 min.

Figure 4.1–18

ELECTRONIC INTERFACE SUMMARY
EXPERIMENT: LIFE DETECTION
(METABOLISM 1 & 2)

1. Operating Modes

MODE	MODE CHARACTERISTICS	SCIENCE PARAMETER SAMPLE RATE
Normal	Operates to twelve hours after soil samples are introduced. In situ portion operates to 5 hours after deployment, daylight only.	One/minute

2. Science Data Characteristics

PARAMETER	FORM	ACCURACY	REMARKS
Science Word I	Digital		One 76 bit word/minute
Science Word II (In Situ)	Digital		One 30 bit word/minute

3. Engineering Data Characteristics

PARAMETER	FORM	ACCURACY	SAMPLE RATE	REMARKS
Engineering Word	Digital		1/10 min.	One 55 bit word/10 min.

4. Command and Sequencing Summary

COMMAND	TYPE	MAXIMUM NRT DELAY & TOLERANCE	MAXIMUM PROPORTIONAL VALUE & TOLERANCE	MODE/TIME OF OCCURRENCE
Power On	D–RT–R&NR			After soil sample ready signal
Power Off	D–NRT–R&NR	12h ± 1m		12h from power on
Recycle	D–RT–R&NR			Once every 10 min.*
Clock	D–RT–NR			10KHz for readout
Start	D–NRT–R&NR			
Read	D–RT–R&NR			Once every minute

*See Status Commands.

5. Status Commands

COMMAND	FUNCTION
Cycle Complete	For sequencing, every 10 min

6. Power Summary

OPERATION	AVERAGE POWER	VOLTAGE & REGULATION	PEAK POWER	DURATION OF OPERATION
Normal (1&2)	6W	28 ± 5Vdc		12 hours
Normal (4 In Situ Modules)	2W each (8W total)	28 ± 4Vdc		5 hours

Figure 4.1–19

ELECTRONIC INTERFACE SUMMARY
EXPERIMENT: LIFE DETECTION
(GROWTH)

1. Operating Modes

MODE	MODE CHARACTERISTICS	SCIENCE PARAMETER SAMPLE RATE
Normal	Operates to 26-2/3 hours after soil samples are introduced	15/5 min.

2. Science Data Characteristics

PARAMETER	FORM	ACCURACY	REMARKS
Housekeeping Turbidity pH	0–5V 0–5V 0–5V	8 Bits 8 Bits 8 Bits	Each line active for 100 seconds sequentially, 5 parameters commutated on each line to be sampled every 20 seconds, "Engineering" data multiplexed as "Housekeeping" data.

3. Engineering Data: None

4. Command and Sequencing Summary

COMMAND	TYPE	MAXIMUM NRT DELAY & TOLERANCE	MAXIMUM PROPORTIONAL VALUE & TOLERANCE	MODE/TIME OF OCCURRENCE
Power On Power Off Clock 1 Recycle Clock 2	D–RT–R&NR D–NRT–R&NR D–RT–NR D–RT–R&NR D–RT–NR	26-2/3h ± 1m		Upon soil sample ready. Timed from power on. Once every 20 sec. Once every 300 sec.* Once every 100 sec.

*See Status Commands

5. Status Commands

COMMAND	FUNCTION
Cycle Complete	For sequencing, every 300 sec.

6. Power Summary

OPERATION	AVERAGE POWER	VOLTAGE & REGULATION	PEAK POWER	DURATION OF OPERATION
Normal	1.5W	28 ± 5Vdc		26-2/3 hours

Figure 4.1–20

ELECTRONIC INTERFACE SUMMARY
EXPERIMENT: SUBSURFACE PROPERTIES
(SUBSURFACE PROBE)

1. Operating Modes

MODE	MODE CHARACTERISTICS	SCIENCE PARAMETER SAMPLE RATE
Day/Night	Temperature sensors energized every 15 minutes, all times other than during sunrise/sunset mode.	One/15 minutes
Sunrise/Sunset	Sensors continuously energized – from one hour prior to sunrise and sunset to one hour after.	One/minute
Gas Sampling	Collects gas samples for gas chromatograph, eight operations total.	See below.

2. Science Data Characteristics

PARAMETER	FORM	ACCURACY	REMARKS
Temperature(10)	0–40 mv	8 bits	
Valve Position	B.L.	2 bits	4 samples during each gas sampling

3. Engineering Data Characteristics

PARAMETER	FORM	ACCURACY	SAMPLE RATE	REMARKS
Pump Gas Pressure	0–40 mv	7 bits	One/hour	

4. Command and Sequencing Summary

COMMAND	TYPE	MAXIMUM NRT, DELAY & TOLERANCE	MAXIMUM PROPORTIONAL VALUE & TOLERANCE	MODE/TIME OF OCCURRENCE
Power On (Temp.)	D–NRT–R&NR	27h ± 1m		
Power Off (Temp.)	D–NRT–R&NR	1m ± 1s, 2h ± 1s		After power o~
Power On (Pump)	D–NRT–R&NR	18h ± 1m		
Power Off (Pump)	D–RT–R&NR			Upon pump cycle complete*
Valve Step	P–RT–NR		8 pulses 5 sec. apart	

*See Status Commands

5. Status Commands

COMMAND	FUNCTION
Unstow complete	For sequencing
Deploy Complete	For sequencing
Pump Cycle Complete	For sequencing

6. Power Summary

OPERATION	AVERAGE POWER	VOLTAGE & REGULATION	PEAK POWER	DURATION OF OPERATION
Temp. Sensing	0.1W	5 ± 0.05 Vdc		6 hours
Gas Pumping	1W	28 ± 5Vdc		4 minutes

Figure 4.1–21

system studies to permit detailed systems design and to provide for a design point system analysis.

4.1.4.2 The Science Data Subsystem Interface - The Science Data Subsystem functions, from Reference 4.1-5, paragraph 4.4.12, are "to provide the sequencing and control of the science instruments, sample acquisition and processing equipment. This function shall include calibration, range switching and data acquisition and data storage." Since the SDS is also the electronic subsystem interfacing directly with the experiments, it has been designed to absorb all the interface flexibility between the science instruments and telemetry (TM) subsystem.

The preferred SDS design consists of a series of Remote Interface Units (RIU's), one for each of the science instruments and for the sample acquisition and processing equipment. Each RIU accepts all data signals from its associated instrument, converts analog signals to digital, provides buffering, and multiplexes all data to the TM subsystem. In addition, the RIU accepts timed, coded commands from the TM programmer and provides the proper activating and sequencing signals to the proper science instrument.

A block diagram of the SDS is presented in Figure 4.1-22.

4.1.4.3 Electrical Power Interface - In general, the operation of the science instruments can be expected to require electrical power at many different voltage levels and degrees of regulation. System studies have indicated that the simplest interfaces can be obtained by providing from the electrical power subsystem only two buses: one raw, unregulated battery power and one regulated level. All other power requirements should then be provided by power supplies in the particular instrument or RIU. The preferred design provides the required power in the form of 28 $\pm$ 5 vdc and 5 $\pm$ 0.05 vdc busses to the science instruments.

4.1.4.4 Integration Assessment - With the preferred Surface Laboratory design the electronic integration functions are:

a) Analyze the science instruments electronics to determine the data, command and power requirements, and

b) coordinate these with the design of the RIU for that instrument.

4.1.5 Science Sequencing and Mission Operations - The sequencing or phasing of the science operations represents a critical interface with the supporting subsystems of the Surface Laboratory. In order to ensure that the science operation objectives can be supported and that no science operation will jeopardize the completion of the entire mission, the science sequence must be resolved through system studies.

SCIENCE DATA SUBSYSTEM
FUNCTIONAL BLOCK DIAGRAM

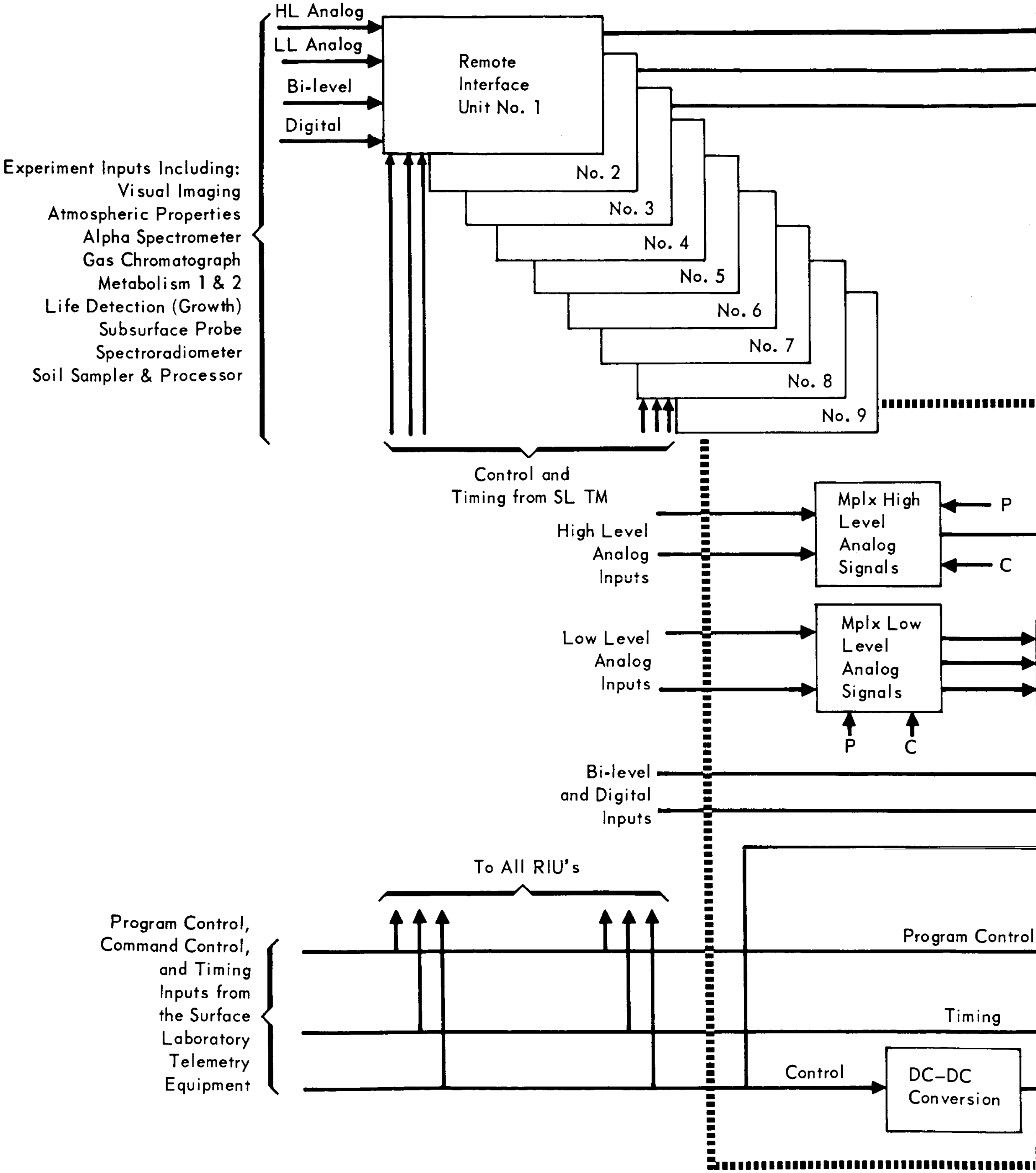
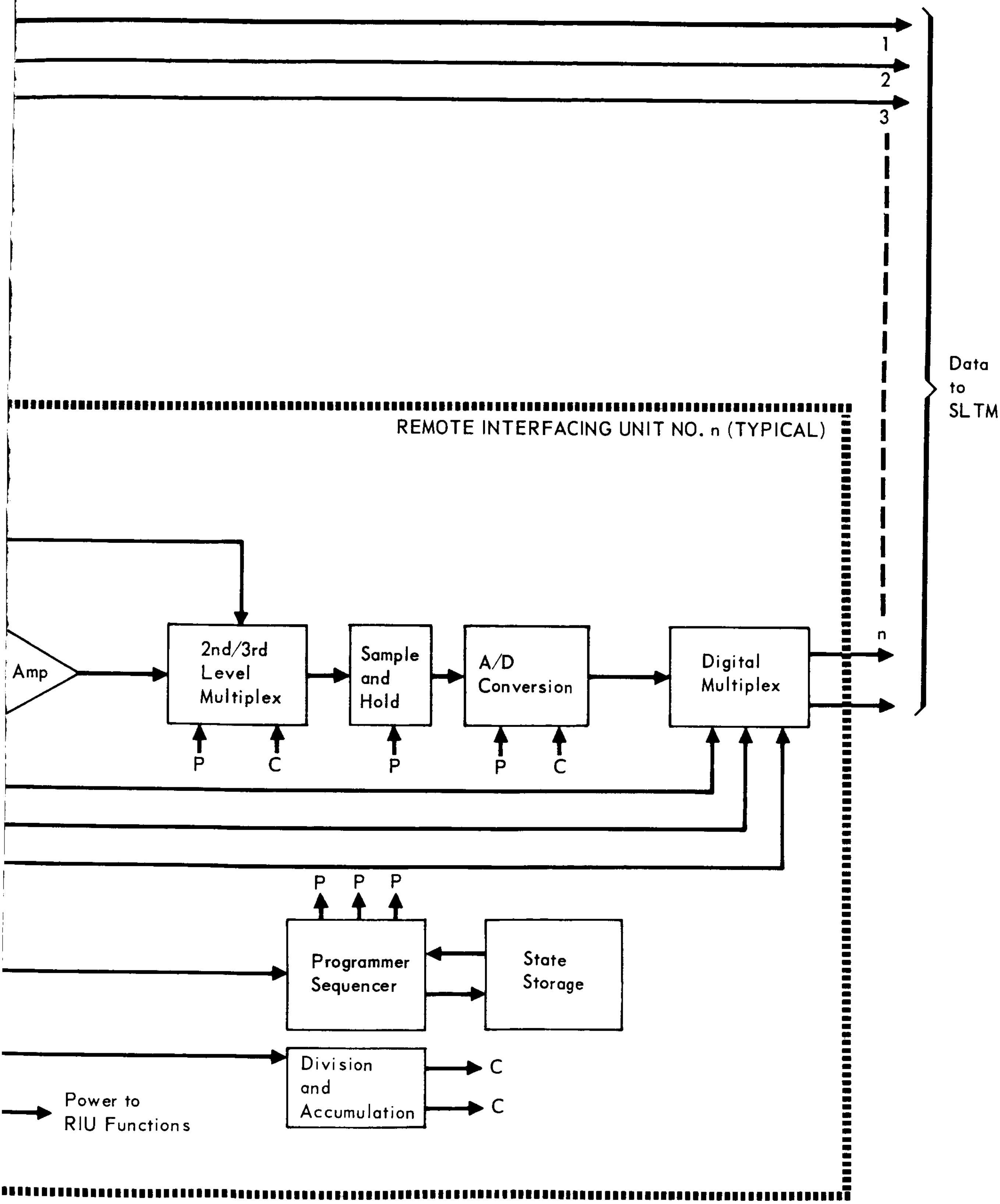


Figure 4.1-22



1
2
3
n
Data
to
SLTM
REMOTE INTERFACING UNIT NO. n (TYPICAL)
Amp
2nd/3rd
Level
Multiplex
Sample
and
Hold
A/D
Conversion
Digital
Multiplex
P
C
P
P
C
P P P
Programmer
Sequencer
State
Storage
Division
and
Accumulation
C
C
Power to
RIU Functions

4.1.5.1 Experiment Constraints - The primary constraint on the mission science sequence is the satisfactory completion of the individual experiment operations. These constraints for the preferred Surface Laboratory complement of experiments and instruments is summarized in Figure 4.1-23. Using these constraints as typical for 1973 mission, it can be seen from the figure that instrument operation can generally be divided into two categories:

1) those operations related to local time of day (solar phase angle) events and
2) those operations independent of time of day, occurring sequentially once the mission is initiated.

These categories can be resolved into a requirement for at least two time references: a local time of day events (i.e. sunrise, noon, sunset and midnight) reference and a mission initiation (i.e. touchdown) reference.

4.1.5.2 Mission and Systems Considerations - Within the individual experiment operating constraints other mission and system related constraints are factors in determining a science sequence; among these are: mission lifetime; timekeeping; landing site selection flexibility, adaptative control of the experiment operations; and the power and data profiles.

Mission Lifetime - For the battery powered Surface Laboratory (preferred concept), mission lifetime is limited and provides a primary constraint on the science sequence. The factors determining mission lifetime are detailed in Section 4.3. It appears that the 1973 mission may be limited to approximately 28 hours if the worst case Martian thermal environment is encountered or may approach 43 hours if a nominal diurnal cyclic thermal environment is encountered. In either case an excess of one diurnal cycle of data can be collected.

Since no prior knowledge of which thermal environment will be encountered is possible, the science sequence should be designed so that all experiment operations can be completed in the minimum time. Extended time, if available can then be used for contingency operations.

Timing and Flexibility for Landing Site Selection - For the preferred Surface Laboratory design, detailed sequencing of the science instruments is provided by the Science Data Subsystem (SDS) within the master timing control of the Sequencer and Timer (S&T) and Telemetry Programmer (TP). The TP contains a prestored list of science instrument commands which are time tagged. When a time word from the S&T matches the time tag on a command, the command is sent to the SDS and the science instrument is activated.

SCIENCE INSTRUMENT OPERATING CONSTRAINTS

EXPERIMENT (INSTRUMENT)	OPERATING MODES	MODE CHARACTERISTICS	OPERATING CONSTRAINTS
Visual Imaging (Facsimile Cameras)	a) Panoramic Stereo	$360^\circ \times 90^\circ$ Total angular coverage by both cameras, 10 minute duration.	Taken morning first day, transmitted first day.
	b) Medium Resolution Selected Area Coverage	Preprogrammed sequence for area coverage of experimentation sites, duration 8 minutes.	Taken afternoon first day, transmitted second day, solar angles $30-60^\circ$ above local landing site plane desired.
	c) Command Coverage	Coverage of Earth commanded areas, may partially or completely override data of medium resolution mode.	Morning of second day reserved for these operations, not necessarily used.
Atmospheric Properties (Atmospheric Package)	a) Sunrise/ Sunset	Sensors continuously energized, each instrument sampled every minute.	From one hour prior to sunrise and sunset continuing to one hour after each event.
	b) Day/Night	Sensors energized every 15 minutes, all instruments sampled within one minute.	All times other than sunrise/sunset mode.
Insolation (Spectro-radiometer)	a) Part 1	Wide field, broad band measurement of insolation using one detector.	One measurement every 15 minutes day and night, <1 minute.
	b) Part 2	Spectral scan, of incident radiation from 4 viewing directions using narrow field, multiple detectors and filters.	Daylight only, total of seven measurements taken at sunrise and sunset with other measurements equally spaced through day. Five minute/measurement (includes 4 views).
Soil Analysis (α Spectrometer)		Instrument takes count of ambient radiation background then analyzes a series of three soil samples prepared by surface sampler and processor.	Background count – 2 hr. soil analysis – 8 hr. each, has a single sample pan which must be dumped and refilled by sample processor prior to each analysis, constrains surface sampler operation.
Gas Analysis (Gas Chromatograph)	a) Subsurface & Atmospheric Gas Analysis	Uses two columns to separate constituents of a gas sample. Subsurface gas sample and atmospheric gas sample analyzed in series (10 minutes each).	Thirty minutes per analysis with flushing operations. Subsurface gas pump cycled at start of mode to observe diurnal variations in composition, analyses at noon, midnight and three times at each sunrise and

Figure 4.1–23

	b) Soil Volatiles Analysis	Uses four columns to separate constitutents in a gas sample obtained by pyrolyzing a soil sample.	Analyzes each of four soil samples gathered. 110 min. per analysis.
	c) Calibration	Calibration on all columns using standard gas sample.	Performed immediately prior to first analysis and immediately after last analysis, 50 min duration each.
Subsurface Probe	a) Sunrise/ Sunset	Nine thermocouples along length of probe + one reference, energized continuously, sampled every minute.	From one hr prior to sunrise and sunset continuing to one hr after each event.
	b) Day/Night	Nine thermocouples and reference energized and sampled every 15 minutes.	All times other than sunrise/sunset mode.
	c) Gas Pumping	Gas pump provides subsurface gas sample for gas analysis.	At start of each gas chromatograph subsurface gas analysis.
Life Detectors 1) Metabolism I	a) Normal	Operates to 12 hours.	After biological soil sample is available.
	b) In Situ	Operates to 5 hours.	After deployment, daylight only.
2) Metabolism II		Operates to 12 hours.	After biological soil sample is available.
3) Growth		Operates to 26-2/3 hours.	After biological soil sample is available.
(Soil Sampler & Processor)	a) Soil Sampling	Takes four soil samples at different locations and depths. First sample is for life detection exps and gas chromatograph, last three for α spectrometer and gas chromatograph, collection times in order 15, 28, 33 and 40 minutes.	First sample started immediately after deployment, last three samples phased to α spectrometer operations to have sample in processor 10 minutes prior to soil analysis.
	b) Sample Processing	Process soil samples and delivers to appropriate exps. Four operations at 10 minutes each.	After each soil sampling.

4,1-45-2

In order to simplify the designs of the S&T and TP and the interfaces between them, it is desirable to keep time references to a minimum. From the experiment constraints it is apparent that at least two time references are required: one reference based on solar phase angle event and one mission reference (e.g. touchdown).

The timekeeping function is intimately related to the problem of providing flexibility for in orbit selection of landing site. Since a change of landing site latitude and longitude will change the relative time of occurance of solar phase angle events, a fixed time base is not possible.

To provide for landing site selection flexibility, the preferred design S&T provides a series of clocks which count down to the nearest solar phase event, and one which counts up from touchdown. These clocks provide for satisfying the experiment constraints and provide for in-orbit updating by providing for Earth's command update of the time to each event for the predicted landing site. In order to take advantage of this simple updating capability, it is desirable to base all experiment operation on times provided by these clocks such that for a range of landing sites no sequencing imcompatibilities will exist.

Flexibility for Adaptive Experiment Control - It is desirable that the Surface Laboratory provide for adaptive control of the science operations; that is to change the experiment operational plan based on initial experiment results. Post-1973 Surface Laboratories will provide this capability through an on-board computer but this is not feasible for the 1973 mission. However, adaptive control is possible for the 1973 mission by utilizing Earth-based decision making capability and the command subsystem. The preferred command subsystem design provides this capability with a command repetoire which includes capability for updating all experiment on-off times and mode selection. To effectively provide for utilization of this capability, the system and science sequence must provide for maximizing the amount of data returned from the experiments during the initial transmission period.

Sequence Interface with Engineering Operations - The science sequence should interface with the operations of the supporting subsystems on a non-interfering basis.

One such interface, for the preferred Surface Laboratory design, is with the engineering sequence. The engineering sequence, activated upon landing, includes a gradual turn on and checkout of the Surface Laboratory subsystems and set up of the high gain S-band antenna. The science sequence should interface with this sequence in two major ways:

1) Deployment of all science equipment should be phased so that deployment can be monitored in the first engineering data frame returned to Earth.
2) Those science associated operations which might interfere with the high gain antenna gyrocompassing (e.g. deployment and surface sampling) should be phased so as not to occur during gyrocompassing.

<u>Power and Data Profiles</u> - The power profile of the science sequence is perhaps the last constraint observed. Although it is desirable to obtain a nearly uniform power drain, this is impossible to accomplish when observing the previously mentioned sequencing factors. The profile has been observed however for the following reasons:

1) to avoid excessive peak power occurrences, and
2) to phase as many operations (internal) as possible during the night so that dissipated power can be utilized for heating.

Due to the use of a tape recorder in the preferred Surface Laboratory design, the science data profile should impose no constraints on sequencing.

4.1.5.3 <u>Preferred Science Sequence</u> - The factors discussed in paragraph 4.1.5.1 are illustrated in the science sequence time line of Figure 4.1-24. This sequence was developed for the preferred Surface Laboratory design and landing site. The sequencing factors are illustrated by the following points.

- Every experiment operation listed in Figure 4.1-23 is completed within the 28 hour minimum duration mission. This is accomplished by initiating the longest duration experiment - the growth life detector - as soon as possible after touchdown.
- Those operations sequenced from each time reference are grouped and identified.
- Within the limits in arrival dates, landing latitude (10°N to 40°S) and landing time (15° to 30°) to morning terminator any landing site will produce no overlapping operations or incompatibilities between solar event referenced and touchdown referenced operations.
- The science sequence interfaces with the engineering sequence in that no instrument operations interfere with gyrocompassing and in that deployment occurs at such time that these operations can be monitored in the first engineering data frame.

SCIENCE SEQUENCE
MORNING LANDING

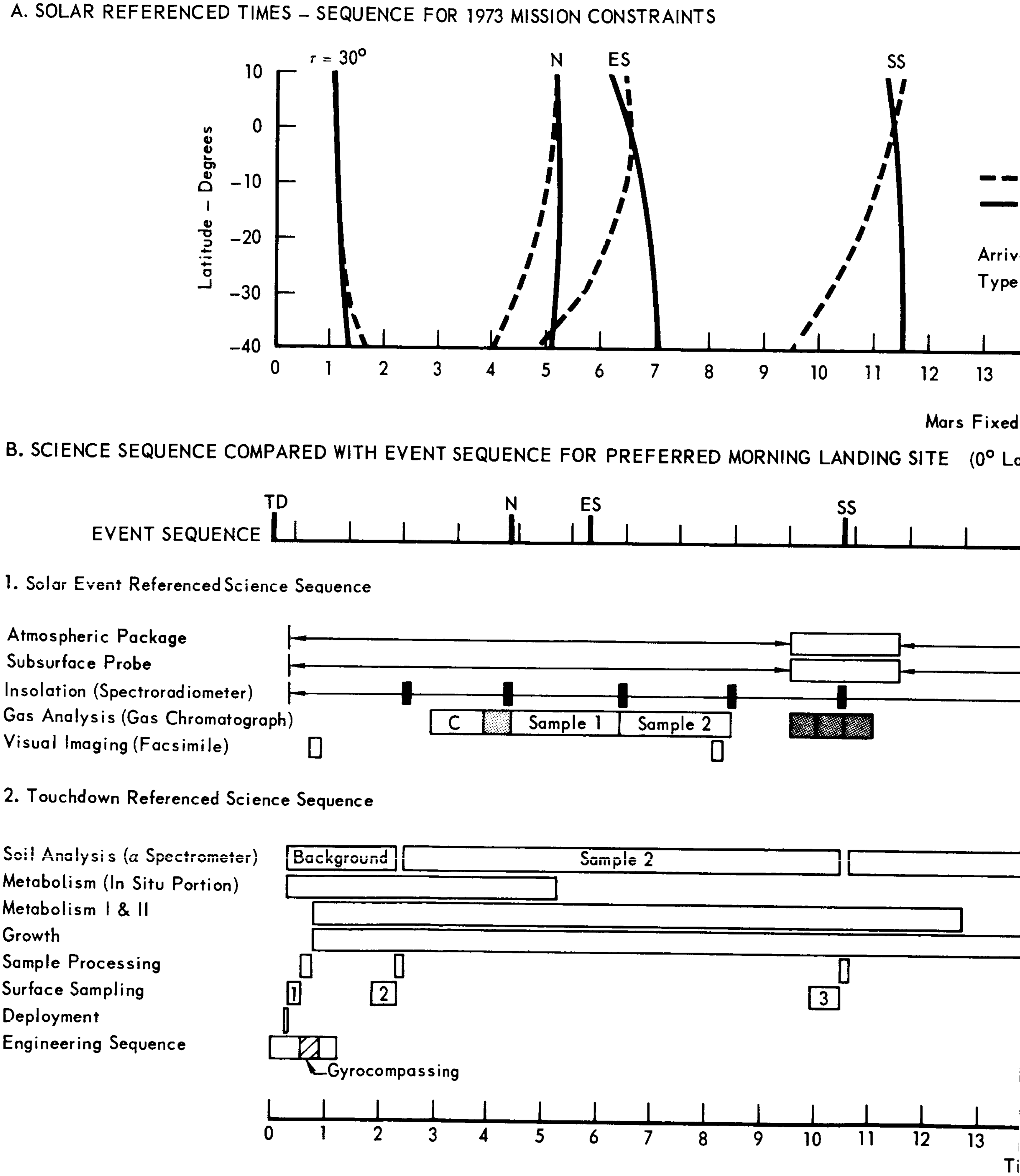
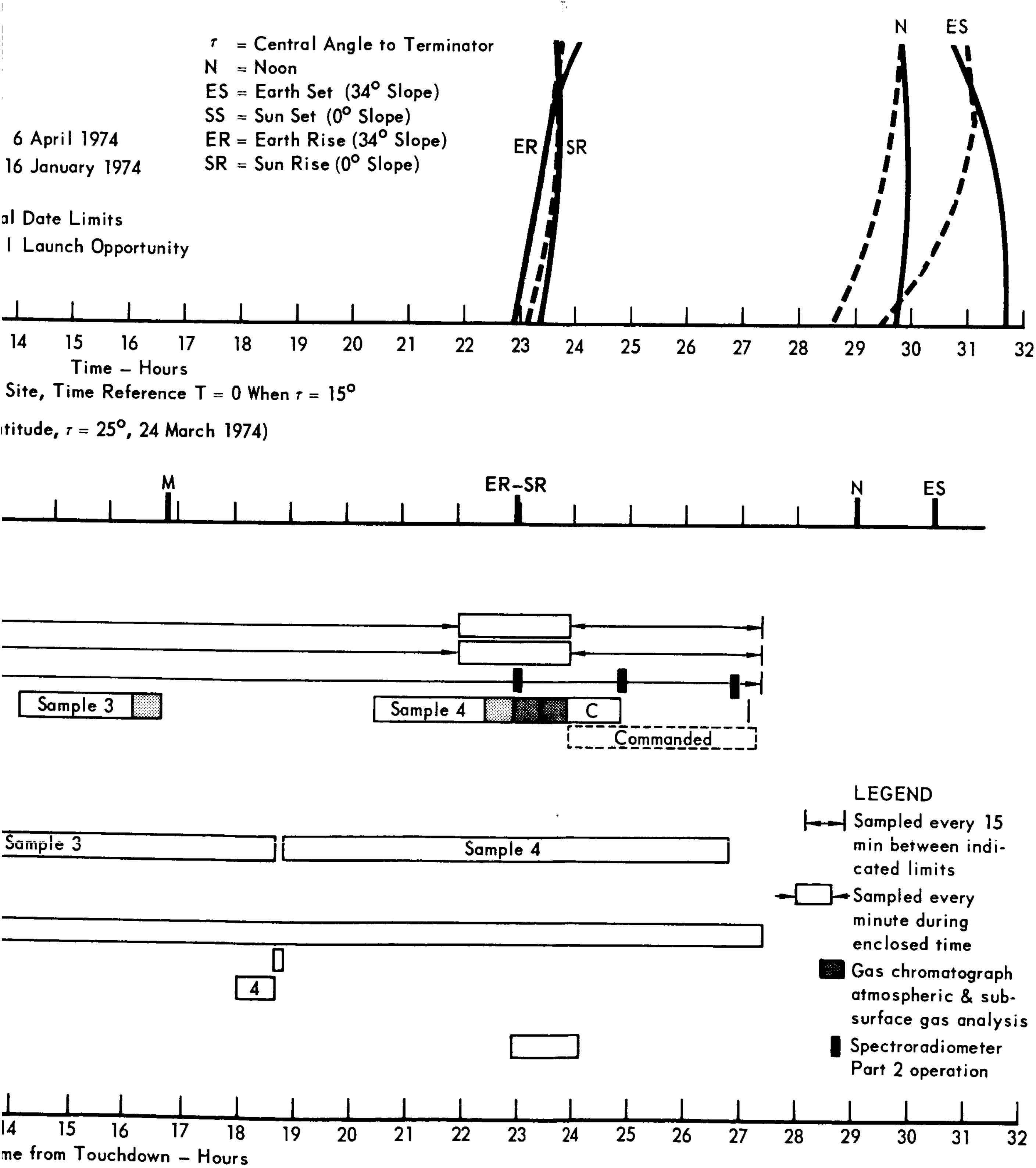


Figure 4.1–24

4.1–48 – I



τ = Central Angle to Terminator
N = Noon
ES = Earth Set (34° Slope)
SS = Sun Set (0° Slope)
ER = Earth Rise (34° Slope)
SR = Sun Rise (0° Slope)
6 April 1974
16 January 1974
al Date Limits
l Launch Opportunity
ER
SR
N
ES
14 15 16 17 18 19 20 21 22 23 24 25 26 27 28 29 30 31 32
Time – Hours
Site, Time Reference T = 0 When τ = 15°
titude, τ = 25°, 24 March 1974)
M
ER–SR
N
ES
Sample 3
Sample 4
C
Commanded
LEGEND
Sampled every 15 min between indicated limits
Sampled every minute during enclosed time
Gas chromatograph atmospheric & subsurface gas analysis
Spectroradiometer Part 2 operation
Sample 3
Sample 4
4
14 15 16 17 18 19 20 21 22 23 24 25 26 27 28 29 30 31 32
me from Touchdown – Hours

4.1-48-2

- To provide for adaptive control, all experiments are started as soon as possible after touchdown. The visual imaging experiment is activated in a low resolution panoramic mode before high gain antenna set up to permit maximizing the photographic data returned during the first transmission period. Medium resolution photos are taken during the afternoon to reserve morning of the second day for possible commanded photographic sequences.

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4.1-3 Martin Co. Preliminary Report "Neutron and Gamma Radiation Measurements of SNAP 19 RTG Subsystem No. 4" 13 March 1967.

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4.1-6 Jumikis, AR; "Soil Mechanics"; D. Van Nostrand Co., Inc.; Princeton, New Jersey; 1962.

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4.2 LEVELING REQUIREMENTS - The subject of leveling the SL to the local gravity vector was examined to determine the need and possible methods of accomplishment. All the subsystems and science experiment instruments on the SL were studied to determine their sensitivity to off-level operation and to establish specific requirements.

4.2.1 Purpose and Scope - We anticipated that an off-level orientation would be degrading to SL performance; the degree of degradation was in question. It was also necessary to determine the feasibility of various methods of leveling the SL. Surface laboratory performance has been evaluated with respect to operation in an off-level orientation of a maximum of 34 degrees from the local gravity vector. (The JPL constaints document requires SL operation after landing on a 34 degree slope.) The performance of support subsystems and science instruments with an off-level orientation is discussed in the following paragraphs. The feasibility of complete SL leveling and the evaluation of the preferred approach to leveling is also covered.

4.2.2 Summary - The recommended approach is to level independently those elements which cannot operate satisfactorily in an off-level orientation. To arrive at this decision two aspects of the leveling problem were examined, namely leveling requirements and leveling methods.

4.2.3 Functional and Technical Requirements - The SL support subsystems and science experiment instruments were examined to determine the performance degradation from off-level orientation, and if off-level operation could be compensated by means other than physical leveling. The following subsystems discussed here are sensitive to leveling; others not discussed here are insensitive to off-level orientation.

4.2.3.1 High Gain S-Band Antenna - The high gain S-band antenna subsystems is designed to be independent of prior leveling for satisfactory operation. However, a penalty is paid to compensate for 34° non-level orientation. The accuracy of the gyro compassing method of antenna pointing control is degraded by off-level orientation. If the off-level orientation has a component adding to the Mars landing latitude angle, the effect is to degrade accuracy. If in a direction so as to subtract from the landing site latitude angle, accuracy improves. If the lander is on the equator, an off-level orientation in any direction is degrading. The pointing accuracy degradation is a function of the sine of the latitude angle, plus the components in the N-S and E-W directions from the orientation. A worst case azimuth error is about 12 degrees for landing at 40 degrees N latitude on a 34 degree slope toward the North Pole. The nominal azimuth error is less than 4 degrees at 40 degrees latitude and vertical orientation. In all orientations and latitude conditions the vertical angle error is less than 2.4 degrees.

It is important that the SL provide a stable platform for the high gain antenna so that disturbances due to wind loading and science equipment motion result in a change of less than one degree from the initial orientation of the antenna.

4.2.3.2 Low Gain S-Band Antenna - Although the low gain S-band antenna subsystem could benefit from leveling, conservative system design dictates that it be capable of satisfactory performance off-level. Even if the SL were leveled, it would be essential for the low rate telemetry and the command subsystem (which operate with the low gain S-band antennas) to be capable of operation under worst case off-level orientation. To accomplish this, a broad antenna pattern is required to assure earth coverage under the worst case orientation conditions.

4.2.3.3 Thermal Control - The heat pipes used for thermal control can be designed to depend on leveling or to be independent of an off-level orientation up to 34 degrees from the local gravity vector. However, the radiator element of the heat pipe is sensitive to the emissivity and illumination of the Mars surface in its field of view for radiant heat transfer. Radiant heat transfer would also be dependent upon the angular relationship to the near-by surface features. This dependence is not necessarily related to the local gravity orientation and leveling would be an advantage only if it would align the radiator to a shadowed surface feature or deep space. Since this is not practical, the thermal control system is designed to operate in the 34° off-level orientation.

4.2.3.4 Spectro Radiometer - The need to level this instrument depends upon the Mars surface terrain in its field of view. Without SL mobility the field of view is fixed. Therefore, this instrument is designed to operate in an off-level orientation up to 34 degrees from the local vertical. If terrain features were known, leveling would permit orienting the field of view for optimum measurements.

4.2.3.5 Sample Acquisition - Since any sample acquisition devices are mounted to the Surface Laboratory structure, leveling of the entire Surface Laboratory could result in a disadvantage to these devices. If they happen on the high end of the SL after leveling, then increased reach and flexibility must be designed into the probe mechanism for this worst case condition. These devices will be designed to not require leveling.

4.2.3.6 Sample Processing - In general, the processing and handling of hard sample materials would benefit from leveling since gravity will play an important role in these functions. Problems in this area can be overcome by proper design of the material handling equipment. The one processing item which must be leveled is the weighing device within the processor. This instrument must be level to within approximately 10 degrees, but this function could be mechanized within the scale itself.

4.2.3.7 Television - The possible benefits of leveling the television camera depend on the nature of local surface features and the extent of surface slopes. If, for example, the capsule lands on a small inclined feature in an otherwise level region, leveling of the camera would benefit the photographic performance. However, if the capsule lands on a slope of large extent, leveling would seriously penalize the performance of the camera or even create a situation in which it could not see some important near surface features or the sample acquisition system. In the presence of this uncertainty, the most favorable strategy would be not to level the camera.

4.2.3.8 Wind Sensor - Since surface winds are expected to be parallel to the local terrain, any advantage in leveling the wind sensor is sensitive to the size of features which would be the cause of off-level situtations of the SL. As in the case of the television camera, the most favorable approach would be not to level the sensor. The performance of the wind sensor would be enhanced by providing it with an all axis sensitivity such that the total wind vector is measured and resolved on the basis of the measured orientation of the capsule.

4.2.4 Design Approaches and Significant Characteristics - Performance penalties incurred by off-level operation of the SL must be compensated by more complex subsystem or science equipment designs. Before accepting this condition, the possibility of leveling the complete SL after landing was examined. Three leveling approaches were explored: 1) level the entire Capsule Bus; 2) level the Surface Laboratory alone; 3) provide a leveling mechanism just to those elements of the subsystem or science instruments that require an on-level orientation. Figure 4.2-1, Leveling Methods Summary, shows the advantages and disadvantages of these three approaches and the following paragraphs discuss some of the details.

4.2.4.1 Level the Capsule Bus Lander - This approach requires a jacking mechanism on the capsule bus capable of lifting 50% of the weight of the lander. The most reliable method of jacking is the electro-mechanical screw actuator. Other

LEVELING METHODS SUMMARY

METHOD	ADVANTAGES	DISADVANTAGES
Level CB Lander	1) All off-level equipment is leveled with one leveling mechanism giving a single design effort for leveling. 2) Reduce complexity of multiple leveling mechanisms.	1) A single point failure mode exist in the leveling subsystem. 2) Leveling mechanism must lift large mass of lander not sensitive to off-level, requiring a large, heavy mechanism and maximum energy storage requirements. 3) Large area foot pads required to support CB mass which are difficult to stow.
Level SL Alone	1) All off-level equipment is leveled with one leveling mechanism. 2) Less mass must be move than the CB leveling which reduces energy requirements and weight of leveling mechanism. 3) Reduce complexity of multiple leveling mechanisms.	1) A single point failure mode exists in the leveling subsystem. 2) Requires a large clearance between the SL and CB to permit 34° angular freedom. This clearance does not exist in the present concept.
Level Individual Elements	1) Less mass must be moved than CB or SL leveling which reduces energy requirements and weight of leveling mechanism. 2) No single point failure mode exists for the SL leveling requirements.	1) Many separate leveling mechanisms must be designed to satisfy the leveling requirements.

Figure 4.2–1

approaches like pneumatic or hydraulic actuators were not considered practical because of added complexity and the requirement for another form of energy storage with its additional weight and degradation of reliability. The proposed electric actuator would drive a foot pad to the Mars surface. This immediately presents an operational reliability problem because of the unknown load bearing properties of the surface material. Loose dry sand, with a bearing strength of 14 lb $/in^2$, would slip free of the pad under vibrations or on a slope. The design constraint of 6 lb $/in^2$ bearing strength would present a still softer surface to support the lander. Any large size rocks strewn on the surface would permit slipping and shifting of the pad as load was impressed upon it. Leveling by this method is unacceptable, it presents a complex equipment design and operational problem.

4.2.4.2 Level the Surface Laboratory Alone - Two basic approaches to leveling the SL, independently of the CB, were examined: 1) mount the SL pallet in a two degree of freedom gimbal frame and unlock after landing; 2) jack the SL pallet to the level position from the lander with an arrangement of jacks and a level sensing system to control the actuator drives. These two approaches will be discussed together because they both present serious mechanical clearance problems from the structure members. The SL pallet would require a clearance of about 20 inches between the capsule bus structure and the SL pallet, to permit an angular displacement approaching 34 degrees in two axes for the worst case leveling condition. The CB lander does not permit a vertical clearance increase of this magnitude for leveling or any other reason.

4.2.4.3 Level Individual Elements - This approach consists of leveling only the elements that are orientation sensitive. Typical elements that would require leveling are:

- Material weighing devices
- Material transport mechanisms
- Material processors that require a sample to cover a detector surface area (alpha spectrometer).

Most of these devices could be individually leveled by pendulous supports that would be released after landing.

4.2.5 Evaluation - It was established in Section 4.2.3, that some science instruments do require leveling and the other subsystems and instruments can be compensated for off-level orientation. The leveling methods discussed were evaluated using the following criteria:

a. Probability of mission success

b. System performance

c. Development risk

d. Versatility

e. Cost

Figure 4.2-2, Leveling Methods Evaluation, shows a comparison of the various leveling methods and their ranking as first, second, or third, based on the above criteria.

4.2.6 Recommended Design Approach - The preferred approach to the SL leveling problems is to level each individual element that is adversely affected by an off-level orientation. If this is not feasible then the equipment should be designed to compensate for an off-level condition.

4.2.6.1 Projected Leveling Requirements for Future Missions - Future mission SL leveling requirements have been examined. The application of the mobility concept (Rover) could reduce the leveling problem and complications resulting from designing for off-level operation. The Rover SL could aid the off-level landing conditions by permitting maneuvers on the Mars surface to position the SL to a level orientation. A local gravity sensor would be required on the Rover vehicle. There probably will be some science instruments on future missions of the SL that will be level sensitive.

- Wet Chemistry - One of the major changes which can be expected to occur in the scientific payload of later missions is the incorporation of wet chemistry processing. Although most wet processing (filtering, dialysis, transport, homogenization, and separation) would seem to require leveling and certainly would benefit from it, little attention has been given (or required) to techniques for off-vertical operation. However, it appears practical to redesign the processing elements for this environment resulting in suitable techniques for operating within 34 degrees of the gravitational vector.
- Drill - Another likely element of experiment equipment which could appear in later missions is a subsurface drill similar to that designed for inclusion in Surveyor. The use of this device and its design could be complicated by leveling of the entire Surface Laboratory. If a drill were located on the downhill side of a leveled laboratory, a considerably greater extension capability would be required before the device could reach the surface and begin operation.

LEVELING METHODS EVALUATION

METHOD	PROBABILITY OF MISSION SUCCESS	SYSTEM PERFORMANCE	DEVELOPMENT RISK	VERSATILITY	COST	CHOICE
LEVEL CB	Low, contains a single point failure mode 3rd	Will do job but with maximum weight and power 3rd	State of art design 2nd	Will satisfy all leveling requirements 2nd	Medium cost 1st	2nd
LEVEL SL	Low, contains a single point failure mode 2nd	Will do job but with medium weight and power 2nd	Very high, no clearance for leveling 3rd	Very difficult to cover worst case leveling 3rd	High cost 3rd	3rd
LEVEL INDIVIDUALLY	No single point failure mode 1st	Will satisfy all leveling requirements, minimum weight and power 1st	Lowest because small leveling mechanisms required 1st	Adaptable to all leveling requirements 1st	High cost because of many mechanisms designed 2nd	1st

Figure 4.2–2

4.3 ANALYSIS OF SURFACE LABORATORY ALTERNATIVES - The objective of this study is to achieve a balanced Surface Laboratory (SL) design by concurrent evaluation of each major subsystem or event to ascertain that the subsystem design satisfies the overall requirements and performance of the entire Surface Laboratory.

4.3.1 Summary - The principal factors governing design of the Surface Laboratory are thermal control, configuration, telecommunications, electrical power, and mission duration. The thermal control subsystem provides the basic boundaries for SL weight and performance. The preferred thermal control design uses electrical heaters to obtain a system design that has maximum capability to adapt to the Martian environment. It allows electrical energy contingency, for redundancy or extended mission operation, if nominal Martian environments exist. Missions with landing sites near the Martian morning terminator and having an operating period (nominally 27-28 hours) of slightly over one duirnal cycle are preferred because they provide maximum data return for fixed SL weight. Configuring the SL to minimize the surface area of the thermally insulated compartment of the SL avoids excessive heat loss during cold environments and maximizes the weight available for experiments or their support subsystems. A telecommunications design utilizing moderate transmitter power, 20 watts RF for the high rate radio link and 5 watts RF for the low rate radio link is preferred. This selection provides adequate data capability, exceeding the requirements of Reference 4.3-1, without significant penalty to either SL weight or equipment temperature. The electrical power subsystem is designed to a zero energy contingency for a -190°F environment; for milder climates, this design yields sufficient contingency by requiring less heater energy.

4.3.2 Factors of SL Design - For this study we will examine the basic design alternatives of the Surface Laboratory. Certain subsystems, such as structure, timing/sequencing, data storage and data handling are essentially constants as far as the major alternatives of the problem are concerned. The science subsystem is also considered fixed for this study. To illustrate the basic effects of the major design alternatives a typical SL, shown in Figure 4.3-1, is used to provide an initial reference point for the analyses. The Entry Science Package (ESP) is included in the weights, although it is not a variable for this problem, to correspond to the weight allocations of Reference 4.3-1. Thermal control and electrical power are excluded from Figure 4.3-1 because they are the major variables of this study.

TYPICAL 1973 SURFACE LABORATORY

ITEM	WEIGHT (lb)	POWER (watts)	COMMENTS
Sequencer & Timer	11.0	12	Provides time-base control & references
Test Programmer	5.0	5.0	Check-out of SL; prelaunch, cruise, pre-separation
Power Switching & Logic Unit	25.0	10.0	
Battery Chargers (4)	12.0	N/A	Not used after landing
Science 5 V Power Supply	3.0	7.0	
Command Subsystem	2.1	2.5	
Data Storage Subsystem	10.0	10.0	Bulk data storage
Telemetry Subsystem			
Telemetry Equipment	28.5	31.0	Includes core memory
Instrumentation Equipment	15.0	25.0	Engineering sensors & data handling
Antenna Mount	30.0	4.0	4 watts tracking, 75 watts erection – includes gimbals, motors, etc.
Antenna Control Subsystem	5.4	.8	Includes gyros, servo electronics, sun sensor and amplifiers
Antenna Subsystem			
Low Rate Transmit Antenna	1.0	–	Radiate low rate data to Earth
Omni Antenna	1.0	–	Receives Earth Commands
High Gain Antenna Assembly	6.5	–	Mounted on antenna mount – 36 in. dia parabola
Diplexer	1.2	–	Separates high rate transmission & received tracking signal
Radio Subsystem			
Low Rate Radio	5.0	30.0	5 watt RF output
High Rate Radio	42.7	129.1	Dual exciters & TWTA's, 20 watt RF output. One system is redundant. Has transponder capability
Tracking Receiver	5.3	5.0	Used for accurate pointing of high gain antenna
Science Subsystem			
Instruments	110.0	22.0*	*Average power. Weight includes sample gatherer & processor
Science Data Subsystem	10.0	11.5	
Basic SL Structure	93.1	–	
Installation Equipment	123.5		Wiring, brackets, connectors, etc.
Entry Science Package	187.7		
Total SL Weight	734.0		Weight exclusive of thermal control & power

Figure 4.3–1

Throughout our studies of the SL and its subsystems, five (5) factors have continually interacted with each other, and the total SL design and performance. These factors are: (1) thermal control, (2) configuration, (3) telecommunications, specifically the radio links, (4) electrical power, and (5) mission duration. Sections 5.8, 5.5, 5.4, 5.1 and 2, respectively, present detailed discussions of these factors.

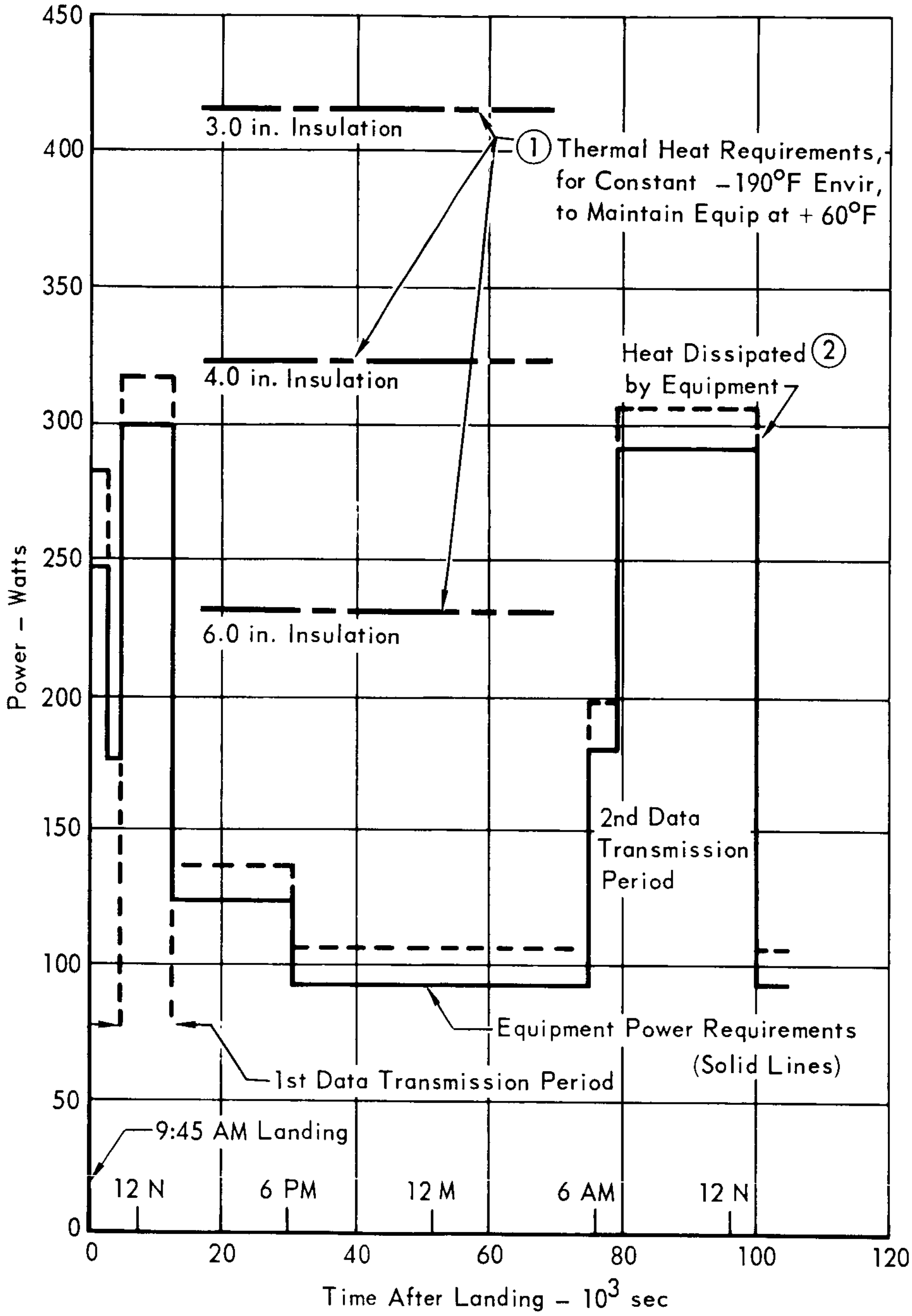
4.3.2.1 Thermal Control - The Mars ambient temperature extremes of 150° to 322°K (-190°F to +120°F) of Reference 4.3-1, set the limits of design in the thermal control subsystem. The extremely low ambient temperature of -190°F necessitates large amounts of insulation and supplementary heaters to maintain moderate equipment temperatures. The high temperature limit of +120°F creates a problem when devices such as power amplifiers for the radio link are operating because large amounts of heat must be dissipated.

The primary influences of thermal control design, and its interaction with other subsystems is discussed below:

- The power (heat) required for thermal control can completely dominate the overall power profile if electrical heaters are used. This is illustrated in Figure 4.3-2, which shows that for insulation thicknesses of 4 inches or less, the heat required for thermal control exceeds the heat dissipated by the equipment.
- The weight required for thermal control increases almost linearly with the surface area of the SL.
- Thermal control insulation can dominate the volume of the SL. Most of our designs have had upwards of 67% of the volume devoted to insulation.
- In order to maintain equipment temperatures to moderate levels, less than 125°F, supplementary heat radiator surface is normally required. In the process of providing this extra dissipation capability, a loss is incurred in heating efficiency for the continuous cold conditions.

4.3.2.2 SL Configuration - The configuration of the SL determines the volume and areas available for the equipment. The surface area of the SL interacts with thermal control designs - large surface area increases heat losses during a continuous cold environment. Unfortunately, the SL does not have much flexibility when the overall requirements of the Flight Capsule are considered. The height of the SL is limited to 20 inches by the combination of the de-orbit motor and parachute requirements of the Capsule Bus. The width is restricted to 55 inches

EQUIPMENT/THERMAL PROFILES – 1973 SURFACE LABORATORY



① Heat requirement based on 54 ft^2 surface area, 4 heat pipes, and 15.5 ft^2 of radiator

② Battery inefficiency 15%, chemical-to-electrical conversion.

3 25 watts radiated by radio links during data transmission periods, hence is not heat dissipation.

Figure 4.3–2

by the tankage and plumbing of the Capsule Bus's terminal propulsion subsystem. As a result, the SL is only able to vary in length to meet its own requirements for volume and equipment installation. The usable internal volume is further constrained by the volume required for thermal control insulation, which normally is 2 to 4 inches thick. Aside from antennas, sample gatherers, heat pipes and radiators, in-situ life detectors and atmospheric measuring instruments, the basic equipment is located in the insulated portion of the SL.

4.3.2.3 Telecommunications - For this study we will consider that the Surface Laboratory uses two radio links, both S-band. One, the low rate link, is used primarily for engineering data, and uses a MFSK modulation technique and a low gain antenna. The other radio link, high rate, uses a PCM/PSK/PM, square wave subcarrier, convolutional coding with sequential decoding modulation technique, and a high gain pointed antenna. The characteristics of both radio links are shown in Figure 4.3-3.

The quantity of data to be transmitted is subject to many trade-offs and engineering judgment. The minimum requirement of 5×10^6 bits per day can be met with less transmitter power than shown in Figure 4.3-1. A typical worst case landing site, 40°S. latitude, at the end of the mission opportunity, April 1974, yields a minimum communication time of 5 hours per day. A data rate of 300 bps and a transmitter output power of 10 watts would meet the 5×10^6 bits per day requirement. This lower transmitter power would offer some rewards in the area of thermal control for cyclic environments, but fixes the amount of data that is transmitted to a quantity that has no growth capacity for changing mission requirements.

The low rate radio link of Figure 4.3-1 has a data rate of .5 bps. This rate places a severe limit on the amount of engineering data that can be transmitted. Increasing the data would not only allow increased engineering data, but would offer a potential redundant path for critical experiment data such as atmospheric properties and life detection.

4.3.2.4 Electrical Power - The preferred approach to the design of the electrical power subsystem for the 1973 mission uses batteries. This approach, although limited to missions no longer than several days, is presently considered as the best overall method for 1973. By limiting ourselves to only battery configurations, there are still basic trade-offs between SL weight and energy contingencies. Energy contingencies can be used for extended mission life, redundancy, for decreases

RADIATED POWER REQUIREMENTS
(SL RADIO LINKS)

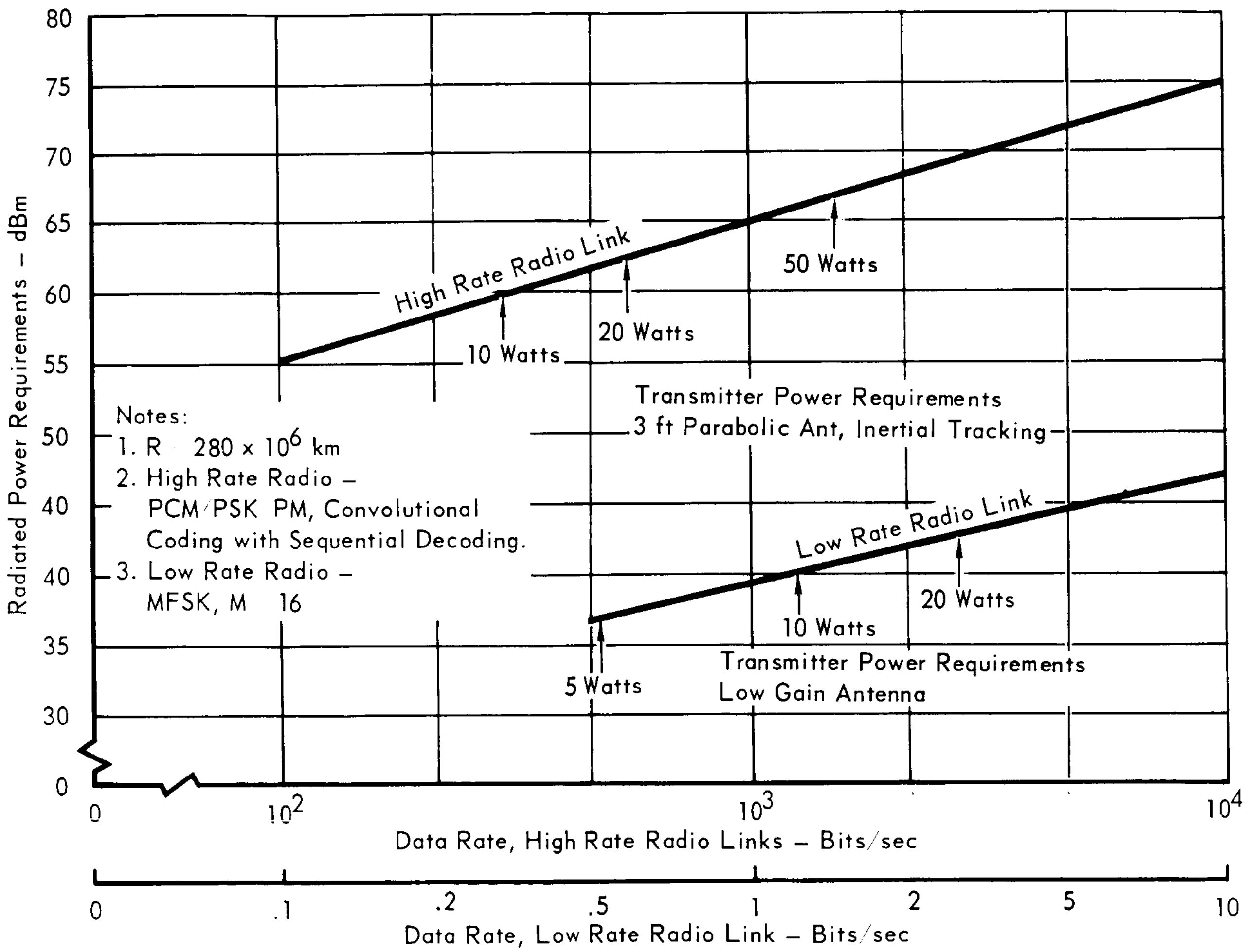


Figure 4.3–3

in battery efficiency after long periods of wet stand without charging throughout the cruise phase of the mission, or to provide a cushion in power/energy estimates because none of the equipment being discussed has been built.

4.3.2.5 Mission Duration - Mission duration is composed of two related parts: landing site and length of the basic mission. Landings near the morning terminator allow immediate communications to Earth. One diurnal cycle of data, as required per Reference 4.3-1, can be acquired and transmitted with a basic mission length of 25 to 30 hours. Evening terminator landings occur out of sight of Earth, and require 12 to 15 hours after landing before communications can be established. At the end of this first communications period, a full diurnal cycle has not been completed; hence, one more Martian night plus a few hours of communications the next morning is required to satisfy Reference 4.3-1. The basic effects of mission duration are in the area of electrical power and thermal control design - increased duration requires more electrical energy.

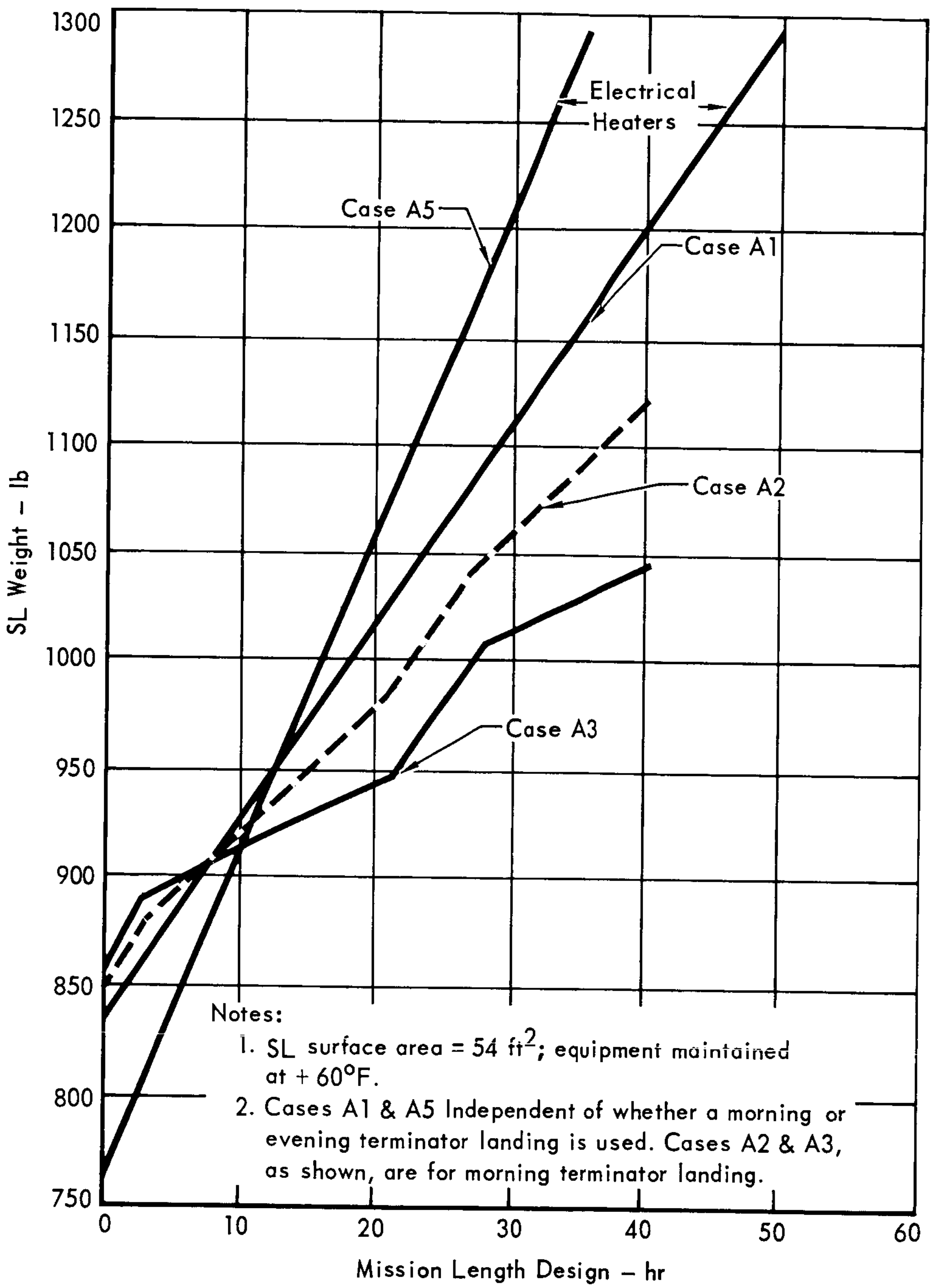
4.3.2.6 Constraints - In order to limit this study to conservative alternatives and configurations, only fixed (non-mobile) vehicles, using battery power and direct communications to Earth, were considered. In addition, the following factors were used in the analyses:

- Communication range to Earth of 280 x 10^6 km
- Battery electrical efficiencies of 35 watt-hours per pound
- High gain antenna (parabolic) of 3 feet in diameter - larger sizes offer better performance, but increase storage and wind loading problems, and require extremely accurate pointing mechanisms because of their narrow beamwidth. Smaller antennas require higher transmitter power to maintain a given data rate.

4.3.3 Evaluation - For our evaluation of the alternatives of the problem we will analyze the effects of SL configuration, telecommunications, mission duration and electrical power options with fixed thermal control concepts. By fixed thermal control concepts we mean that the basic construction of the system - heat pipes, radiator area, thickness of insulation and heat source (radioisotope heaters or battery powered) - is held constant.

In order to minimize the number of thermal control/SL combinations we will eliminate configurations that do not use heat pipes and radiators. As illustrated in Figure 4.3-4, these configurations are either too heavy or elevate the equipment temperature above the operating limit of the batteries. Of the remaining options we will consider only all electrical or all radioisotope heaters because

SURFACE LABORATORY WEIGHT
(THERMAL CONTROL DESIGNED FOR CONSTANT −190°F ENVIRONMENT)



4.3-8-1

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SUMMARY OF THERMAL CONTROL ALTERNATIVES

CASE NO.	GENERAL DESCRIPTION	MAXIMUM EQUIPMENT TEMPERATURE
A1	Electrical heaters, 4 inches of insulation, 4 heat pipes plus 15.5 ft^2 of radiator surfaces.	100°F
A2	Electrical and radiosotope heaters, 4 inches of insulation, 4 heat pipes plus 15.5 ft^2 of radiator surface	111°F
A3	Radioisotope heaters, 4 inches of insulation, 4 heat pipes plus 15.5 ft^2 of radiator surface.	122°F
A4	Electrical heaters, 4 inches of insulation, no heat pipes or radiators.	2 139°F
A5	Electrical heaters, 2 inches of insulation, no heat pipes or radiators.	112°F

Notes:

1 Maximum equipment temperature based on power profile of Figure 4.3–2 and cyclic environment.

2 Temperature beyond the operating limits of the batteries.

4.3-8-2

Figure 4.3–4

~~4.3-8~~

they represent the extremes of the feasible alternatives.

4.3.3.1 Configuration - The internal volume of the SL affects, as shown in Figure 4.3-5, the packaging density of the installed equipment and the useful fraction of SL weight. As payload volume is increased:

- o The packaging density required to install a fixed amount of equipment decreases.
- o The weight of the insulation surrounding the interior of the SL increases. For electrically heated vehicles, the weight available to instruments and
- o their support equipment (telecommunications, data handling, etc.), decreases. As volume increases, the battery and insulation weight for thermal control also increase. These weights cut into the usable part of the SL (instruments, telecommunications, etc.) if the total weight of the SL is kept constant.
- o For radioisotope heated vehicles, the weight associated with thermal control increases primarily with insulation weight. The increase in heater weight is just a few pounds for the range of volumes shown in Figure 4.3-5.

4.3.3.2 Telecommunications - The nominal operating point for both radio links is 25 watts RF; 5 watts for the low rate link, 20 watts for the high rate link. As the total radiated power is increased from this point, the equipment temperature increases. As shown in Figure 4.3-6, the cut-off point for transmitter power is about 30 watts RF for radioisotope heated vehicles. Beyond this point, the equipment temperature is above the operating limit of the batteries (125°F).

For SL using electrical heaters SL weight and equipment temperature are factors. As the total transmitter power reaches about 70 watts, the performance of both radio links becomes high. The low rate link can use a 20 watt transmitter, which increases its data rate to 3 bps; the high rate link can use a 50 watt transmitter, which increases its data rate to 1500 bps. Unfortunately this approach raises the equipment temperature to near the operating limit of the batteries (which is 125°F), and increases the SL weight by over 160 pounds, relative to its nominal operating point.

Operation at low transmitter powers does not significantly reduce the weight of an electrically heated vehicle. As the transmitter power is reduced, the heat dissipated by the transmitter(s) is reduced. As shown in Figure 4.3-2, a given amount of heat is required for operation in a constant -190°F climate. This heat can be obtained from electrical heaters or by equipment operation. One way of supplying the required heat is by relatively high transmitter power - 20 to 30 watts

EFFECTS OF SL VOLUME UPON SL CAPABILITY

Notes:

1 Thermal Configuration A1 – Electrical heaters, 4 in. of insulation 4 heat pipes + 15.5 ft^2 radiator.
Thermal Configuration A3 same except uses radioisotope heaters.

2 Applicable to Configuration A1 only

3 Mission length – 28 hours.

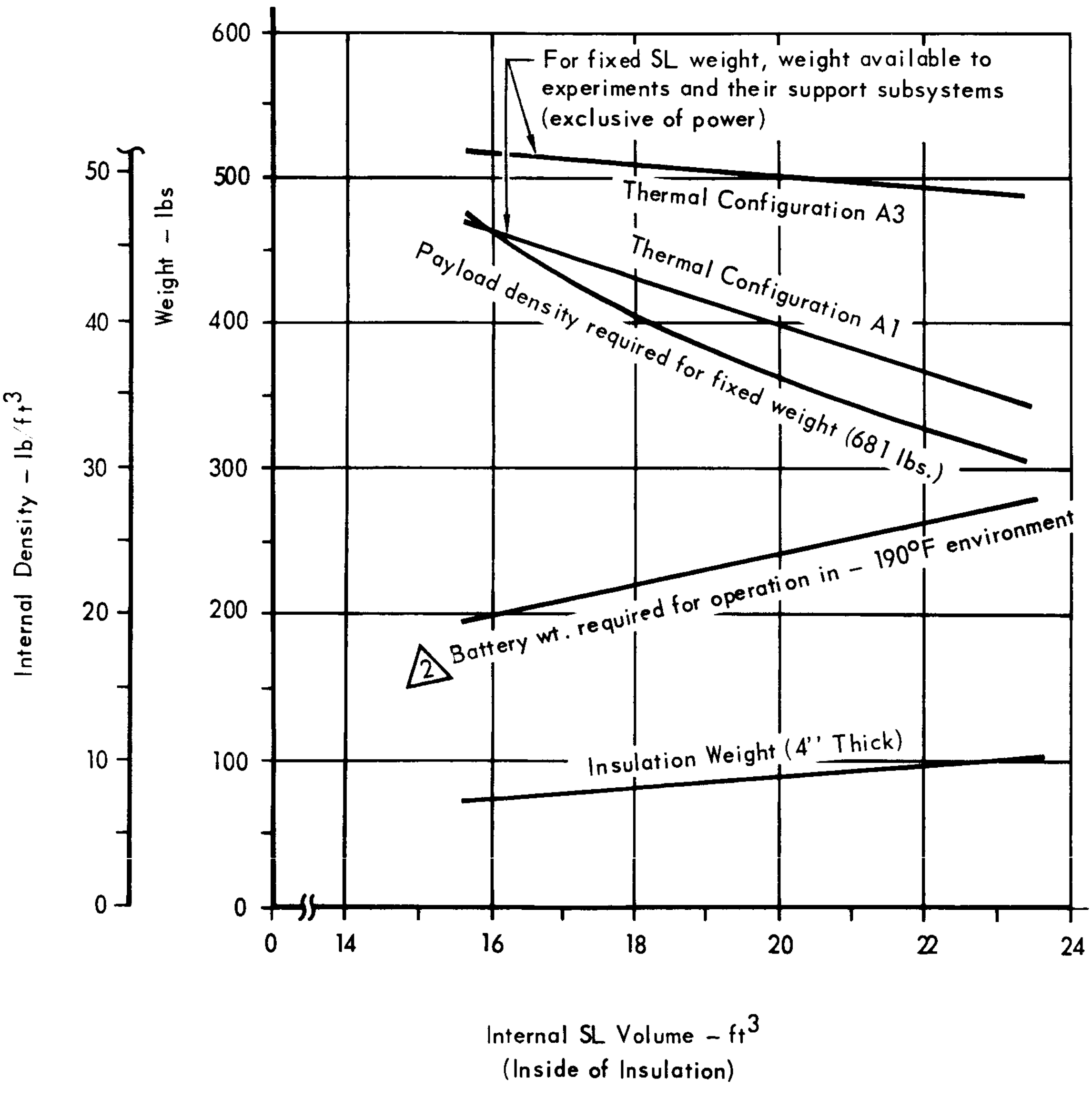
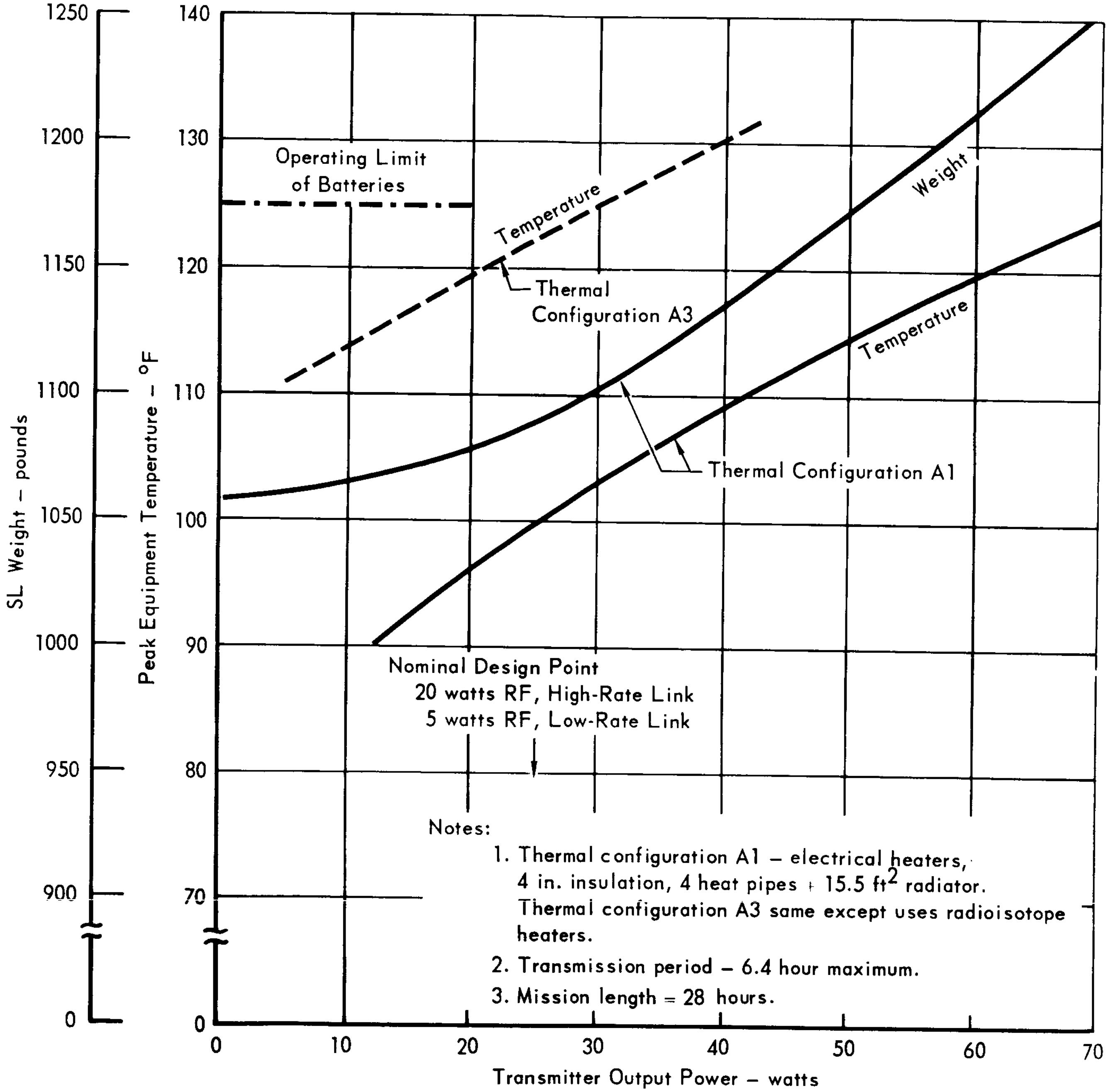


Figure 4.3-5

EFFECTS OF TRANSMITTER POWER UPON SL WEIGHT AND TEMPERATURE
(PEAK TEMPERATURE BASED ON CYCLIC ENVIRONMENT – SEE REFERENCE 4.3-2)



Notes:

1. Thermal configuration A1 – electrical heaters, 4 in. insulation, 4 heat pipes + 15.5 ft^2 radiator. Thermal configuration A3 same except uses radioisotope heaters.
2. Transmission period – 6.4 hour maximum.
3. Mission length = 28 hours.

Figure 4.3-6

RF. Reduction of transmitter power below this point doesn't save an appreciable amount of electrical energy, in a -190°F environment; as transmitter power is decreased, heater power must be increased to maintain adaquate equipment temperature.

4.3.3.3 Electrical Power - As shown in Figure 4.3-7 the SL weight is a function of mission length and percentage of battery contingency. For a mission design life of 28 hours, a 20% contingency in electrical energy increases the SL weight by 52 pounds for battery powered thermal heaters and by 30 pounds for radioisotope heaters.

Another way to view contingency is by consideration of the thermal control/ power interactions. Design of the SL is presently limited by the -190°F ambient temperature requirement of Reference 4.3-1. In practice, we believe that cyclic environments will exist. For cyclic environments, warm days and cold nights, less heater power and energy is required, thus depleting the energy resources of the battery for electrically heated vehicles at a slower rate. An example of this energy savings is shown in Figure 4.3-8. At the end of a nominal 28 hour mission, over 2400 watthours of energy remain if a cyclic environment is encountered. This reservoir can be considered contingency, or, as discussed below, can be used for extended mission operations. Zero contingency results for a radioisotope heated SL because no expendibles are used for thermal control.

4.3.3.4 Mission Duration - The length of the mission, combined with the requirement to design for a continuous -190°F environment, can be equated into SL weight. A SL designed with electrical heaters, 4 inches of insulation, and for a 40 hour mission length is about 110 pounds heavier than one designed for a 28 hour mission. The relationship of SL weight vs. mission length, shown in Figure 4.3-4, is not as dramatic for radioisotope heated vehicles because no expendibles (batteries) are used for thermal control. Electrically heated designs must use battery energy for heater and equipment operation.

An electrically heated vehicle designed for a 28 hour life at -190°F environments has a reservoir of energy remaining at the end of the nominal mission if a cyclic climate is encountered. For the design and conditions of Figure 4.3-8, this reservoir, over 2400 watthours, is sufficient to extend the mission through another night into another transmission period, the 3rd. The length of this 3rd transmission period is a function of how the batteries were conserved through the nominal mission. If a regular 28 hour mission were performed, the third transmission period would last only an hour or two before the batteries were depleted. If the second transmission period were terminated before Earth-set, the energy for

EFFECTS OF ELECTRICAL ENERGY CONTINGENCY UPON SL WEIGHT

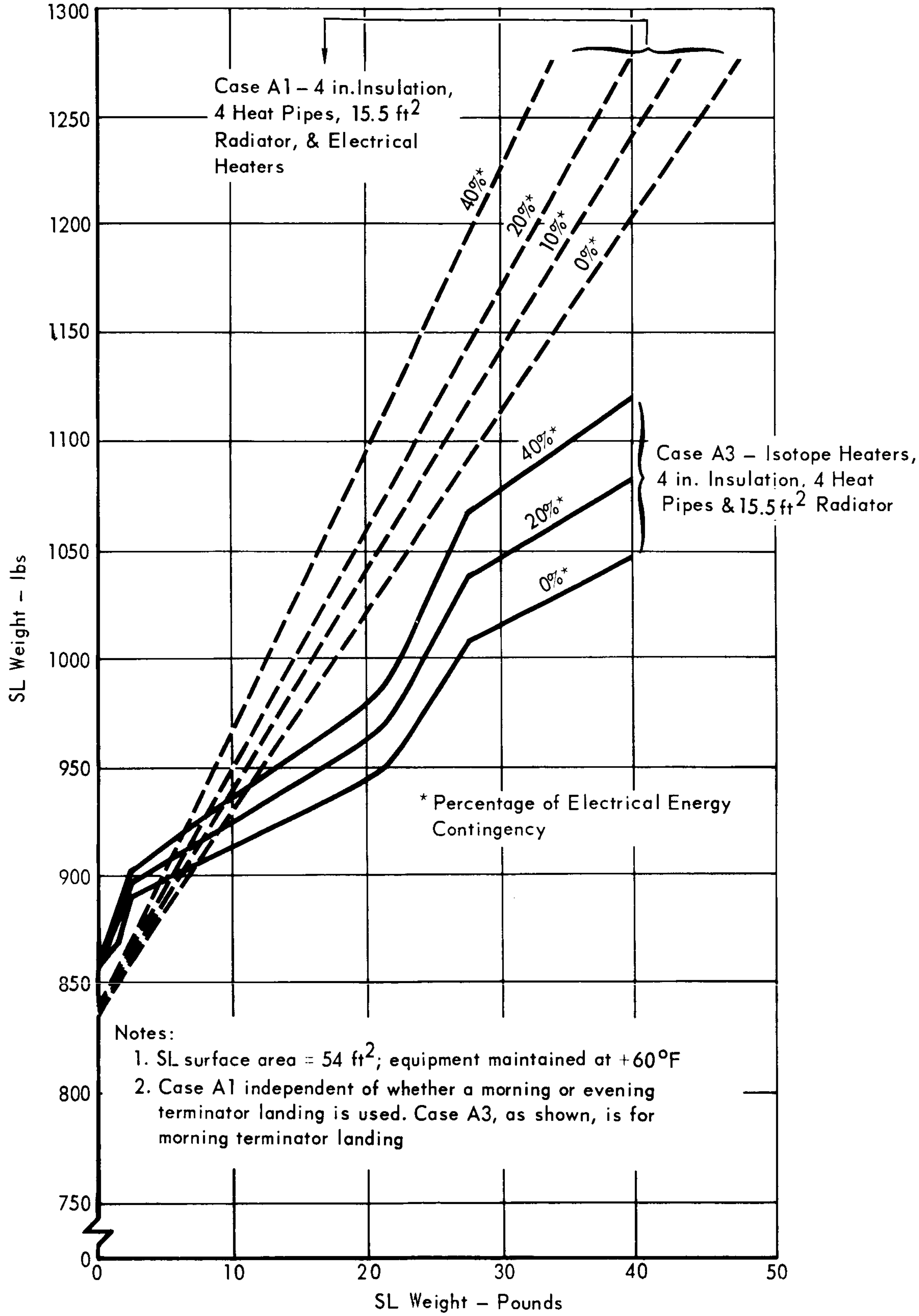
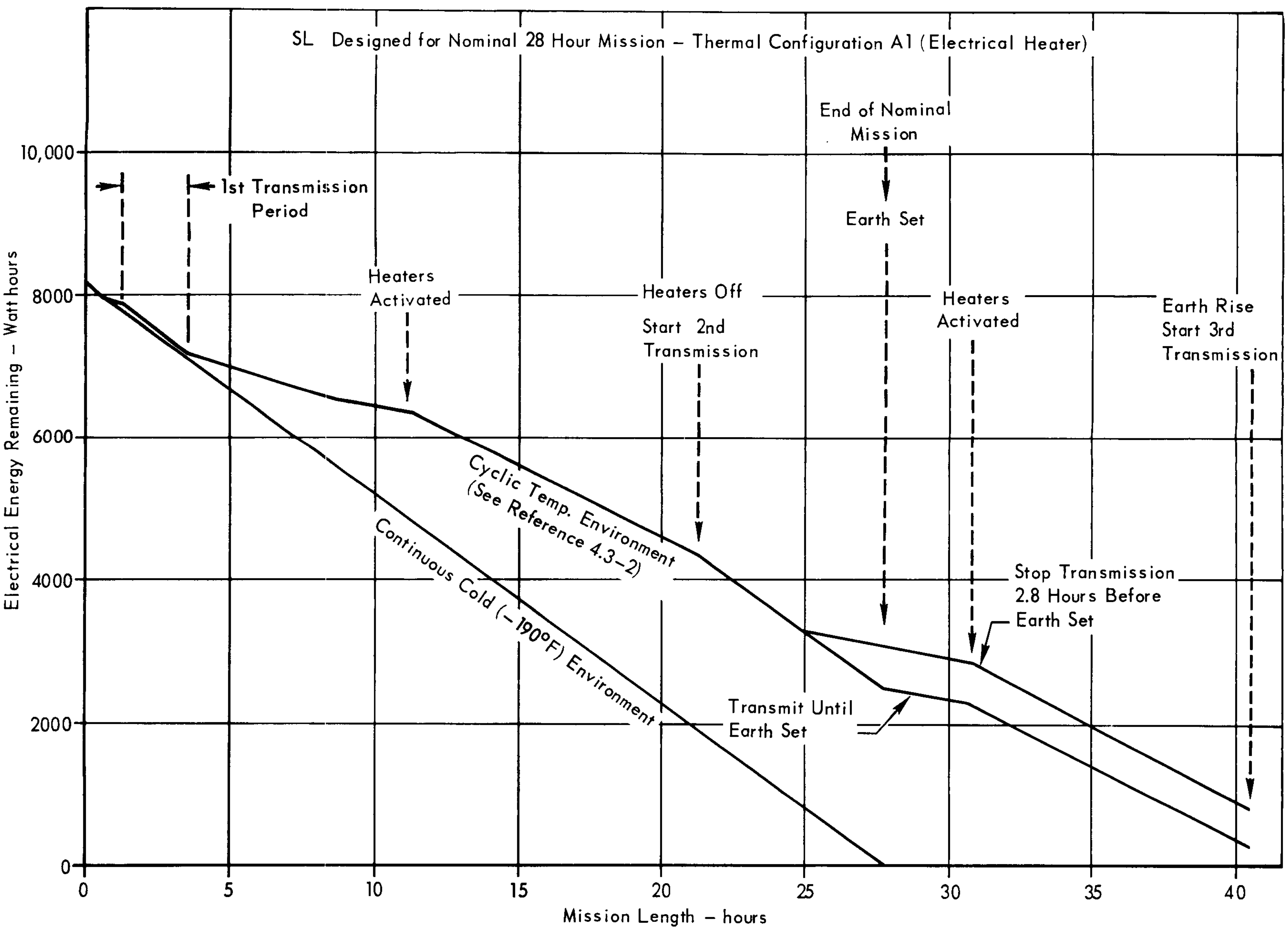


Figure 4.3–7

ELECTRICAL ENERGY CONTINGENCY



4.3–8

4.3–14

the start of the 3rd transmission period would be enough for several hours of transmission.

4.3.4 Conclusions - The thermal control design preferred in Reference 4.3-2 electrical heaters, 4 inches of insulation, heat pipes and radiator - is also best when the overall SL is evaluated. Although it is the heaviest of the feasible alternatives, it allows extended mission operations if a warm or cyclic climate prevails, and adapts to the actual environment because it can control the amount of heat required. Designs using radioisotope heaters are attractive from a weight standpoint, but complicate the SL design because they introduce problems of radioactivity and elevate the equipment temperatures during warm days. However, radioisotope heaters appear to be a good method of adding increments of heat to specific, but limited, portions of subsystems as the requirements for extra heat arise during detailed equipment design.

SL Configuration - The SL should be configured to a minimum size condition, consistent with realistic packaging densities, to minimize heat loss in a constant cold environment. A minimum size configuration reduces the weight of the batteries and insulation required for thermal control, thus increasing the weight available to telecommunications or experiments.

Telecommunications - A transmitter power of 20 watts RF for the high rate radio link represents a reasonable compromise between maximum data capability, and increased payload weight and thermal control problems. Higher transmitter power increases the battery weight and raises the average equipment temperature. Lower transmitter power offers no substantial weight benefit because the reduction of electrical energy required by the transmitter is compensated by an increase in electrical energy required for thermal control in -190°F climates. In effect, the transmitter, at the 20 watt RF point, dissipates, in conjunction with the other equipment, the heat required for thermal equilibrium in a constant cold environment.

The low rate radio link should be operated with a transmitter power of 5 watts RF. Although this admittedly limits the data rate to an extremely low number, .5 bps, higher power increases weight and equipment temperature. In addition, the 5 watt RF point allows the transmitter design to be solid state, which is a reliability advantage.

Electrical Power - An electrical power subsystem designed to a zero energy contingency for operation in a continuous -190°F environment is preferred. This design for zero energy reserve at -190°F climates yields in excess of 25% reserve if a cyclic environment is encountered. Since a cyclic environment is more probable than a constant cold one, an allocation of energy reserve for -190°F conditions not only increases the SL weight but provides an exorbitant amount of reserve if a cyclic climate prevails. For example, if a 20% energy reserve is allocated for -190°F conditions and a nominal 28 hour mission, the SL weight is increased by 50 pounds, and a 50% energy reserve exists at the end of the basic mission if a nominal cyclic climate is encountered.

Mission Duration - A basic mission length of 25 to 30 hours, combined with a landing near the morning terminator, satisfies the mission constraints. A design for morning landings is over 100 pounds lighter than a design for evening landings. In addition to its weight advantage, a morning terminator landing allows immediate communications to Earth after landing, with large amounts of data transmitted before a Martian night is encountered.

The second aspect of mission duration occurs during the actual mission. With a design for a nominal 28 hour mission and continuous -190°F ambient temperatures, there is a high probability of electrical energy remaining at the end of the basic mission if a mild climate prevails. This remaining energy may be sufficient to carry the SL through another Martian night to another data transmission period, thus extending the basic mission. The reservoir of energy can be controlled somewhat during the actual mission by the duration of the data transmission periods. If, during the 2nd period of data transmission, it is decided to attempt another data transmission after the nominal mission is completed, the 2nd transmission may be terminated prematurely to conserve the batteries. An illustration of the effects of changing the length of data transmission upon extended mission operations was shown in Figure 4.3-8.

REFERENCES

4.3-1 "1973 VOYAGER Capsule Systems Constraints and Requirements Document - Revision 2", JPL, 12 June 1967

4.4 IN-FLIGHT/LANDED MONITORING AND CHECKOUT - A Surface Laboratory Systems in-flight and Mars landed status monitor/checkout/control plan has been developed in parallel with Surface Laboratory (SL) equipment design and has been integrated into the overall Mission Support Plan. The automatic monitor/checkout activity includes:

a. Continuous passive monitoring from Earth launch through Mars orbit (interplanetary cruise).

b. Subsystems and science instruments activation and checkout of landed mission readiness prior to Flight Capsule/Spacecraft separation.

c. Continuous monitoring of cruise parameters during the Capsule Bus (CB) orbital descent, but at 10 times faster sampling rate.

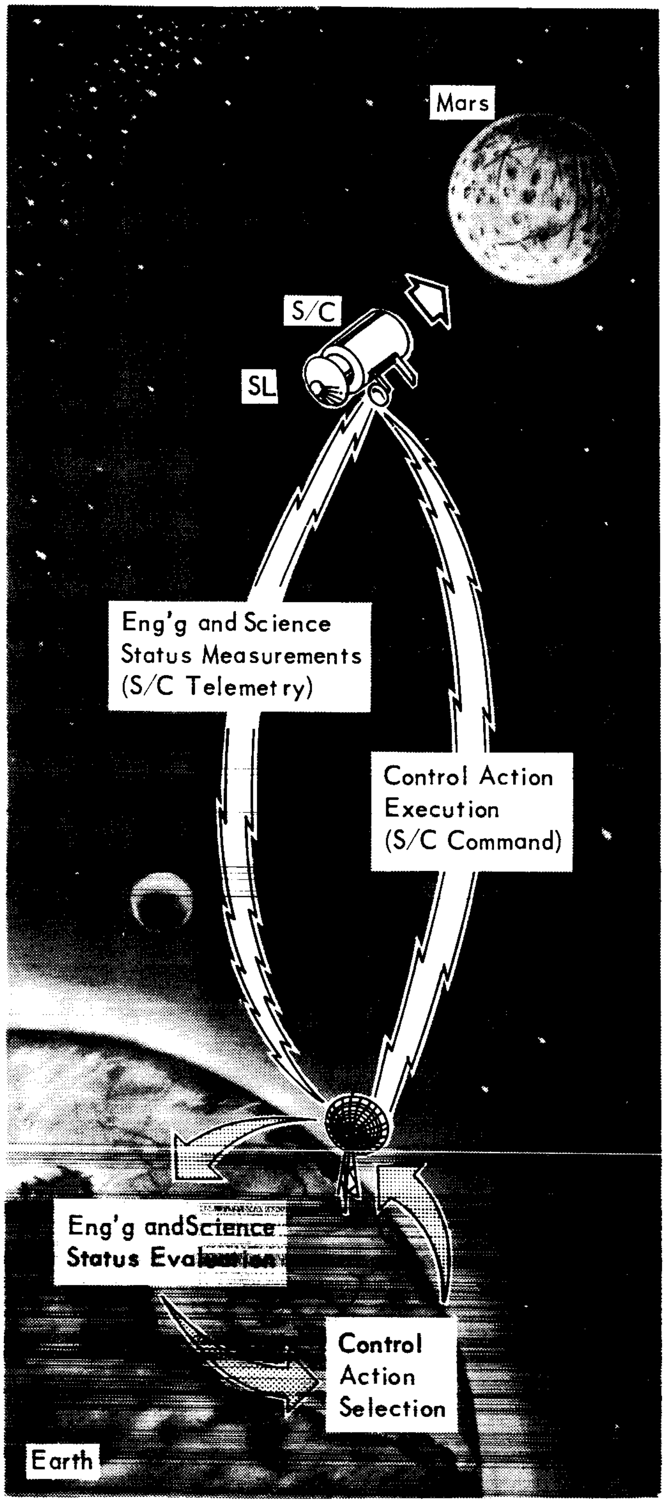
d. Continuous monitoring of each subsystem and science instrument during its period of activity in the landed mission profile.

All of this monitor/checkout data is generated automatically and ultimately telemetered to the Earth stations, where the SL mission operations personnel analyze and judge the integrity and/or performance of the engineering and science subsystems. The ground operations personnel have continuous recourse to corrective/preventive equipment control actions or mission sequence modifications, via Earth command, except during the CB orbital descent and SL post-landed out of sight periods. During cruise and pre-separation checkout the back-up command capability is to the Spacecraft-to-SL; after landing the commands are received by the SL itself.

Figures 4.4-1, -2, -3, and -4 present functional descriptions of the status monitor/checkout/control activities for all SL mission phases. It is noted that data is continuously gathered on the SL subsystems; the cruise parameter monitoring continues in orbit both before and after the pre-separation checkout period. After landing, the operational status of certain engineering subsystems and science instruments are continuously monitored; these data are stored and transmitted to earth during available communications reception periods. 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 D 4.5. Except for the landed phase, this status monitor/checkout/control activity is performed sequentially with similar functions for the Capsule Bus and Entry Science Package.

4.4.1 Test Purpose and Selection Criteria - The purpose of in-flight/landed monitoring and checkout is to maximize the probability of mission success. Whether or not mission objectives or operations will be changed will depend on the condition of the SL equipment, as determined from the monitor and checkout data. The test

SLS 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
- Existing Dangerous Conditions
- Impending Failures or Failures
- Ground Equipment Accuracy Verification

Control Action Selection (Mission Operations Personnel)

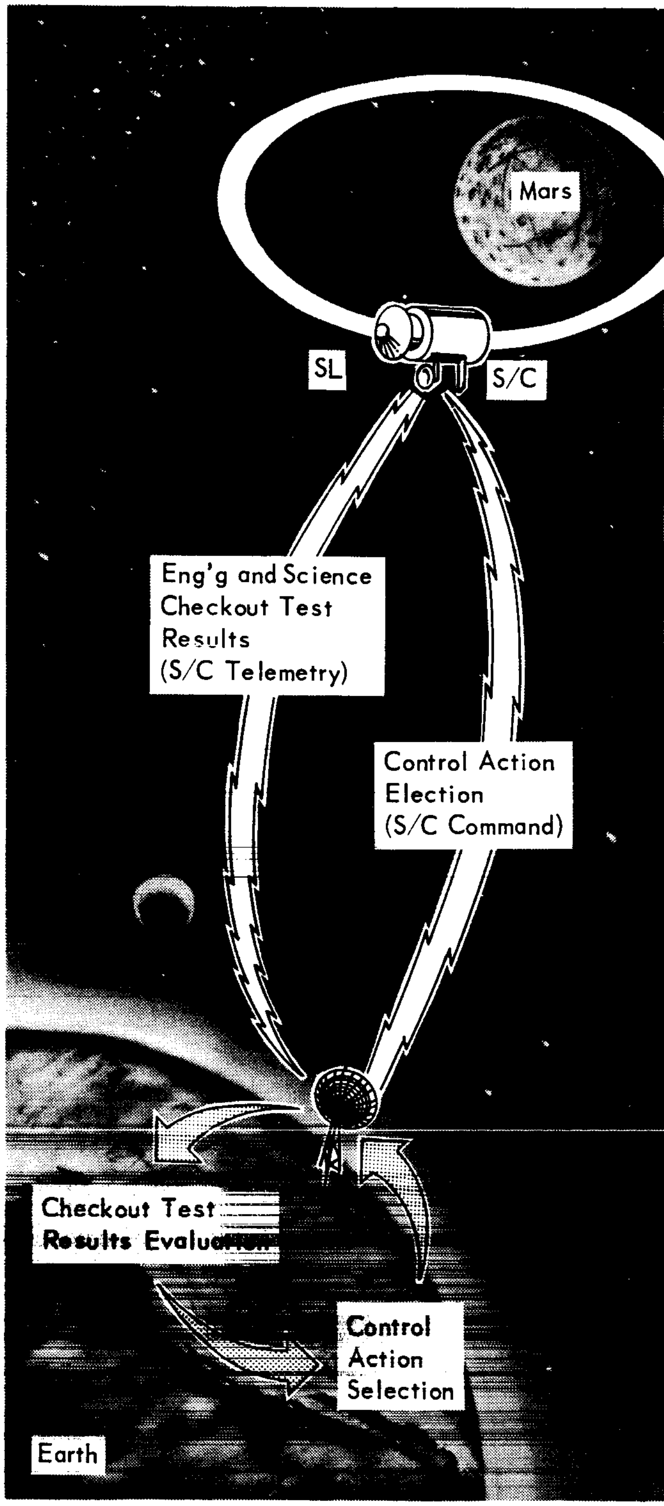
- Choose Equipment Corrective/Preventive Action
- Select Mission Contingency Plan
- Check Ground Equipment

Control Action Execution

- Send Command to Spacecraft -to-SL

Figure 4.4–1

SL PRE-SEPARATION CHECKOUT TEST/CONTROL PLAN



Eng'g and Science Checkout Test Results

- Eng'g and Science Equipment Test Inputs and Responses Monitored
  - Test Sequence Automatically Controled by the SL Test Programmer

Checkout Test Results Evaluation (Mission Operations Personnel)

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

Control Action Selection (Mission Operations Personnel)

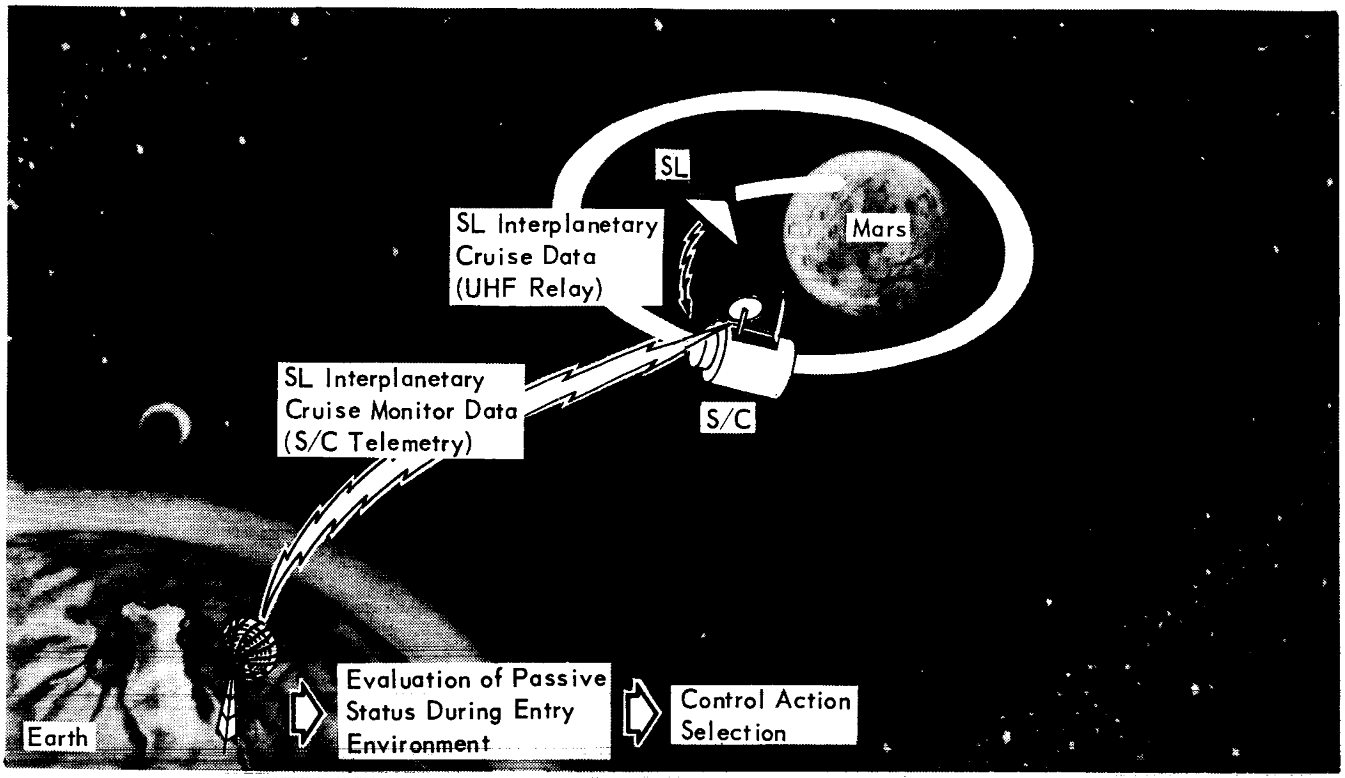
- Select SL Mission Contingency Plan
- Change SL Mission Sequence
- Repeat Particular Checkout Test

Control Action Execution

- Send Command to Spacecraft–to–SL Subsystem
- Send Command to Spacecraft–to–SL Test Programmer

Figure 4.4–2

SL MARS ORBITAL DESCENT PHASE MONITOR PLAN



Mars Orbital Descent Monitor Parameters (UHF Relay and S/C Telemetry)

- Continuous Passive Monitoring
- Same Parameters as Monitored during Interplanetary Cruise, but at x 10 sample rate
- SL Cruise Data to CB Telemetry (vis hardline); CB to S/C (via UHF Relay Radio); S/C Telemetry to Earth.

SL Orbital Descent Status Evaluation (Mission Operations Personnel)

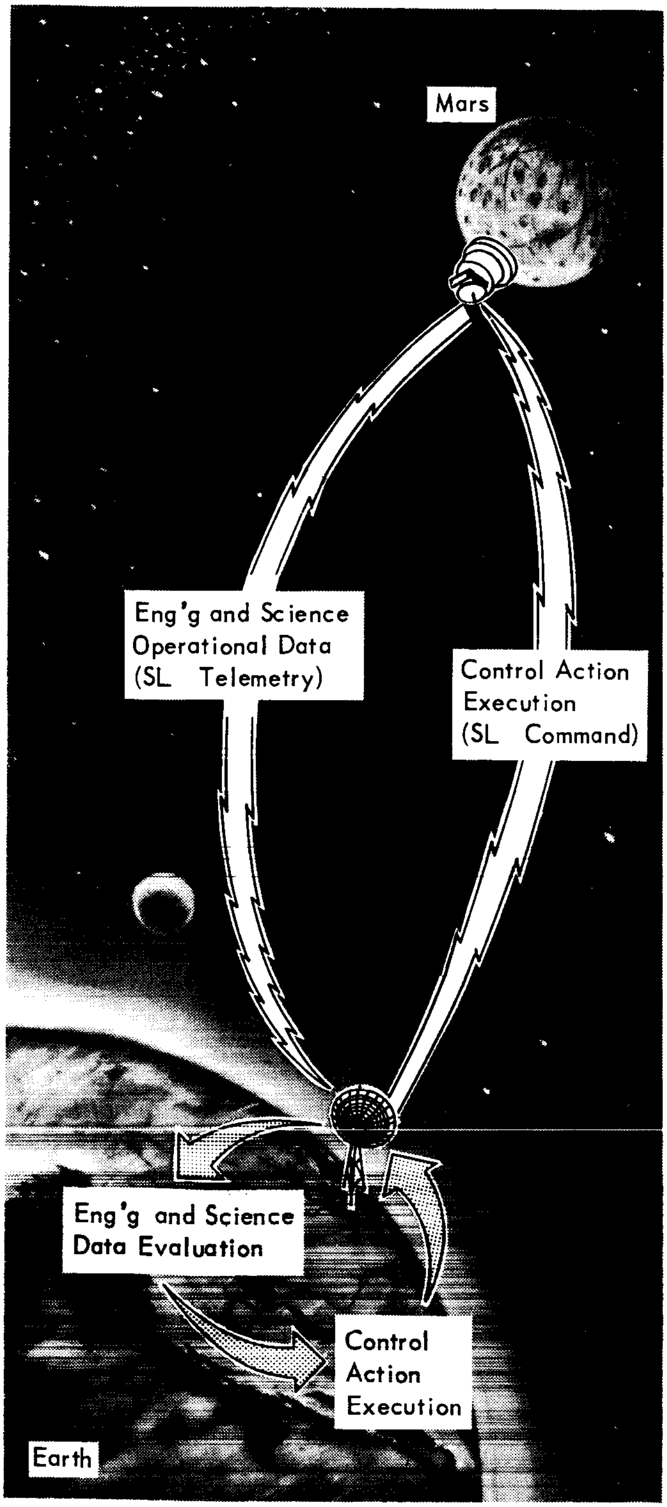
- Engineering and Science Integrity Verification
- Isolate Failures

Control Action Selection (Mission Dependent Personnel)

- Select Immediate Post-Landing Command Activities

Figure 4.4-3

SL LANDED MISSION MONITOR/CHECKOUT/CONTROL PLAN



Eng'g and Science Operational Data Results

- Eng'g and Science Data to Earth During Periods of Available Earth Reception

Eng'g and Science Operation Evaluation (Mission Operations Personnel)

- Operational Performance Verified
- Fault Isolation
- Remaining Mission Capability/Capacity
- Ground Equipment Accuracy Verification
- Cursory Surface Model Construction to Provide Pre-Mission Planning for Second Planetary Vehicle

Control Action Selection (Mission Operations Personnel)

- Update Experiment Sequence
- Change Engineering Equipment Operation
- Recall Checkout Test used In-Flight

Control Action Execution

- Send Command to Subsystem
- Send Command to SL Test Programmer

Figure 4.4-4

selection criteria establish the tests necessary to perform this function with proven engineering techniques.

Interplanetary Cruise Monitoring - From Earth launch through Mars orbit a special SL cruise telemetry commutator continuously monitors:

a. Temperatures and pressures of the subsystems and science instruments.

b. Thermal control and electrical power subsystems conditions.

These data are required to maintain confidence of SL survival during its inactive transit to Mars. The confidence level of the received data is established by having the telemetry unit also monitor its own operating conditions. These data allow early detection of impending problem conditions or failures and guide the ground personnel in selecting the best command control action.

Pre-Separation Checkout Tests - Prior to the Flight Capsule/Spacecraft separation the SL 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 on-board, pre-programmed, automatic Test Programmer. Equipment operational parameters and test responses are evaluated by the mission operations personnel and are used to:

a. Determine the operational performance of engineering and science equipment prior to landed mission commitment.

b. Isolate faults to the subsystem and science instrument module level.

c. Select redundant components or functional back-up modes of operation.

d. Modify landed mission profile or event sequence.

e. Provide correlation data to facilitate post-flight analyses and to compare with pre-launch calibration data.

This pre-mission performance evaluation/update capability provides mission success enhancement for both data quantity and quality from the landed mission initiation.

The test details are constrained by the requirements for minimizing mission reliability degradation, on-board 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.

Monitoring During Capsule Bus Orbital Descent - From completion of the dynamic checkout tests until Capsule Bus surface landing the SL equipment is effectively again in an inactive state. During this period the cruise telemetry commutator is again employed to monitor the equipment. The sampling rates during this phase are increased (x 10) to detect the transients in the SL conditions due to the CB atmospheric entry environment. Monitoring is sufficient during the CB orbital descent because no command capability exists for corrective actions in this phase.

Landed Mission Operation Monitor - Critical engineering parameters are also monitored for each subsystem and science instrument during their active landed mission periods. The science instruments engineering parameters (i.e., temperature, detector voltage, carrier gas pressure, etc.) are used to:

a. Establish confidence level of received experiment data.
b. Isolate cause of instrument fault.
c. Assess remaining instrument operational lifetime and subsequent experiment program changes.

The monitored thermal, electrical and mechanical conditions and events of the engineering subsystems help establish safe equipment environments, required equipment mode changes, and remaining mission capability and capacity.

4.4.2 Subsystem/Instrument Design Implications - The decision to include the continuous interplanetary cruise monitoring capability has resulted in the design of a special purpose cruise telemetry commutator and equipment interfaces with this unit. This commutator has two modes since it is also employed during the Capsule Bus orbital descent phase at a ten times faster commutation speed. Also, this unit's output is connected to the Capsule Bus telemetry subsystem so that the data may be sent via the UHF Radio Subsystem during the CB orbital descent phase. This small, separate telemetry unit design approach provides low power consumption over a long period and best overall mission reliability.

The decision to include the on-board dynamic checkout test capability has resulted in a parallel design requirement for subsystem and science instrument compatibility with the required synthetic test input stimuli, the test programmer, and the telemetry subsystem. Due to this early integration effort 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. It is noted that this on-board, 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 SL equipment is required to interface with the SL test programmer, which automatically commands the elements into their test modes 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.

For the landed mission the telemetry subsystem has modes for continuous, periodic, and aperiodic engineering operational and science data acquisition, storage, or transmission. These different modes are required for efficient data gathering and handling.

4.4.3 In-Flight/Landed Monitor and Checkout Test Descriptions/Discussion - Figure 5.4-2 of Section B 5.4 (SL telemetry instrumentation list) presents the monitor/checkout data which are transmitted to Earth during all SL mission phases. Also shown are the accuracy of these measurements and their sampling periods and rates. Figures 4.4-5, -6 present the functional descriptions and test objectives for the in-flight checkout tests performed on each engineering subsystem and science instrument, respectively. The actual subsystem sensor data gathered during these tests is listed in the Figure 5.4-2 of Section B 5.4, under the heading of pre-separation checkout.

The SL checkout tests are conducted in a pre-programmed, automatic step sequence for 2 hours, starting at 21 hours and 52 minutes prior to separation from the Spacecraft. The Capsule Bus (122 minutes duration) and Entry Science Package (6 minutes duration) tests are completed before these tests are initiated. This integrated test phasing and timeline are designed to allow adequate time for selecting any desirable overall mission updates or re-run particular tests. It is noted that the test results of one system can affect the mission decisions for other systems; for example, a change in Capsule Bus de-orbit and entry trajectory changes the SL touchdown time and position. For this reason the equipment corrective and mission update actions cited in the Figures 4.4-5, -6 are examples. Note that continuous data on the SL subsystems are available in orbit because the cruise parameters are monitored both before and after the pre-separation checkout test period.

All the SL in-flight checkout tests are performed on Spacecraft power. The 2 hour period is determined by a 1 hour High Gain Antenna Control Subsystem gyros warm up period followed by a one hour drift rate test. All other engineering subsystems and science instruments tests are conducted concurrently in such a manner as not to exceed the 200 watt Spacecraft power limitation.

The in-flight checkout test duty cycle and turn on/off reliability considerations for all SL elements has been included in the overall SL mission reliability

SURFACE LABORATORY ENGINEERING SUBSYSTEMS PRE-SEPARATION CHECKOUT TESTS

TEST NAME	TEST DESCRIPTION	TEST OBJECTIVE/CORRECTIVE ACTION/SPECIAL REMARKS
S–Band Radio Subsystem 1. Low Rate (16-FSK) Radio Link Check	• Test word generator drives frequency synthesizer • Transmitter radiates RF into a dummy load – this eliminates need for RF absorbent antenna cap for RFI control • The frequency synthesizer output, transmitter S-Band frequency and power are measured	• Verifies SL low rate data link or fault isolates • Loss of this link is not catastrophic since all experiment data but cameras are also interleaved on high rate link • Transmitter S-Band frequency measured by mixing with Spacecraft transmitter frequency so that counting is done in low MHz range.
2. High Rate (PSK) Radio Link Check	• Test word generator drives phase modulator • Transmitter radiates RF into a dummy load – this eliminates need for RF absorbent antenna cap for RFI control • The phase modulator output, transmitter S-Band frequency and power are measured	• Verifies SL high rate data link or fault isolates • With failure only low rate engineering and science data available (i.e., no facsimile data) – can plan not to attempt visual imaging and high rate transmission and use subsequent extra battery capacity to run low rate experiments for longer periods • Transmitter S-Band frequency counted same way as mentioned above.
Command Subsystem 1. Detector/Decoder Check	• Test word generator inputed to detector • Decoder output word monitored	• Verifies detector/decoder and allows fault isolation with receiver • Receiver not checked in-flight because of RF command generator complexity • If detector/decoder failed then update landed mission sequence estimate because it cannot be changed after separation from Spacecraft
High Gain Antanna Control Subsystem 1. Gyro Drift Test	• Warm up gyros and then operate in gyro uncaged mode while spacecraft limit cycles in sun/canopus attitude hold mode • Ground personnel determine gyro drift characteristics	• If drift rates are excessive or gyro failed then sequence first earth acquisition attempt in the RF monopulse tracking mode
2. Gimball Drive/Readout Units Check	• Turn on gimbal servo electronics and command small ($<1^{\circ}$) angle maneuver in each axis • Monitor gimbal position transducer outputs	• Verifies gimbal drive control and position readout electronics
Sequencer and Timer Subsystem 1. Operational Verification	• Turn on and run stored program in fast time – all output drivers are exercised • Test programmer functions as OSE during test to hold all squib circuits in the "safe" test condition.	• Test verifies proper time generation and all output driver circuit closures
2. Memory Check and Update	• Readout memory and update as required • Readout memory after update (if update action performed)	• This test verifies memory retention during cruise • The update words depend in many instances on results of other checkout tests • Memory readout after update verifies proper update
Science Data Subsystem 1. Operational Verification	• The unit is turned on and all output drivers are exercised • Test programmer functions as OSE during test to hold all squib circuits in the "safe" test condition	• This test verifies capability of experiment actuation and mode control
Telemetry Subsystem 1. Linearity Test	• Calibration voltages applied to all differential amplifiers and A/D converter outputs monitored • Multi-level reference signals provide linearity (accuracy) test of these elements over their entire input ranges	• Test data indicates how to bias interpretation of received data • An abbreviated version of this test is also performed during landed mission to provide confidence level of mission gathered information
2. Memory Check and Update	• Readout memory and update as required • Readout memory after update (if update action performed)	• Verifies memory retention during cruise • The update words depend in many instances on results of other checkout test results • Memory readout after update verifies proper update
Data Storage Subsystem 1. Data Storage Function Verification	• The test picture taken on each facsimile camera is stored in this subsystem and delayed transmitted at lower than data acquisition bit rate	• Verifies data storage and delayed transmission at designed bit rate
Thermal Control Subsystem 1. Performance Monitor	• Monitor same radiator and cold plate temperatures and heat pipe fluid pressures as monitored during cruise • Sample rate is increased over cruise phase to determine thermal interactions of test operations	• Verifies temperature environment during test and thermal control element integrity
Electrical Power Subsystem 1. Batteries Condition Monitor	• Monitor same battery temperature, output voltage/current, and charging current as during cruise	• Verifies battery charge state, cell status, and detects any self discharge

Figure 4.4–5

SURFACE LABORATORY SCIENCE INSTRUMENTS IN-FLIGHT CHECKOUT TESTS

TEST NAME	TEST DESCRIPTION
Visual Imaging Facsimile Camera Calibration (2 Cameras)	• Turn cameras and test lamps on. • Take one low resolution 300 element x 50 lines picture per camera. • Monitor camera engineering and test lamp parameters.
Atmospheric Properties Check of All Pressure, Temperature, Humidity, and Wind Velocity Sensors	• All sensors are turned on and measurements of the ambient conditions are m
Alpha Spectrometer	• Calibrated test sources implanted on detectors – test sources do not interf • Conduct 1 sample of both alpha and proton spectrums. • Monitor pulse height analyzer channels engineering parameters.
Gas Chromatograph	• Turn sensor on and monitor engineering parameters.
Metabolism I and II	• Turn on instruments in calibrate mode – no disturbance or checks of cultur • In Situ modules monitor background radiation.
Life Detector (Growth)	• Turn on instruments in calibrate mode – no disturbance of culture chambers
Spectroradiometer Calibration	• Turn on and activate special broadband test light source.
Sample Acquisition and Processing 1 Subsurface Probe Check	• Cycle mechanical bellows pump (soil gas sample collection mechanism). • Monitor sensor output which measures soil temperature. • Deployment mechanism is not perturbed.
2 Soil Sampler	• None.
3 Sample Processor	• Monitor gas supply pressure.

Figure 4.4–6

	TEST OBJECTIVE/CORRECTIVE ACTION/SPECIAL REMARKS
	• Verifies camera operation and calibrates photodetector (i.e., 6 bit gray code calibration).
ıitored.	• Temperature sensor outputs can be compared with engineering temp. gagues in near vicinity. • Electronic integrity of humidity and wind velocity sensors is verified. • Verifies pressure transducer outputs in known orbital vacuum environment. • Verifies all detectors and interface electronics.
with landed data.	• Verifies bandwidth integrity of detectors. • Verifies analyzer operation and all interface electronics.
	• Verifies electronics and carrier gas pressure integrity. • With any carrier gas leakage, lifetime can be re-estimated and operational sequence modified.
chambers.	• Checks out all sensor electronics. • Verifies integrity of In Situ radiation detector elements. • Checks out all sensor electronics. • Compare data readout scan of 40 wavelength bands with known test light wavelength spectrum.
	• Verifies gas sample gathering pump and soil temperature sensors. • With any gas leakage, lifetime can be re-estimated and operational sequence modified.
	• Not desirable to perturb deployment mechanism for test. • With gas leakage, lifetime can be re-estimated and operational period modified.

calculations. It has been determined that the change in probability of equipment failure is less than one percent as a result of these test operations. This change is insignificant relative to the mission success enhancement available as a result of the test information and command capability to optimally update the mission sequence and equipment modes.

The Figure 5.4-2 of Section B 5.4 also shows the engineering parameter monitor schedule on the surface. The thermal environment, thermal control/electrical power subsystems, and continuously operating science instruments are monitored continuously. All other subsystems and science instruments are monitored only during their active periods. The appropriate engineering status parameter channels are automatically selected by the telemetry subsystem when it switches modes to acquire or transmit the experiment data planned for these periods. Note the monitoring of such events as pyrotechnic actuations and mechanism deployment, which could not be checked out in the pre-separation tests.

4.4.4 Monitor/Checkout Test Data/Command Interfaces - The data generation/gathering techniques and rates differ, as does the command interface to perform status control, for each of the SL mission phases. Figures 4.4-7 and 4.4-8 present functional data and command interface diagrams, respectively, for the SL for all mission phases.

Data Interface Description - From launch to pre-separation checkout tests, the SL cruise telemetry commutator is the only SL data source. This data is hardlined to the Spacecraft, which transmits the data to Earth.

The SL Test Programmer outputs result in accomplishing the tests of Figures 4.4-5 and 4.4-6 according to the pre-programmed two hour schedule. A detailed description of this test programmer is given in Section C 2.2. The Test Programmer first commands the SL Telemetry Subsystem into the checkout mode to select all equipment checkout test parameter channels, which include test stimuli and programmer outputs. The Test Programmer then sequences the equipment on and appropriately cycles the internal equipment test stimuli (i.e., turns on target light lamp, switches output into dummy load, activates "test word" generator, etc.). The Test Programmer outputs also interface with the normal mission equipment control units, the Sequencer and Timer and Science Data Subsystem. This allows maximum utilization of these units' output drivers to control the test operation of all equipment. The Test Programmer output time delay between successive steps in any one test are based on estimated times for equipment stabilization; for example, the High Gain Antenna gyros spin motor power is applied one hour after gyro heaters turn on for warm-up. The start/stop times of each test are scheduled so that equipment test operation

SL MONITOR AND CHECKOUT DATA INTERFACE DIAGRAM

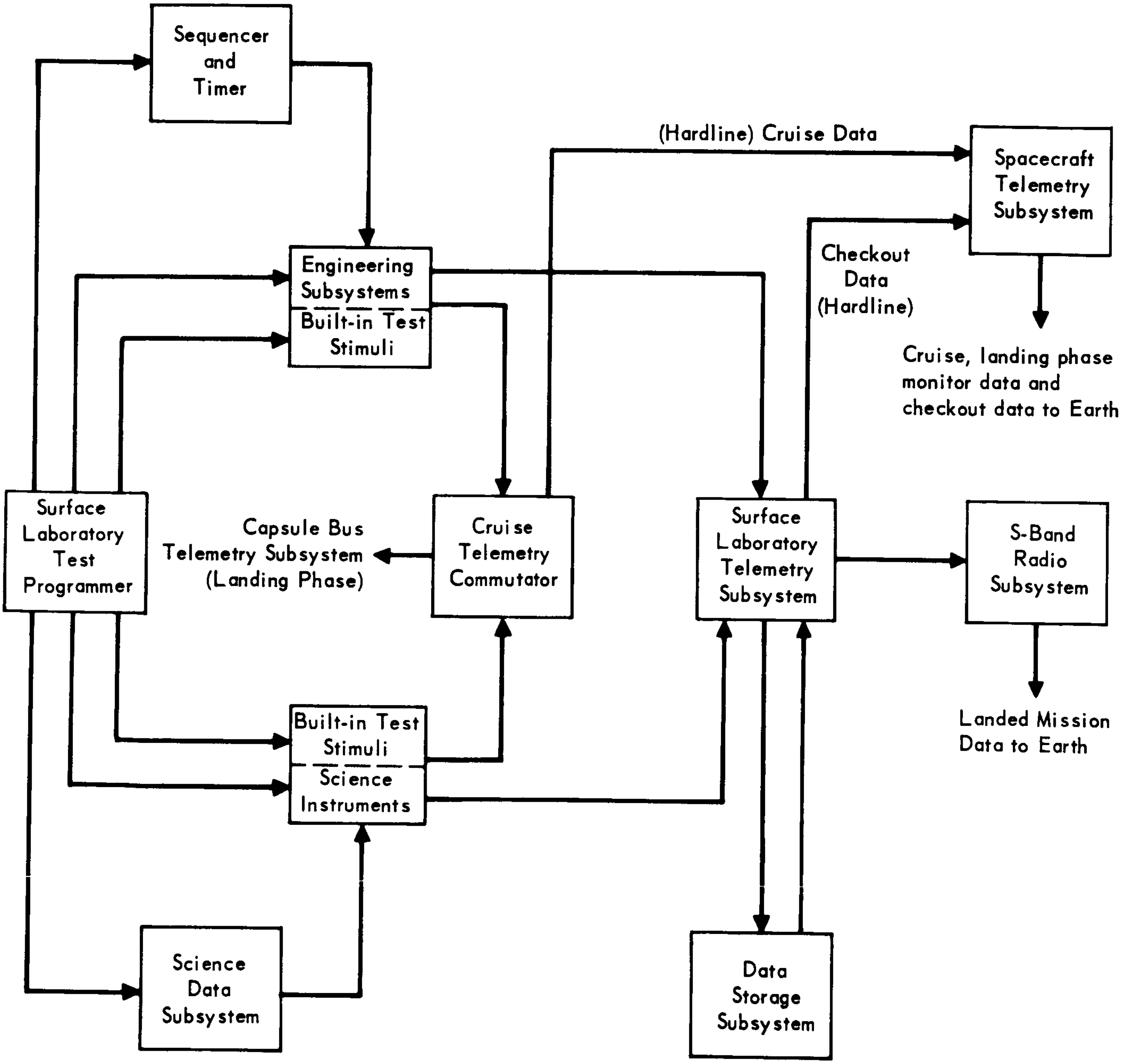


Figure 4.4-7

SURFACE LABORATORY MONITOR AND CHECKOUT COMMAND INTERFACE DIAGRAM

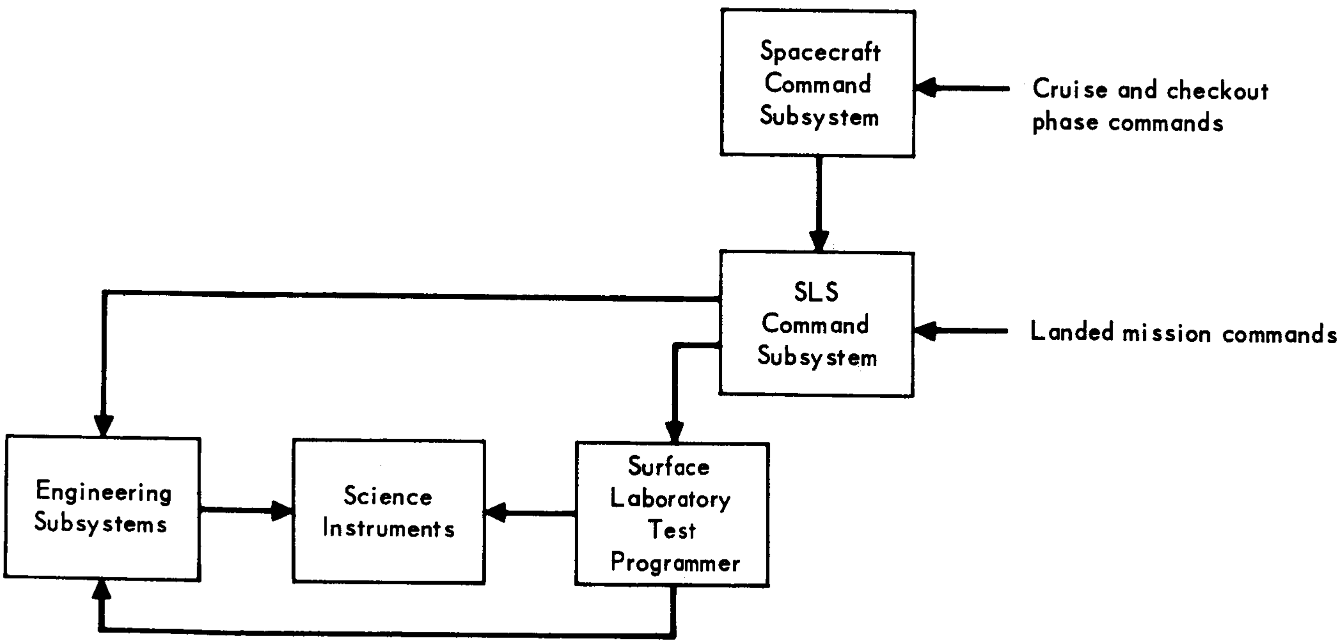


Figure 4.4-8

power level does not exceed the 200 watt spacecraft power delivery limitation. All the checkout data is sent hardline from the SL telemetry subsystem to the Spacecraft telemetry subsystem, which controls transmission to the Earth station.

During the CB orbital descent period the sampling rate of the SL cruise telemetry commutator is increased by a factor of ten. This data is interleaved with the CB telemetry subsystem data and relayed to the Spacecraft by the Capsule Bus UHF Radio Subsystem. The Spacecraft then transmits this data to Earth.

All landed mission monitored equipment data is transmitted directly to Earth by the Surface Laboratory S-Band Radio Subsystem, during periods of available Earth reception.

Command Interface Description - The SL mission is designed to the automatic from launch through post-landed. However, a command change capability is designed to safeguard its status and/or modify its mission. Figure 6-12 of Section C 6 presents all the SL command word capability from Earth to exercise the subsystems and science instruments or update subsystem memories for mission sequence changes. The Figure 4.4-8 presents the command interfaces for each SL mission phase.

During the cruise and pre-separation checkout phases all back-up commands to the SL are hardlined from the Spacecraft via the single interconnect as shown. This single interconnect minimizes interface wiring between these two systems. The design approach shown also allows all the landed mission command word capability to be available during the cruise and checkout phases. The cruise telemetry commutator speed can also be increased during cruise, because this unit's dual speed modes capability is designed for faster sampling during the CB orbital descent phase.

The Test Programmer is capable of being completely updated. 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 first test data shows any pre-programmed estimated times inadequate. The Test Programmer can also be employed to increase equipment temperature during cruise by equipment turn-on with subsequent power dissipation and warming. An early checkout test or routine can be initiated via command to the Test Programmer (hardline from Spacecraft).

No SL command capability exists from the time of Flight Capsule/Spacecraft separation to Capsule Bus touchdown (i.e., Capsule Bus orbital descent and entry flight time).

All post-landed SL commands are directly received and distributed by the SL

Command Subsystem. It is to be noted that the Test Programmer can be reactivated during this mission phase.

4.5 INDEPENDENT DATA PACKAGE STUDY - The purpose of an Independent Data Package (IDP) is to add functional redundancy to the VOYAGER and thereby improve the probability of obtaining some diagnostic and surface environmental data. This study was conducted to examine the utility of an IDP as an adjunct to the entry and landing portion of the first VOYAGER mission. The parameters included in this study were reliability, weight, development risk, state of the art limitations, IDP integration into the Flight Capsule, and effectiveness.

4.5.1 Summary - The usefulness of an IDP was evaluated by adding, as improvements, an equivalent weight to the Flight Capsule. The Flight Capsule with an IDP and the Flight Capsule with improvements were compared using reliability and system effectiveness analysis. In addition, the IDP development problems and its installation into the Flight Capsule were investigated. As a result of this study, the IDP was not incorporated into our preferred design.

4.5.2 Requirements and Criteria - Special constraints are established in references 4.5-1 and 4.5-2. The constraints which affect this study are given in paragraph 4.1.2.3.1 of the constraints document (Reference 4.5-1) and paragraph 3.1.3 of the General Specification (Reference 4.5-2).

The IDP concept was evolved and investigated in accordance with these paragraphs. Achievement of a Flight Capsule landing, performance of entry science experiments, and performance of landed science experiments are the competing characteristics considered in our analysis of the IDP concept. Note that performance of landed science experiments is fifth priority and also the lowest priority of the competing characteristics which applied to the IDP.

4.5.3 Design Considerations - The limitations imposed by the IDP design were evaluated to determine the overall impact on the Flight Capsule. This evaluation is presented in the following subsections.

4.5.3.1 Preferred Design Description - The IDP subsystem would monitor critical Capsule Bus and Surface Laboratory engineering data; separate from the Capsule Bus or lander early in the descent sequence; descend to the surface via parachute; survive omni-directional impact at 250 ft/sec; and transmit the engineering and surface science data direct to Earth. The general characteristics of the subsystem and basic science instrument complement are given in Figure 4.5-1. The design constraints, optimization studies, and supporting analyses which were conducted to establish this configuration are presented in Section II B 5.15.

The preferred concept employs a separable, hard landing, disk shaped capsule which

INDEPENDENT DATA PACKAGE GENERAL CHARACTERISTICS

BASIC CONFIGURATION

- Disk: 38 Inches Diameter x 14 Inches High
- Omnidirectional Impact Protection
- 250 Ft./Sec. Design Impact Velocity
- 3100 g Peak Impact Deceleration
- Parachute Descent Retardation
- 100 Pounds Gross System Weight
- Payload Size: 15.6 Inches Diameter x 5 Inches High
- Payload Weight Fraction 0.5 (Nominal)
- Balsa Wood Impact Limiter (6 lb/ft^3)
- Two Atmospheric Sensor Masts (Selective Deployment)
- Six Fixed Cavity-Backed Cross Slot Antennas
- 4π Steradian Data Transmission
- 24 Hour Surface Operating Lifetime
- Silver-Zinc, 25 Watt-Hour/Pound, Battery
- Direct MFSK Telecommunication Link
- 20 Watts Transmitter Output Power, 1.2 BPS
- 800 Bit Magnetic Core Memory

BASIC INSTRUMENTS

- Vibrating Diaphragm Pressure Transducer
- Gas Chromatograph for Atmospheric Composition
- Hygroscopic Sensor for Water Vapor Detection
- Hot-Wire Anemometer for Wind Velocity
- IDP/CB Diagnostic Sensors

Figure 4.5-1

is deployed near Aeroshell/lander separation. The essential elements of the landed payload and the IDP subsystem as they would appear installed on the Capsule Bus are shown in Figures 4.5-2 and -3. A simplified functional block diagram is shown in Figure 4.5-4.

4.5.3.2 Installation of IDP into Flight Capsule - Installation of IDP within the Flight Capsule must consider many factors, principally:

- Locating the IDP off the centerline requires additional weight for ballast and may have adverse affects on the reaction control subsystem when the IDP is deployed.
- Locating the IDP on the Surface Laboratory requires beefed-up support structure and may interfere with externally mounted experiments.
- The physical location should not interfere with or degrade the overall performance of the Flight Capsule.
- A simple, highly reliable deployment technique should be used.

The installations considered most desirable are shown in Figure 4.5-5. A summary of the major problems encountered while trying to install the IDP in our baseline design is given in Figure 4.5-6.

4.5.3.3 Deployment Techniques - The deployment of the IDP for all preferred separation altitudes requires pyrotechnic devices, parachute, sequencing and timing, and electrical power. The IDP deployment is a complicated procedure. Deployment must occur with minimum reaction torque on the Flight Capsule to prevent tumbling the Capsule Bus during the terminal descent phase. Three deployment sequences were considered for each IDP location:

Forward Location	Aft Location
o Deployment through nose	o Deployment prior to parachute deployment
o Deployment from lander	o Deployment from de-orbit motor structure
o Deployment from Aeroshell	o Deployment from lander

A typical sequence is given below for each location:

Forward Location - Aeroshell Deployment - As the IDP is released with the Aeroshell at 18,000 feet, the following events occur:

- IDP remains on the aeroshell (Time Delay)
- IDP released from Aeroshell Section (5000 feet) by exploding bolt holding clamp ring.
- IDP parachute released at separation (5000 feet)
- IDP descent on parachute
- IDP parachute separation (50 feet or below)
- IDP hard landing

INDEPENDENT DATA PACKAGE PAYLOAD

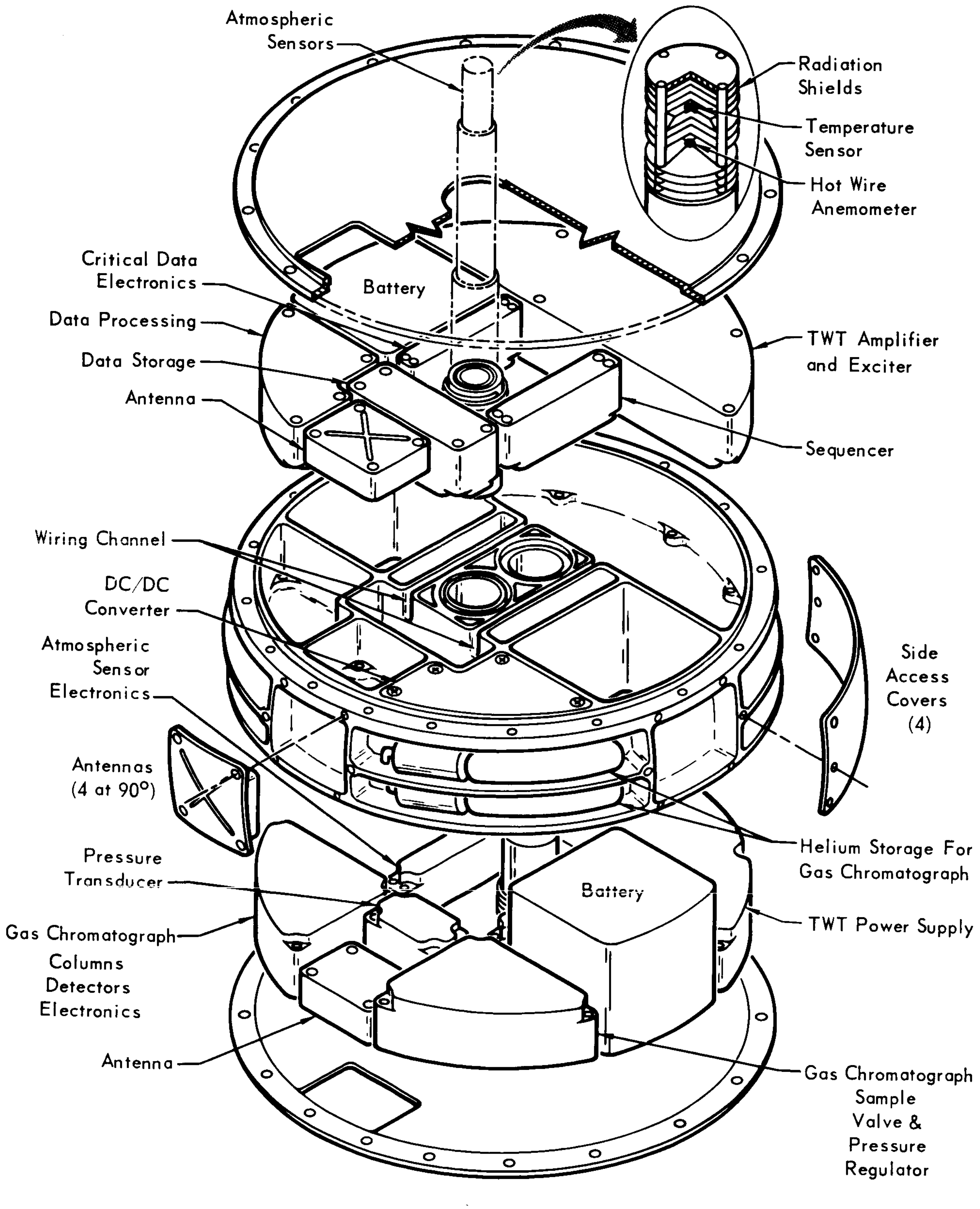


Figure 4.5-2

INDEPENDENT DATA PACKAGE SUBSYSTEM

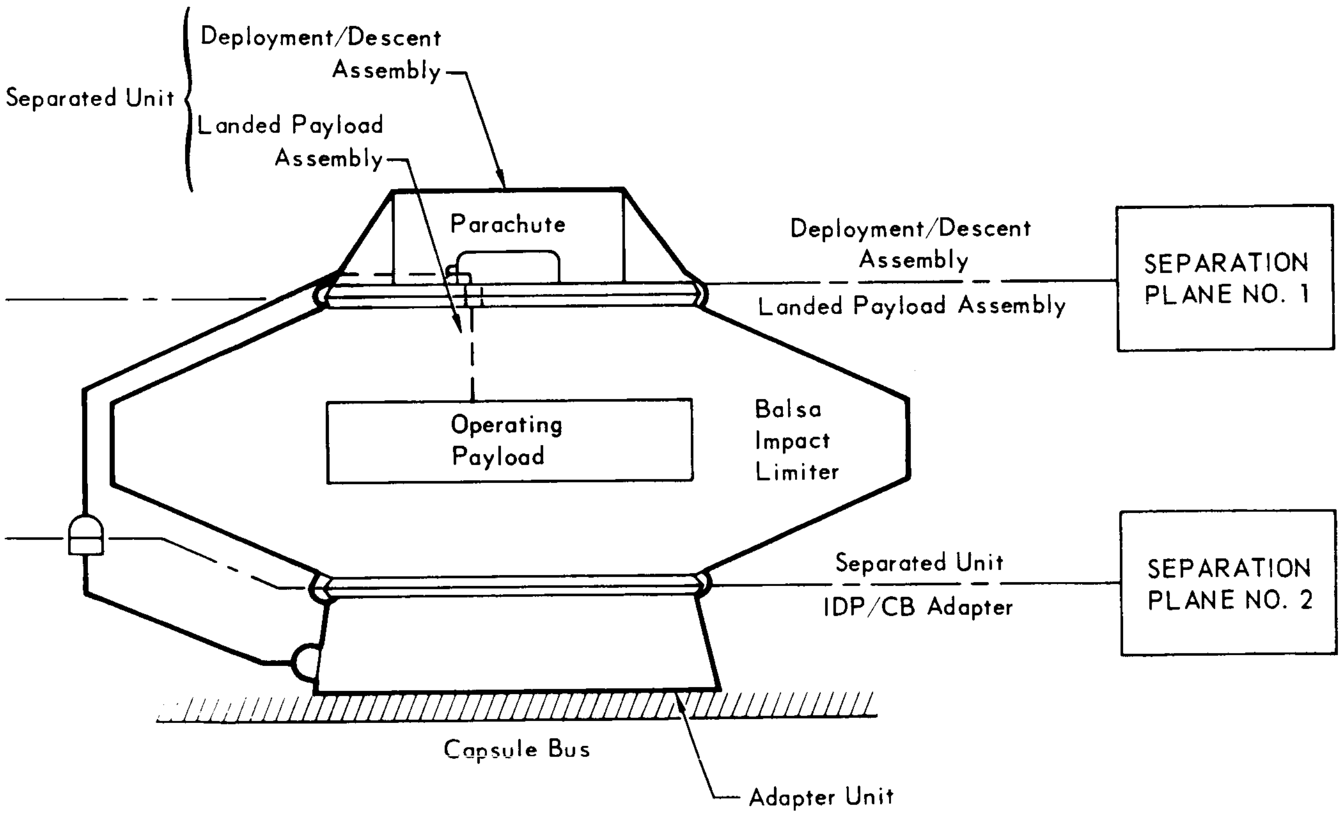
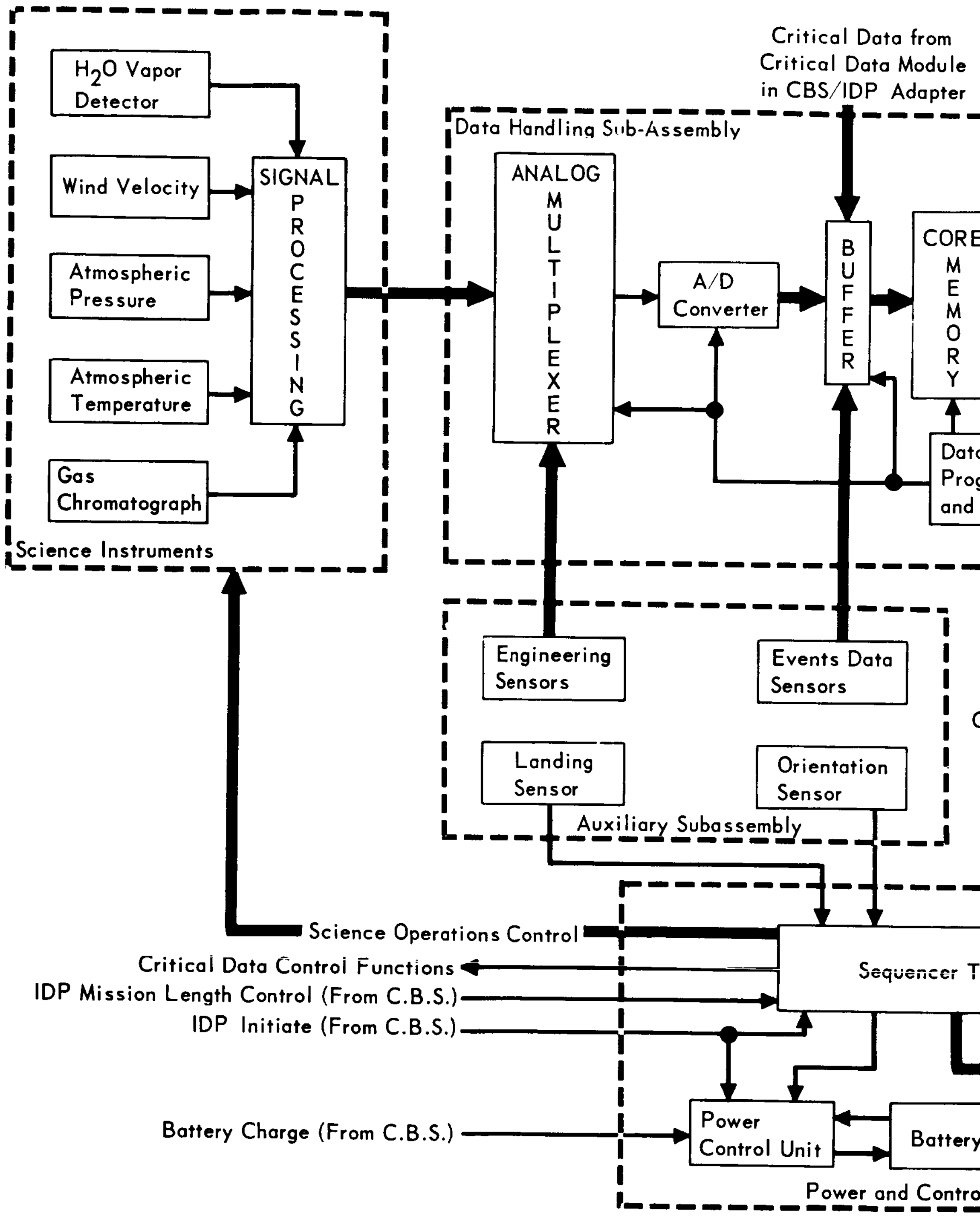


Figure 4.5-3

FUNCTIONAL BLOCK DIAGRAM – INDEPENDENT DATA PACKAGE



4.5-6-1

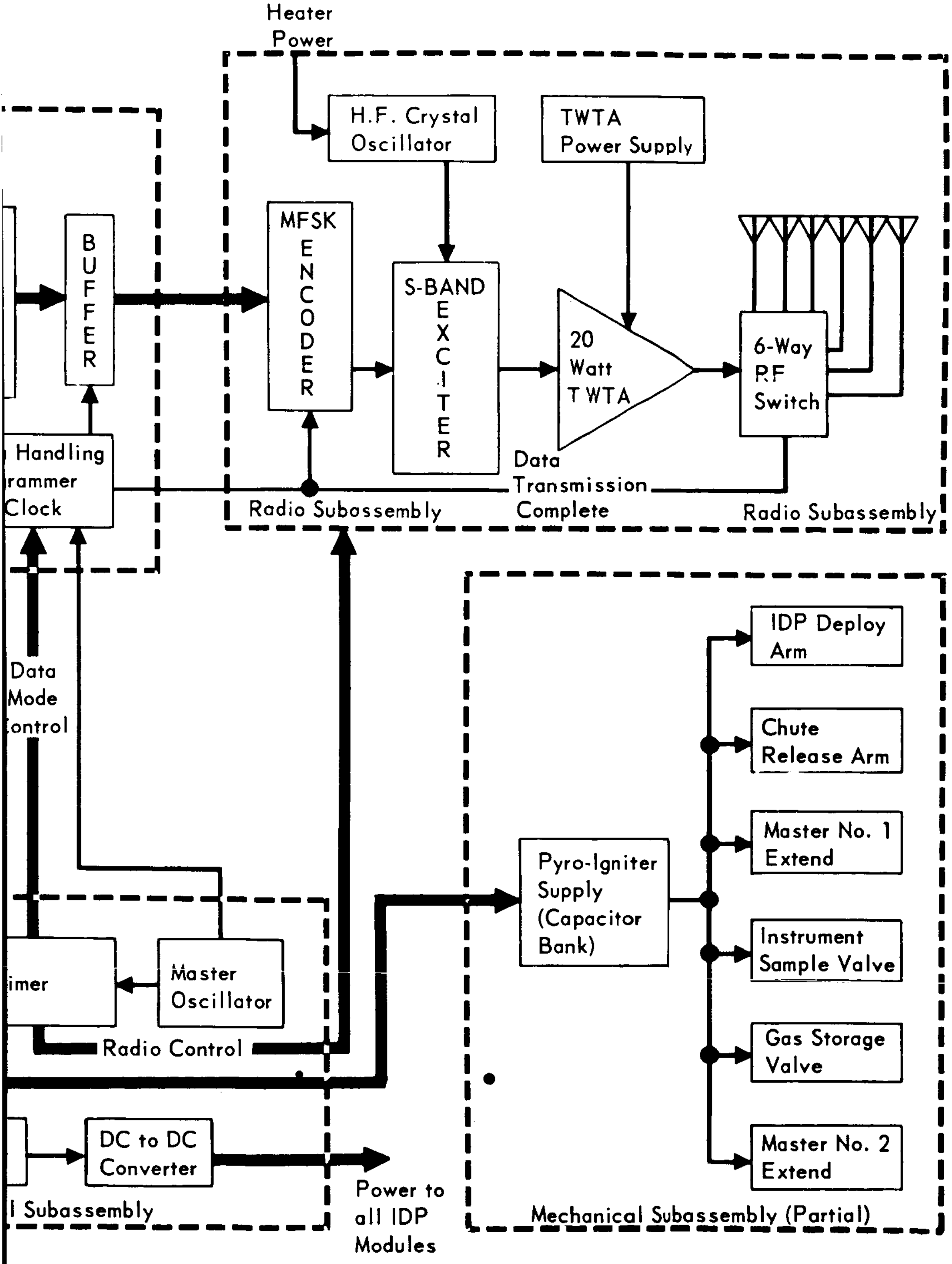


Figure 4.5-4

~~4.5-6~~

4.5-6-2

INDEPENDENT DATA PACKAGE INSTALLATION

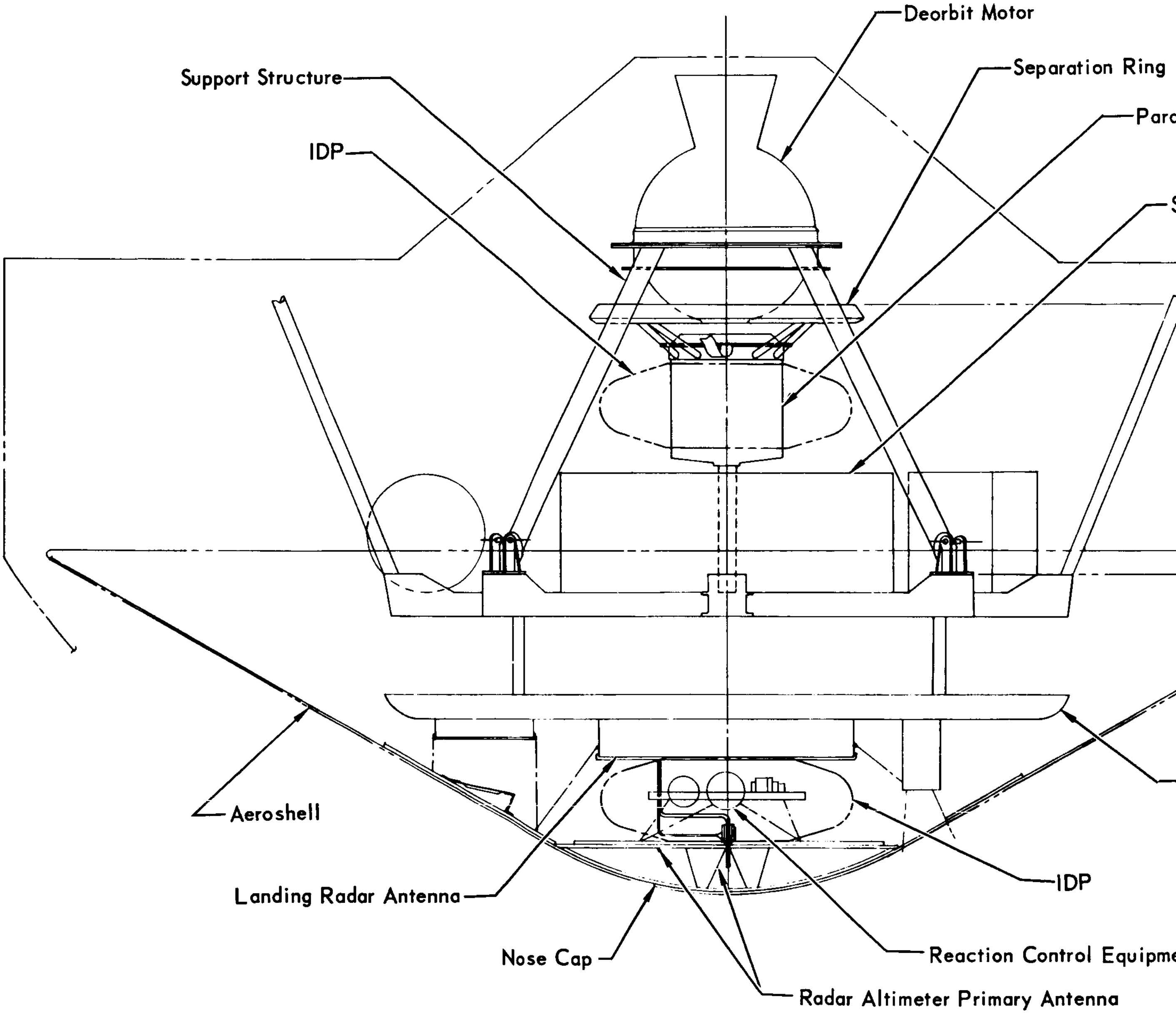
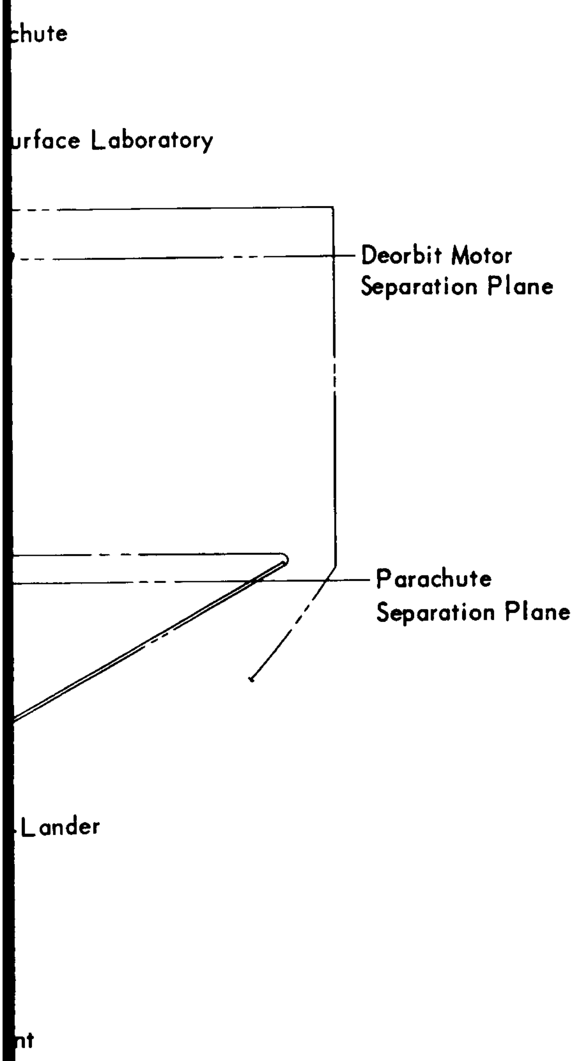


Figure 4.5-5



chute
urface Laboratory
Deorbit Motor
Separation Plane
Parachute
Separation Plane
Lander
nt

SUMMARY OF IDP INSTALLATION PROBLEMS

FORWARD AEROSHELL INSTALLATION	AFT DE-ORBIT STRUCTURE INSTALLATION
o The IDP configuration does not fit in the space forward of the Lander without modification of the radar antenna structure. o The IDP is inaccessible after installation. o The reaction control equipment must be relocated. o If deployment fails, the IDP may not survive impact. o The IDP position causes major deployment sequencing problems: 1) Deployment through the nose. a) The present radar design cannot be used. b) Design of the Aeroshell structure is complicated by the need for hinge and or cutting mechanism. 2) Deployment by removal with Lander. a) Additional weight of the IDP, IDP attach structure, and IDP parachute system could tend to retard separation of the Lander from the Aeroshell b) Failure of the IDP to deploy from the Lander would nullify the benefits of the Lander design. 3) Deployment from Aeroshell after Lander separation a) Failure of release mechanism to operate would cause the IDP to remain entrapped in the Aeroshell.	o The IDP position causes unsatisfactory relocation of the parachute off the center of gravity. o Mounting of external equipment on the Surface Laboratory is limited. o Additional structure is required for IDP mounting to the de-orbit motor support structure. o If the IDP fails its mounting, it would crush any externally mounted Surface Laboratory equipment. If deployment fails, the S-Band (high rate) antenna cannot be deployed and would severely limit T M data transmission. o The IDP must be shielded from high entry heat. o If deployment fails, the IDP may not survive impact. o The IDP position causes major deployment sequencing problems. 1) Deployment prior to parachute deployment. a) Additional booster system is required to eject the IDP. b) If deployment fails, the Surface Laboratory deployment experiments located beneath the IDP are unusable. 2) Deployment with main parachute separation from Lander. (IDP remains with de-orbit motor structure). a) Additional weight of the IDP, IDP attach Structure, and IDP parachute system could tend to retard separation of (1) Lander from Aeroshell and (2) parachute and de-orbit motor structure from the Lander. b) Low altitude parachute separation (less than 5000 feet) provides little time for deployment of the IDP parachute. 3) Deployment with main parachute separation from Lander (IDP remains with Lander). a) Additional weight of the IDP, IDP attachment structure, and IDP parachute subsystem could tend to retard separation of Lander from Aeroshell. b) If deployment fails, the extra weight may degrade Lander landing performance.

Figure 4.5-6

Aft Location - De-Orbit Motor Structure Deployment - IDP deployment from the de-orbit motor structure requires a two step structure separation rather than one step. This results from the need to keep the IDP mounting structure until IDP deployment.

- o De-orbit motor separation (800,000 feet)
- o Lander parachute deployment (23,000-20,000 feet)
- o Aeroshell separation (prior to 18,000 feet)
- o Lander on parachute (timed delay)
- o Parachute release (5,000 feet) and Terminal Propulsion System activates
- o IDP released from Capsule Lander
- o IDP separation released parachute
- o IDP parachute separation (50 feet or below)
- o IDP hard landing

4.5.3.4 Development Problems - There is a degree of development risk associated with the IDP design. The cost to design, test and manufacture five IDP's could be $23 to 30 million. Five vehicles (two IDP's for Flight, two for back-up, and one for testing) are required for the 1973 mission. The potential development problems are sterilizable high g batteries, impactable 20 watt transmitter, and impactable instruments.

Sterilizable High g Battery - A key problem area in the IDP design is the method of power generation. The best solution at this time is the silver-zinc battery. Several studies are in process to determine the best design for a silver-zinc battery to survive the two major environmental requirements of VOYAGER sterilization and a 3100 g impact. Although higher estimates have been given by some battery manufacturers, the best conservative estimate for battery specific energy is 25 Wh/lb.

Impactable Transmitter (20 Watt) - Hardware design problems include those of crystal oscillator instability and traveling wave tube amplifier design. The crystal oscillator design is especially difficult in the case of the IDP since a shock level of 3100 g is combined with the wide temperature variation during a Mars diurnal cycle. In order to withstand the shock, and to reduce the crystal oscillator drift as a function of changing temperature, it is necessary to house the crystal and the oscillator and buffer stages within a shock resistant isothermal environment. It is apparent that a traveling wave tube amplifier (TWTA) is necessary to generate efficiently 20 Watts of RF power at S-Band. This approach presents a problem in the case of the IDP shock environment (3100 g). Watkins-Johnson Inc.

has done the only known work to date in implementing a shock resistant TWTA. Their tube, Model No. WJ-398 (22 Watts at S-Band), has been successfully tested at a 10,000 g peak, 1 millisecond duration shock level.

Impactable Instruments - The instruments considered for the IDP and their development status are given in Figure 4.5-7. The necessary instruments should be developed in approximately two years; however, the capability required is beyond the present state of the art.

4.5.3.5 IDP Operation After Landing - The uncertainties of the horizontal wind velocity and Martian surface will have a significant affect on the successful operation of the IDP. Throughout this study the assumption was made that the IDP would land and operate satisfactorily. However, the landing loads on the IDP and its final position could prevent instrument mast deployment and/or cause damage to some of the antennas. These and similar types of landing problems could limit or prevent useful data being transmitted by the IDP. In evaluating the IDP these uncertainties must also be considered along with all the other facts presented in this study.

4.5.4 Evaluation - After the IDP design was established, the value of the IDP as a part of the Flight Capsule was analyzed. The weight required for the IDP, 100 lb, can be allocated in three alternate ways: (1) the IDP can be incorporated into the design; (2) the 100 lb can be used to improve the Surface Laboratory through redundancy additions; (3) the 100 lb can be used to improve the effectiveness of the Flight Capsule by the technique described in Section 4.10. The uncertainties of the IDP and Capsule Bus System interference (i.e. recontact, parachute entanglement, parachutes landing on IDP or Surface Laboratory) and the Martian surface conditions affects on the IDP landing were not included in this analysis.

4.5.4.1 Reliability Analysis - The reliability analysis was conducted based on obtaining minimum, low rate surface environmental data. For simplicity in conducting the analysis, the instrument reliability (for the surface pressure, temperature, wind speed, water vapor, and composition measurements) in the Surface Laboratory and IDP were assumed to be the same so they were not included in the IDP reliability estimates. Inclusion of the experiments would not significantly affect the results.

Eleven different configuration were initially analyzed for reliability, including eight different IDP release and deployment sequence times during the descent and landing mission phases. The IDP release and deployment sequence

IDP INSTRUMENT DEVELOPMENT STATUS

INSTRUMENT	LOCATION	REMAINING DEVELOPMENT
Vibrating Diaphragm Pressure Transducer	Interior (Electronics) – Access port for static pressure	Completion of integrated circuit design Shock hardening Sterilizability Production methods
Platinum Resistance Thermometer	Interior (Electronics) – Sensor deployed on extendable mast	Completion of integrated circuit design Completion of deployment method and radiation shield design Shock hardening Sterilizability
Hot Thermocouple Anemometer	Interior (Electronics) – Sensor deployed on expendable mast	Completion of integrated circuit design Completion of deployment design Completion of study of calibration methods Study of atmospheric composition effect Shock hardening Sterilizability
Gas Chromatograph	Interior	Fabrication and test of the two column gas chromotograph Completion of study of sampling and calibration methods Shock hardening Perfection of double dynamic range Sterilizability
Hygroscopic Water Vapor Sensor	Interior	Completion of integrated circuit design Temperature effects Sensitivity Calibration methods Shock hardening Sterilizability

Figure 4.5-7

analysis is presented in Volume II, Part B, Section 5.15.3. Four configurations were selected for detailed analysis as follows:

Configuration 1 - Baseline Flight Capsule without IDP.

Configuration 2 - Baseline Flight Capsule without IDP but with 100 lb. of reliability improvements incorporated into the Surface Laboratory only.

Configuration 3 - Baseline Flight Capsule with IDP; IDP is released from the de-orbit motor structure or the Aeroshell, but prior to terminal propulsion motor ignition.

Configuration 4 - Baseline Flight Capsule without IDP but with 100 lb. of reliability improvements incorporated anywhere within the Flight Capsule.

The reliability estimates include values for an IDP, an individual Flight Capsule, an individual Flight Capsule with an IDP, dual Flight Capsules, and dual Flight Capsules each with an IDP. The estimate are based on partial mission success (minimum surface experiment data) and include all Flight Capsule mission phases beginning at launch. The estimates do not include unreliability associated with the Flight Spacecraft and Launch Vehicle.

Figure 4.5-8 summarizes the reliability estimates. Configurations 2, 3 and 4 were compared to the Baseline (Configuration 1) by use of the Reliability Improvement Factor (RIF). The RIF is the ratio of the natural logarithms of the estimated reliabilities. The RIF is a measure of the reduction in unreliability and, therefore, is an indicator of the reliability improvement. Compared to Configuration 1, the reliability improvement factor is 1.5, 2.5, and 5.82 for Configurations 2, 3, and 4. Configuration 4, therefore, is the best based on the numerical reliability estimates. In addition to the numerical estimates, failure modes and effects, including critical single point failure possibilities, were considered in this analysis. The reliability models of the four Configurations studied are shown in Figure 4.5-9.

Configuration 2 (Improved Surface Laboratory) - This configuration enhances the reliability of Surface Laboratory electrical power, sequencer and timer, and telemetry subsystems. No major function single point failures are totally bypassed. Capsule Bus terminal propulsion and landing radar reliability are unchanged from the baseline design.

INDEPENDENT DATA PACKAGE RELIABILITY ESTIMATE

CONFIGURATION	EQUIPMENT RELIABILITY					* RELIABILITY IMPROVEMENT FACTOR
	FC	2 FC	IDP	FC + IDP	2 FC + 2 IDP	
1 (Baseline w/o IDP)	.858	.977	–	–	–	1.00
2 (Baseline + 100 lb in S.L.)	.882	.984	–	–	–	1.44
3 (Baseline + IDP)	.858	–	.841	.908	.991	2.50
4 (Baseline + 100 lb in FC)	.942	.996	–	–	–	5.82

$$* \text{R.I.F.} = \frac{\text{Baseline (LnR)}}{\text{Configuration (LnR)}}$$

Figure 4.5-8

RELIABILITY MODE

MINIMUM SURFACE D
IDP RELEASE PRIOR TO TERMINAL

CONFIGURATION 1 (BASELINE)

FC Electrical Power — FC Staging — CB Sequencing and Timing — CB Guidance and Control — CB Attitude Control and De-Orbit Propulsion —

CONFIGURATION 2 (100 LB. OF IMPROVEMENTS ADDED TO THE SURFACE LABORATORY)

√ FC Electrical Power — FC Staging — CB Sequencing and Timing — CB Guidance and Control — CB Attitude Control and De-Orbit Propulsion —

CONFIGURATION 3 (BASELINE WITH IDP)

FC Electrical Power — FC Staging — CB Sequencing and Timing — CB Guidance and Control Less the Landing Radar — CB Attitude Control and De-Orbit Propulsion —

CONFIGURATION 4 (100 LB. OF IMPROVEMENTS ADDED THROUGHOUT THE FLIGHT CAPSULE)

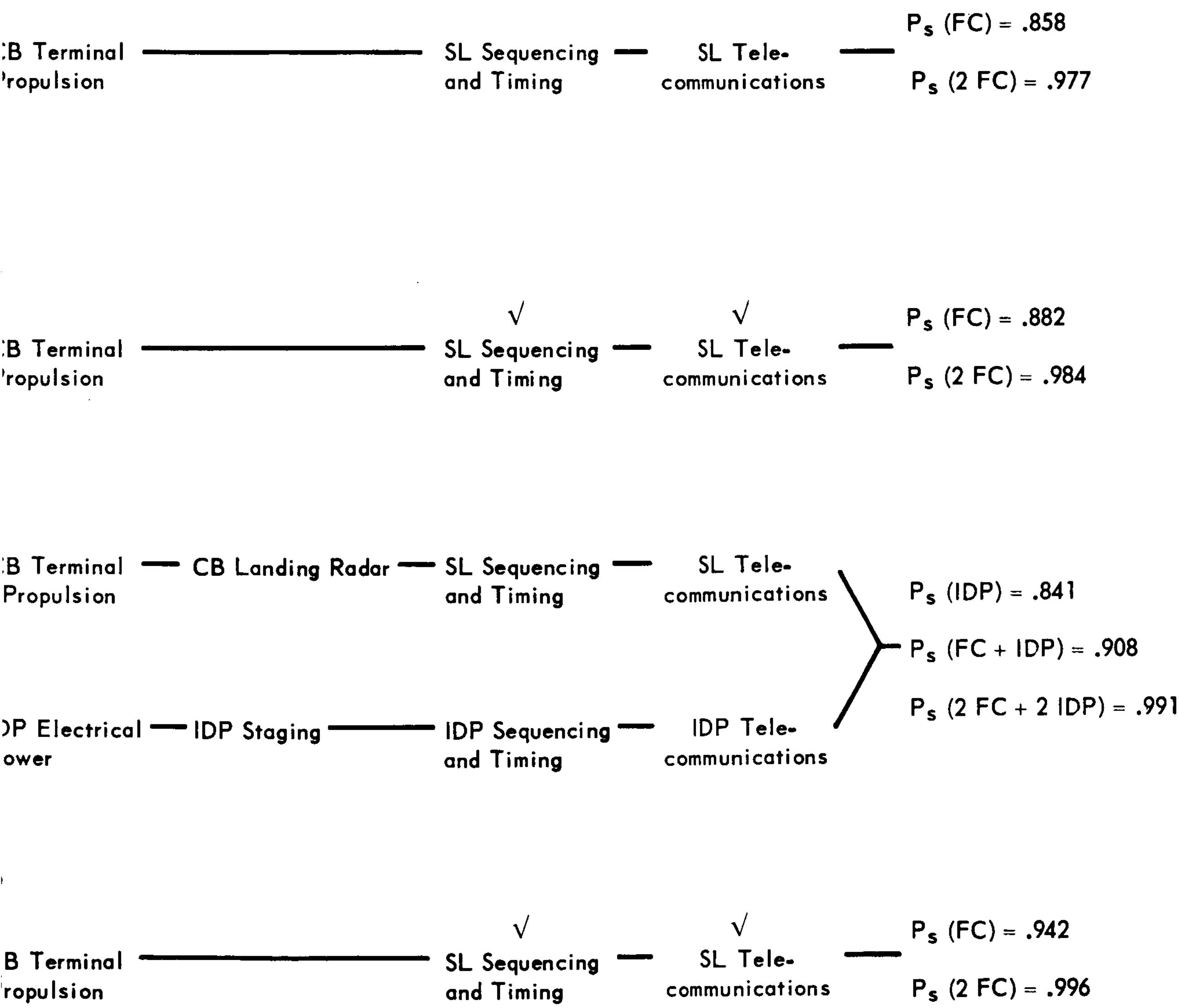
√ FC Electrical Power — √ FC Staging — √ CB Sequencing and Timing — √ CB Guidance and Control — √ CB Attitude Control and De-Orbit Propulsion —

√ – Denotes the functions which have improvements.

Figure 4.5–9

IDP STUDY

RETRIEVAL
PULSION MOTOR IGNITION



4.5-14-2

Configuration 3 (Baseline with IDP) - This configuration is the only design which effectively by-passes four major function single point failure possibilities, namely, Capsule Bus terminal propulsion and landing radar, and Surface Laboratory sequencing and timing and telecommunications. The other design configurations totally by-pass none. Capsule Bus terminal propulsion and landing radar are considered to be the most critical single point failure possibilities because proper performance of all four engines and the radar is required for landing, survival and retrieval of minimum surface environmental data.

Configuration 4 (Improved Flight Capsule) - The reliability of all major functions is improved with the exception of Capsule Bus terminal propulsion. Reliability improvements for the terminal propulsion function requires too large a share of the 100 lb and still neither improves the numerical reliability estimate significantly nor alters the critical single point failure possibilities (4 of 4 engines) for this subsystem.

4.5.4.2 IDP Value Assessment - The IDP deployment point (Concept A) does minimize the chance of IDP interference with the Surface Laboratory. If both the Surface Laboratory and the IDP land and operate successfully, the separation distance will enhance the measurements through correlation of similar measurements, and by measuring data in an uncontaminated area. The relative value of the IDP and the Surface Laboratory data is very difficult to define. The nominal information capacity of the Surface Laboratory is 450,000 non-imaging bits compared to 800 bits for the IDP. The IDP data is clearly a small addition to the total number of bits. However, the IDP data, being obtained from an uncontaminated area, may make this small amount of data of significant importance. If the Surface Laboratory lands and operates nominally, the primary value of the IDP is to supply composition data from an uncontaminated area. For this case we estimate the overall value of the additional IDP data to be 17% of the total Surface Laboratory value. If the Surface Laboratory does not land successfully, the IDP data becomes more important, it being the sole source of surface data. Thus, we consider the overall value of the IDP data to be 33% of the total Surface Laboratory value in this case.

4.5.4.3 Effectiveness Analysis - The IDP contribution to the total system effectiveness was evaluated relative to the effectiveness of adding 100 lbs of improvements to the Flight Capsule. The comparison was made using the effectiveness technique described in Section 4.6. The study considered both the total system effectiveness (E) and the effectiveness of achieving landed experiments (E_3).

However, when an IDP is added the effectiveness model is modified as shown in Figure 4.5-10. Then the term $E_3 = V_3R_3$ must be redefined:

$$E_3 = V_3R_3 + (1-R_3)\ (K_1V_3)\ R_4 + (K_2V_3)\ R_3R_4$$

where

V_3 = Value of landed experiments

K_1V_3 = Value given to IDP if Surface Laboratory fails (K_1 = .33)

K_2V_3 = Value given to IDP if Surface Laboratory is successful (K_2 = .17)

R_3 = Reliability of Surface Laboratory equipment

R_4 = Reliability of IDP equipment

$(1-R_3)$ = Probability of Surface Laboratory equipment failure

The system effectiveness, landed experiments effectiveness, and improvement in system effectiveness were determined for the two configurations. The results are given in Figure 4.5-11. The addition of an IDP improves both the system effectiveness and effectiveness of landed experiments by .0465. The 100 pounds of improvements in the Flight Capsule increases the System Effectiveness by .0846 and the effectiveness of landed experiments by .0232. Although the IDP gives the best improvement in effectiveness of the landed experiments, it is not effective on a total system basis. The system effectiveness improvement for 100 pounds of improvements is approximately twice that for the IDP. If the IDP equipment reliability and estimated value for the 100 pound IDP could be retained while the weight was reduced the system effectiveness break-even point for an IDP is twenty-five pounds.

It is recognized that the value assigned to K_1 is argumentative. Therefore, an examination of the system effectiveness (E) sensitivity to K_1 was made for the configuration with an IDP. The system effectiveness break-even point is at K_1 = 0.94. However, the value of obtaining IDP data (minimum surface data) is clearly not 94% of the value of the Surface Laboratory data.

4.5.5 Conclusions - The results of this analysis are presented with an evaluation of the IDP in Figure 4.5-12.

Installation of an IDP into the Flight Capsule System presents many problems. The size of the IDP severly restricts location within the Flight Capsule. The weight and reaction torque resulting from deployment requires location near (preferably on) the centerline of the Flight Capsule to minimize ballast weight. Installation of the IDP in either location considered, results in inefficient installation of other equipment to provide space for the IDP. A weight penalty, over the IDP weight, is required to install the IDP in the Flight Capsule because of relocating some of the Capsule Bus equipment.

RELIABILITY DIAGRAM
THREE PHASE MISSION RELIABILITY/EFFECTIVENESS ANALYSIS

(1) Not Included in the Calculations

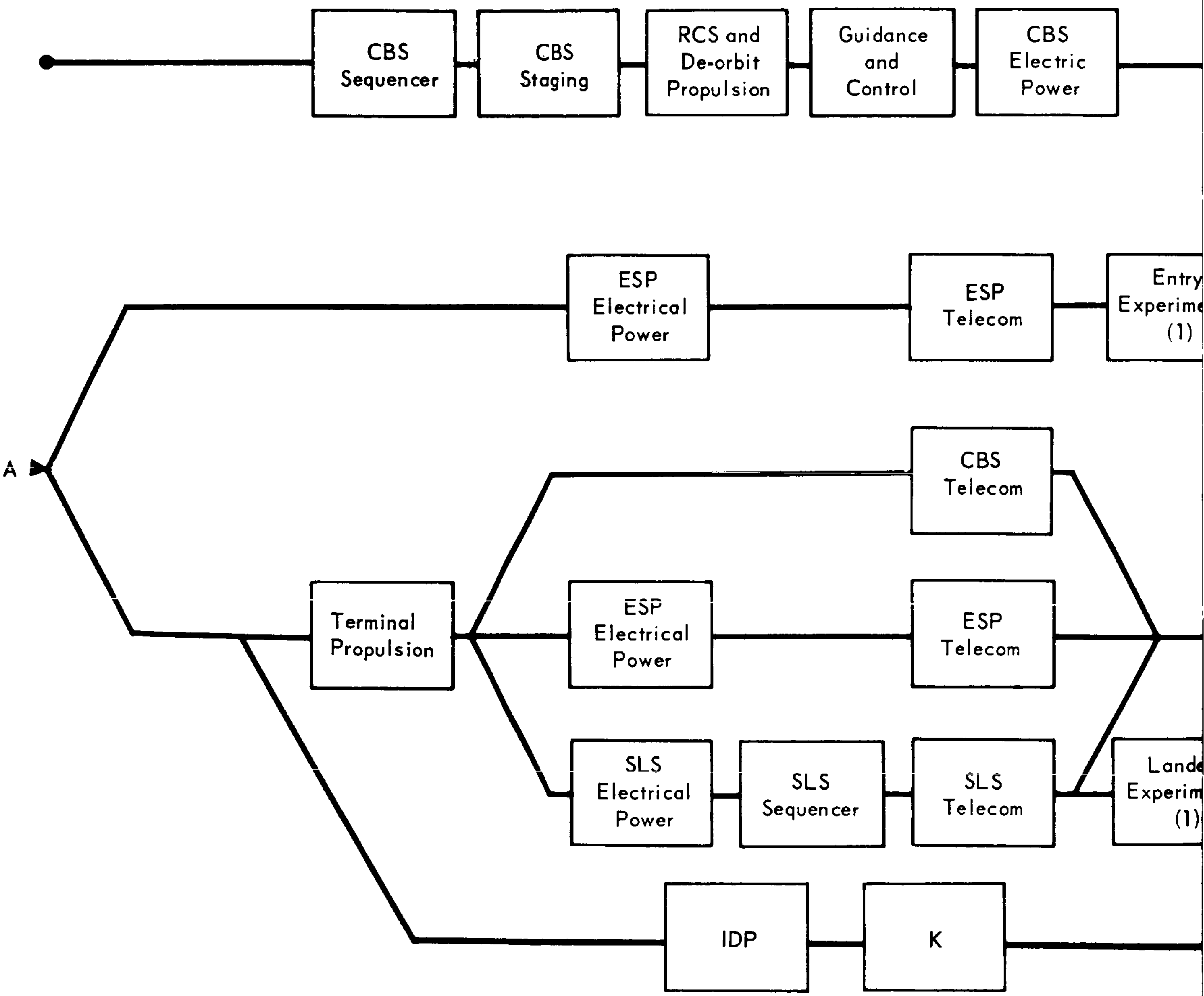
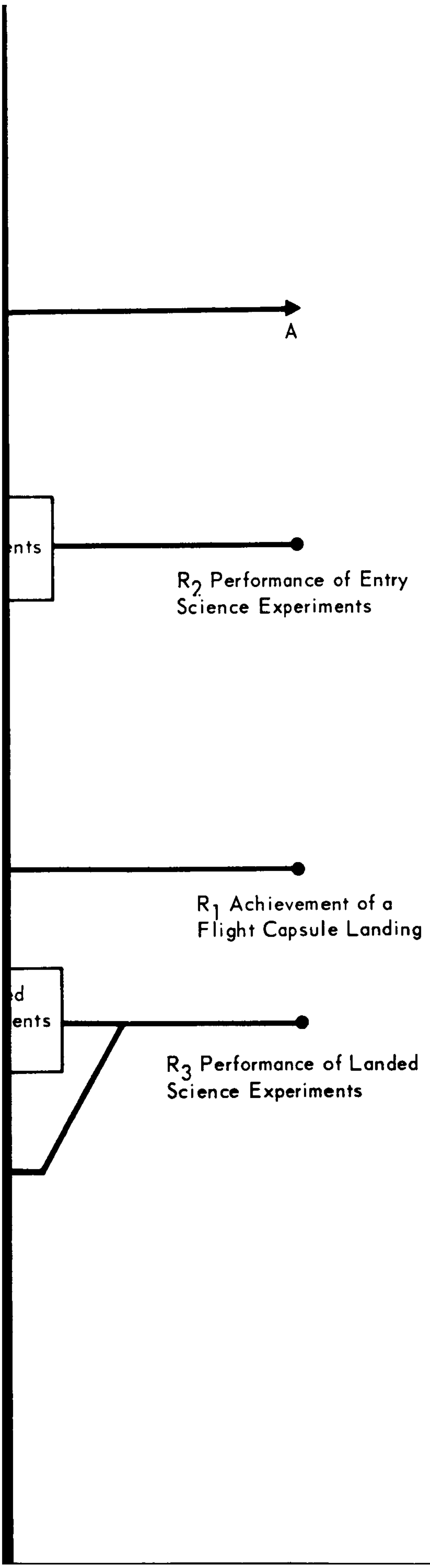


Figure 4.5–10


A
nts
R2 Performance of Entry
Science Experiments
R1 Achievement of a
Flight Capsule Landing
d
ents
R3 Performance of Landed
Science Experiments

EFFECTIVENESS

WEIGHT FACTOR	ΔE	E	E_3
Zero Redundant Baseline	–	0.7814	0.1715
With 100 lb. IDP	0.0465	0.8279	0.2180
With 100 lb. Redundancies	0.0846	0.8660	0.1947

$$E = E_1 + E_2 + E_3$$

E_1 = Effectiveness of achievement of Landing

E_2 = Effectiveness of achievement of Entry Experiment

E_3 = Effectiveness of achievement of Landed Experiment.

Figure 4.5–11

INDEPENDENT DATA PACKAGE EVALUATION

Favorable Conclusions
• The IDP increases probability of obtaining the surface environmental measurements, gives correlation to the Surface Laboratory (if both operate) measurements, and makes measurements in an uncontaminated area. • The IDP by-passes four single point failure possibilities, the Terminal Propulsion Subsystem being the only one not improved by adding 100 pounds of improvements to the Flight Capsule. • The optimum IDP separation point is concept A (approximately 5000 feet). This occurs prior to activation of the Terminal Propulsion Subsystem.

Unfavorable Conclusions
• Deployment of the IDP is complicated. • Installation of the IDP within the Flight Capsule is difficult and would result in an additional weight penalty over the IDP weight. • Unsuccessful deployment of the IDP, a single point failure, could interfere with the Surface Laboratory experiments and possibly result in total mission failure. • If the IDP fails to deploy, the S-Band (high rate) antenna cannot be deployed and the T/M data transmission is severely limited. • The IDP contributes only temperature, pressure, composition, water vapor, wind speed, and some diagnostic measurements. No subsurface or life measurements are made. • There is development risk associated with the sterilizable high g silver-zinc batteries (20-watt/wh/lb), impactable 20-watt transmitter, and impactable instruments. • The uncertainties of successful landing and instrument mast deployment are factors which contribute to a reduction in the total value of the IDP. • The improvement (100-pounds) to the Flight Capsule gives a greater improvement in probability of success of obtaining the surface environmental measurements than the IDP. • The 100-pounds Flight Capsule improvement also increases the probability of total mission success, while the IDP makes no contribution to the total mission success. • The 100-pound improvement to the Flight Capsule increases the landed experiments effectiveness and give approximately twice the improvement in the System Effectiveness as the IDP.

Figure 4.5-12

An IDP capability (Configuration 3) or adding the weight as improvements within the Flight Capsule (Configuration 4) are justified based on the reliability analysis. However, the ability of the IDP to by-pass some of the critical single point failure possibilities must also be considered.

The IDP should be released from the Flight Capsule prior to terminal propulsion ignition for maximum mission reliability and maximum independence from critical Flight Capsule single point failure possibilities. IDP release from the Capsule Bus after terminal propulsion ignition results in a very questionable reliability improvement principally because of the criticality of a terminal propulsion engine or landing radar failure.

The IDP is not the most effective method of utilizing excess weight. Adding redundancies into the Flight Capsule improves the probability of total mission success and the overall system effectiveness.

As a result of the installation problems, weight penalties, IDP deployment and interference uncertainties, IDP operational uncertainties after hard landing, and reliability and effectiveness evaluations, it is recommended that the IDP not be incorporated into the baseline design.

However, it is desirable to by-pass the most critical failure possibilities and thereby improve the probability of obtaining some data. The two areas of greatest concern are the surface conditions and the terminal propulsion subsystem. It may be feasible to obtain a better improvement in the probability of collecting some surface data by hardening the Entry Science Package or a portion of the Surface Laboratory. However, the weight penalty should be carefully evaluated to insure effective use of the excess weight.

SECTION 4.5

REFERENCES

4.5-1 1973 VOYAGER Capsule Systems Constraints and Requirements Document Revision 2, Jet Propulsion Laboratory, PD606-4, 12 June 1967

4.5-2 General Specification for Performance and Design Requirements for the 1973 VOYAGER Mission, Jet Propulsion Laboratory, SE002BB001-1B21, 1 January 1967

4.6 SYSTEM EFFECTIVENESS ANALYSIS - Proper allocation of weight available for reliability improvement has been an important factor in our Flight Capsule studies. Our preferred design includes redundant components weighing 73 pounds. Six items (weighing 43 pounds) were included because our engineering judgment indicated they were necessary to satisfy design constraints. An additional 21 items were selected from a list of 45 recommended as a result of our system effectiveness analysis. This technique identifies redundancies which yield the maximum effectiveness gain per unit weight. Weight limitation, development risk, and potential subsystem integration problems are among the reasons for not including the other 24 at this time. Additional redundancy could be achieved by utilizing a portion of the Flight Capsule's 220 pound weight margin. Use of redundancy in our preferred design has increased, to 71%, the probability that all Flight Capsule equipment (excluding experiments) will function properly. (See Figure 4.6-1) The estimated reliability of the capsule (excluding experiments) for achieving landing is increased to 87% for performing Surface Laboratory experiments is increased to 78%.

4.6.1 Technique - A system effectiveness analysis is used to evaluate redundancies in terms of reliability improvement, change in weight and mission objectives. We have used the effectiveness equation $E = V_1 R_1 + V_2 R_2 + V_3 R_3$ in our calculations. Each term of this equation represents a single mission objective. The first term, $V_1 R_1$, represents "Achievement of a Flight Capsule Landing." V_1 is the value assigned to the event, and R_1 the estimated reliability of the Capsule Bus for performing that event. The second term, $V_2 R_2$, represents "Performance of Entry Experiments." V_2 is the value assigned to the performance of entry science experiments and R_2 is the estimated reliability of the Flight Capsule (excluding experiments) to complete that phase of the mission and transmit the data. The third terms, $V_3 R_3$, represent "Performance of Landed Experiments." V_3 is the value assigned to the performance of landed experiments; R_3 is the estimated reliability of the Flight Capsule (excluding experiments) to perform landed science experiments. Derivation of this method is described in Reference 4.6-1.

4.6.1.1 Value - Mission objectives, in order of decreasing importance, are listed in Reference 4.6-2. This ordering was used in the selection of values such that $V_1 > V_2 > V_3$. Various value distributions (with $V_1 + V_2 + V_3 = 1$) were used to determine the influence of value assignments in the effectiveness analysis. Significant effect of value assignment occurred only when V_1, the value assigned to landing, exceeded 0.50. For our current analysis we have used:

FLIGHT CAPSULE RELIABILITY

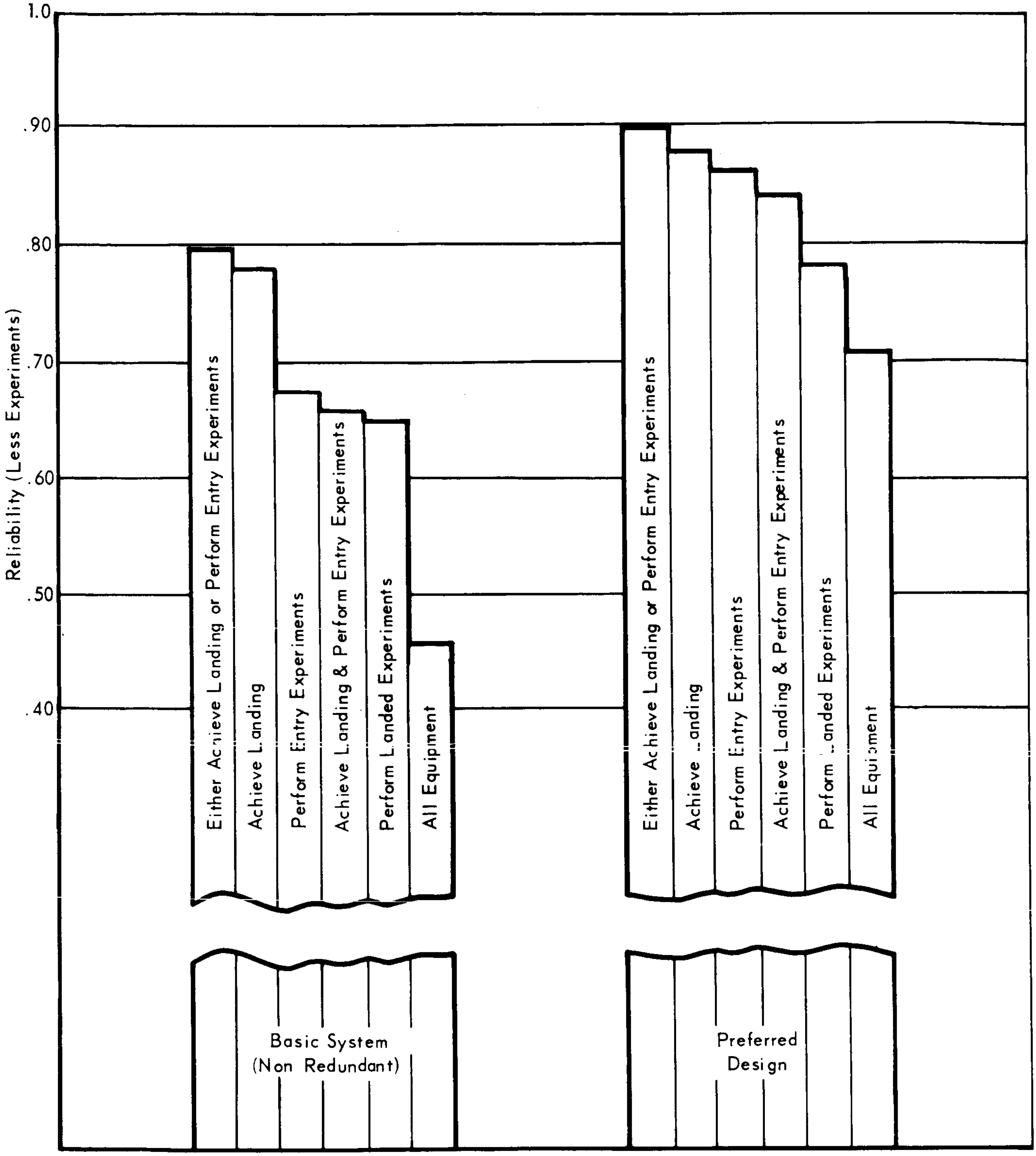


Figure 4.6-1

V_1 (successful landing) = 0.40

V_2 (entry experiment) = 0.35

V_3 (landed experiments) = 0.25

This approach to system effectiveness emphasizes the reliability of the components which are essential to completion of two or more of the objectives. Further, it de-emphasizes component redundancy where a component is used for a single objective or multiple paths are available. The reliability of the Flight Capsule components required to accomplish each objective enters the computation in conjunction with the value of that objective. The subsystem components (Figure 4.6-2) - CB sequencing, staging, reaction and de-orbit propulsion, guidance and control, thermal control and electrical power - are required to operate to perform any of three events. A value of 1.00 is associated with these components since their reliability enters all three terms of the equation. Terminal propulsion and landing components operations are required to perform the first and third events; value $V_1 + V_3$ was associated with these components. Successful operation of any one of the three parallel paths (See the lower part of Figure 4.6-2) of telecommunications will provide verification of a successful landing. Therefore, the resultant reliability of this parallel combination was assigned the value, V_1.

4.6.1.2 Weight Factors - The weight increment directly associated with each redundancy is that of the component itself. However, incorporation of each redundancy imposes an additional weight increment to support that component. For example, the addition of a component to the Surface Laboratory requires a corresponding increase in the weight of those Flight Capsule subsystems required to land that component on the surface. The terminal propulsion subsystem would require more fuel, the parachutes would be a little larger, and so on. This effect was included in the analysis by using weight factors. The weight factor is the partial derivative of the Flight Capsule weight with respect to component weight. In this analysis we have used a difference ratio to approximate the derivative as indicated in Figure 4.6-3. For example, a recommended redundant relay, actual weight 0.3 pounds, added to the Surface Laboratory electrical power subsystem would cause a resultant weight addition of 0.54 pounds to the Flight Capsule weight:

RELIABILITY DIAGRAM

EFFECTIVENESS ANALYSIS $E = R_1 V_1 + R_2 V_2 + R_3 V_3$

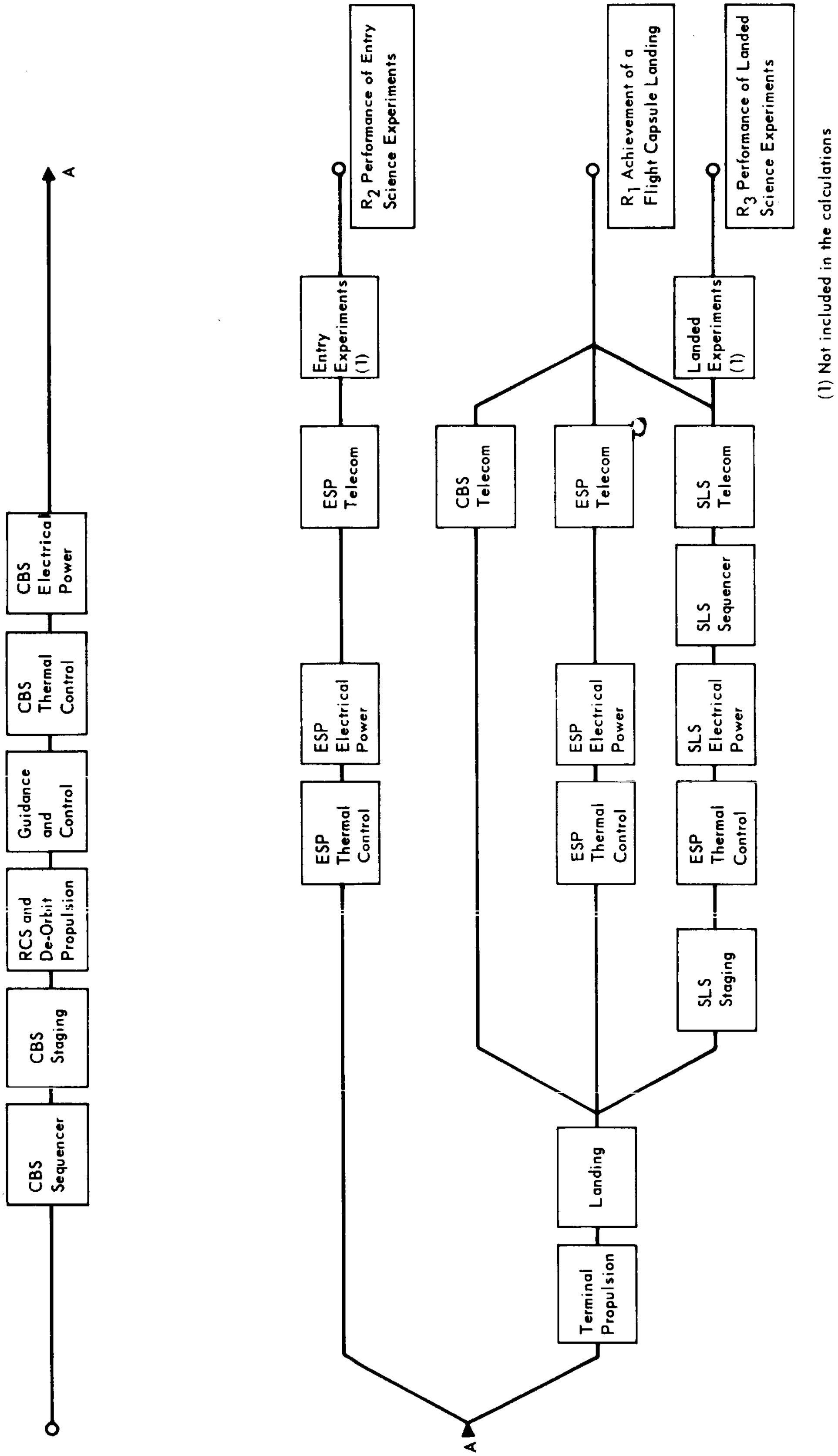


Figure 4.6-2

WEIGHT FACTORS

	FLIGHT CAPSULE	MISSION PHASE: PRE DE-ORBIT	POST DE-ORBIT	ENTRY	TERMINAL DESCENT	TOUCH DOWN	SURFACE LAB
5500 lb Capsule (1)(2)	5500.0	4678.2	4169.9	4084.7	3164.6	3089.6	1159.6
4500 lb Capsule (1)(3)	4500.0	3684.5	3283.6	3214.6	2396.1	2321.1	607.7
ΔW (4)	1000.0	993.7	886.3	870.1	768.5	768.5	551.9
Weight Factor (5)	1.00	1.01	1.13	1.15	1.30	1.30	1.81

(1) VOYAGER Weight Program Computer Run 22 June 1967 (Not Published)
(2) 5500 lb Weight Summary Values
(3) 4500 lb Weight Summary Values
(4) ΔW = Difference in Mission Phase Weight (2)–(3).
(5) Weight Factors $= \frac{1000}{\Delta W}$

Figure 4.6–3

$$\Delta W_{FC} = W_A \times F_{SL}$$

$$\Delta W_{FC} = .3 \times 1.81$$

$$\Delta W_{FC} = .54$$

where:

ΔW_{FC} = The added Flight Capsule weight to accommodate the redundancy.

W_A = The actual weight of the added component.

F_{SL} = The weight factor associated with components added to the Surface Laboratory

4.6.1.3 Selection - The selection of recommended items of redundancy is made by evaluating the ratio of the weight addition to the change in the effectiveness. The magnitude of this ratio, $\Delta W/_{\Delta E}$, established the order of selection. The component with the lowest $\Delta W/_{\Delta E}$ is selected first, followed by components of increasing $\Delta W/_{\Delta E}$. This ordering is similar to that of the reliability analysis. (See Section E2)

We applied the effectiveness analysis in recommending items to be made redundant for reliability improvement. A portion of the effectiveness analysis results is shown in Figure 4.6-4 with a similar portion of the reliability analysis for comparison.

A guidance and a propulsion redundancy, G10C and P1C, were selected for a specific example of the differing results which sould be obtained from the two approaches:

G10C - Active redundant receivers and trackers in radar altimeter

P1C - Redundant cartridge in each of three normally closed (RCS) pyro valves

The reliability analysis, as shown in Figure 4.6-4, indicates the desirability of incorporating the guidance redundancy, G10C if up to 100 pounds of redundancy were allowable. Effectiveness analysis gives higher priority to that redundancy and indicates inclusion of this redundancy if only 70 pounds of redundancy were allowable. A similar example is shown with P1C, the propulsion component redundancy. The reliability analyses method shows that 169 pounds must be available to warrant adding the redundant cartridges; the comparable threshold is 89 pounds using the effectiveness analysis.

4.6.2 Final Selection - Satisfying the design constraints was considered most important in the final selection. Six elements were made redundant because our engineering judgment indicated they were necessary to satisfy the Flight Capsule

RECOMMENDED COMPONENT REDUNDANCY

RELIABILITY ANALYSIS

REDUNDANCY CODE (1)	ΔW/ ΔLnR	RELIABILITY (2)	REDUNDANCY WEIGHT (3)	REDUNDANCIES
BASIC SYSTEM	–	.4579	–	
T11C	9	.4680	.19	STANDBY REDUNDANT CBS CRUISE ENCODER
T15E	12	.4757	.39	STANDBY REDUNDANT ESP CRUISE ENCODER
T10C	12	.4991	.97	SERIES ACTIVE REDUNDANT CBS CRUISE COMMUTATOR DATA SWITCHES & DRIVERS
T1E	15	.5101	1.30	INTERLEAVE LOW RATE ESP DATA ON CBS RADIO LINK
T23S	16	.5185	1.57	STANDBY REDUNDANT SLS CRUISE ENCODER
T14E	18	.5357	2.15	SERIES ACTIVE REDUNDANT ESP CRUISE COMMUTATOR DATA SWITCHES & DRIVERS
T2C	20	.5447	2.48	INTERLEAVE LOW RATE CBS DATA ON ESP RADIO LINK
T22S	22	.5655	3.29	SERIES ACTIVE REDUNDANT SLS CRUISE COMMUTATOR, DATA SWITCHES & DRIVERS
T37C	24	.5749	3.68	SERIES ACTIVE REDUNDANT CBS CRUISE MONITOR CONTROL DATA SWITCHES & DRIVERS
T12C	28	.5875	4.28	REDUNDANT ADAPTOR CRUISE COMMUTATOR AND CRUISE ENCODER
E1A	50	.6626	10.26	CBS & ESP BATTERY REDUNDANCE PROVIDED BY THE SLS BATTERY
E11C	165	.6788	14.26	ACTIVE REDUNDANT DC–DC CONVERTER REGULATORS
E17S	245	.6804	14.80	ACTIVE REDUNDANT BATTERY CHARGER RELAY
E18S	245	.6819	15.34	ACTIVE REDUNDANT BATTERY CHARGER RELAY
E15S	245	.6834	15.89	ACTIVE REDUNDANT BATTERY CHARGER RELAY
E16S	245	.6849	16.43	ACTIVE REDUNDANT BATTERY CHARGER RELAY
E16E	287	.7006	22.94	STANDBY REDUNDANT BATTERY FLOAT CHARGERS
E23C	287	.7167	29.44	PROVIDE STANDBY REDUNDANT BATTERY FLOAT CHARGERS
S1C	347	.7176	29.86	REDUNDANT REEFING CUTTERS FOR EACH PARACHUTE REEFING LINE (1 OF 3 REQUIRED)
T8C	356	.7233	32.72	STANDBY REDUNDANT CBS COMMUTATOR AND ENCODER
G1C	365	.7334	37.76	MULTI AXIS GYRO SENSING
E18C	440	.7350	38.73	ACTIVE REDUNDANT COMMAND DECODER RELAYS
E19C	440	.7366	39.71	ACTIVE REDUNDANT CRUISE COMMUTATOR RELAYS
E13C	441	.7383	40.68	ACTIVE REDUNDANT BATTERY CHARGER RELAYS
E12E	441	.7399	41.66	ACTIVE REDUNDANT BATTERY CHARGER RELAYS
T20E	478	.7450	44.91	ACTIVE REDUNDANT ESP TV BUFFER
T17E	525	.7477	46.86	ACTIVE REDUNDANT ESP TV DATA PROCESS ELECTRONICS
T7C	567	.7489	47.78	STANDBY REDUNDANT CBS PROGRAMMER
G8C	581	.7574	54.28	FOUR LANDING RADAR VELOCITY SENSOR CHANNELS (3 REQUIRED)
T28S	611	.7619	57.91	STANDBY REDUNDANT SLS TV DATA PROCESS
T16E	649	.7631	58.95	ACTIVE REDUNDANT ESP SCIENCE DATA REMOTE INTERFACE ELECTRONICS
T18E	660	.7664	61.81	STANDBY REDUNDANT ESP COMMUTATOR AND ENCODER
G3C	676	.7919	83.93	ONE OF TWO G AND C COMPUTERS SELECTED DURING IN-FLIGHT CHECKOUT
T13E	738	.7929	84.84	STANDBY REDUNDANT ESP PROGRAMMER
T24S	819	.8015	93.72	STANDBY REDUNDANT SLS COMMUTATOR AND ENCODER
T26S	859	.8024	94.62	STANDBY REDUNDANT SLS CONVOLUTION CODER
G10C	900	.8074	100.22	ACTIVE REDUNDANT RECEIVERS AND TRACKERS IN RADAR ALTIMETER
T36S	962	.8106	104.02	ACTIVE REDUNDANT SLS COMMAND SUBSYSTEM DECODER
T30S	987	.8256	122.14	FUNCTIONAL REDUNDANT SLS TAPE RECORDER STORAGE
H3E	1055	.8259	122.49	ACTIVE REDUNDANT THERMOSTATS
H2C	1101	.8273	124.45	ACTIVE REDUNDANT THERMOSTATS
T21S	1184	.8305	128.98	STANDBY REDUNDANT SLS PROGRAMMER
S2C	1241	.8325	131.96	DUAL CARTRIDGE EXPLOSIVE BOLTS – CAPSULE BUS/ADAPTER SEPARATION
G2C	1260	.8349	135.60	ONE OF TWO ACCELEROMETERS AND ELECTRONICS SELECTED DURING CHECKOUT
T5S	1312	.8422	147.01	FUNCTIONAL REDUNDANT SLS LOW RATE RADIO LINK
T3S	1318	.8533	164.23	FUNCTIONAL REDUNDANT SLS MONOPULSE TRACKING
C29C	1328	.8534	164.36	ACTIVE REDUNDANT CRYSTAL CONTROLLED OSCILLATORS. (CBS)
C24S	1349	.8535	164.53	ACTIVE REDUNDANT CRYSTAL CONTROL OSCILLATORS (SLS)
S3C	1391	.8545	166.20	DUAL CARTRIDGE EXPLOSIVE BOLTS – DEORBIT MOTOR RELEASE
S5C	1417	.8555	167.91	DUAL CARTRIDGE EXPLOSIVE BOLTS – AEROSHELL RELEASE
P1C	1436	.8560	168.77	REDUNDANT CARTRIDGE IN EACH OF 3 N.C. PYRO VALVES (RCS)
S6C	1436	.8562	169.05	REDUNDANT INITIATORS IN PARACHUTE CATAPULT

	REDUNDANCY CODE (1)
	BASIC SYSTEM
(7)	T15E
(7)	T1E
(7)	T14E
(7)	T23S
(7)	E1A
(7)	T22S
(7)	E11C
(7)	T11C
(7)	T10C
	E23C
(7)	S1C
	G1C
	E19C
	E18C
	E13C
(7)	T37C
(7)	G8C
(7)	T12C
	G3C
	E16E
(7)	G10C
	E17S
	E18S
	E15S
	E16S
	H2C
(7)	T2C
(7)	S2C
	E12E
	G2C
	C29C
	T20E
(7)	S3C
(7)	S5C
	P1C
(7)	S6C
	C46C
	T17E
	C42C
(7)	S4C
	T16E
	T18E
	P2C
	T13E
(7)	G9C
	H1C
	C11C
	T28S
	G4C
	C37C
	H3E
	T24S

(1) THE FIRST LETTER IN THE COLUMN IDENTIFIES THE SUBSYSTEM.

C – SEQUENCER AND TIMER　L – LANDING　H – THERMAL CONTROL
G – GUIDANCE AND CONTROL　P – PROPULSION　T – TELECOMMUNICATIONS
E – ELECTRICAL POWER　S – STAGING (SEPARATION, DEPLOYMENT AND RELEASE DEVICES)

THE LETTER FOLLOWING THE NUMBER IDENTIFIES THE SYSTEM.

A – ALL THREE SYSTEMS　E – ENTRY SCIENCE PACKAGE
C – CAPSULE BUS SYSTEM　S – SCIENCE LABORATORY SYSTEM

(2) DOES NOT INCLUDE EXPERIMEN
(3) INCLUDES A WEIGHT FACTOR
(4) ACHIEVEMENT OF A FLIGHT CA
(5) PERFORMANCE OF ENTRY EXPE
(6) PERFORMANCE OF LANDED EXI
(7) INCORPORATED REDUNDANCIES

Figure 4.6–4

ΔW/ΔE	EFFECTIVENESS $E = V_1R_1 \cdot V_2R_2 \cdot V_3R_3$	REDUNDANCY WEIGHT (3)	RELIABILITY (2) (4)	(5)	(6)	REDUNDANCIES
–	.7103	–	.7793	.6761	.6480	
47	.7145	.19	.7798	.6873	.6480	STANDBY REDUNDANT ESP CRUISE ENCODER
59	.7200	.52	.7805	.7025	.6480	INTERLEAVE LOW RATE ESP DATA ON CBS RADIO LINK
71	.7286	1.10	.7815	.7256	.6480	SERIES ACTIVE REDUNDANT ESP CRUISE COMMUTATOR DATA SWITCHES & DRIVERS
94	.7315	1.37	.7820	.7256	.6587	STANDBY REDUNDANT SLS CRUISE ENCODER
99	.7941	7.34	.8311	.8196	.6990	CBS & ESP BATTERY REDUNDANCY PROVIDED BY SLS BATTERY
122	.8013	8.16	.8324	.8196	.7258	SERIES ACTIVE REDUNDANT SLS CRUISE COMMUTATOR DATA SWITCHES & DRIVERS
229	.8209	12.16	.8528	.8398	.7436	PROVIDE ACTIVE REDUNDANT DC–DC CONVERTER REGULATORS
286	.8217	12.35	.8547	.8398	.7436	STANDBY REDUNDANT CBS CRUISE ENCODER
392	.8233	12.94	.8588	.8398	.7436	SERIES ACTIVE REDUNDANT CBS CRUISE COMMUTATOR DATA SWITCHES & DRIVERS
399	.8422	19.45	.8785	.8590	.7607	STANDBY REDUNDANT BATTERY FLOAT CHARGERS
489	.8432	19.86	.8795	.8601	.7616	REDUNDANT REEFING CUTTERS FOR EACH PARACHUTE REEFING LINE (1 OF 3 REQUIRED)
511	.8549	24.90	.8917	.8720	.7722	MULTI AXIS GYRO SENSING.
618	.8568	25.87	.8937	.8739	.7739	ACTIVE REDUNDANT CRUISE COMMUTATOR RELAYS
618	.8587	26.85	.8957	.8759	.7756	ACTIVE REDUNDANT COMMAND DECODER RELAYS
620	.8607	27.83	.8977	.8778	.7773	ACTIVE REDUNDANT BATTERY CHARGER RELAYS
763	.8612	28.22	.8992	.8778	.7773	SERIES ACTIVE REDUNDANT CBS CRUISE MONITOR CONTROL DATA SWITCHES & DRIVERS
814	.8709	34.72	.9093	.8877	.7861	FOUR LANDING RADAR VELOCITY SENSOR CHANNELS, (3 REQUIRED)
884	.8717	35.32	.9113	.8877	.7861	REDUNDANT ADAPTOR CRUISE COMMUTATOR AND CRUISE ENCODER
937	.9007	57.44	.9416	.9172	.8122	ONE OF TWO G AND C COMPUTERS SELECTED DURING IN-FLIGHT CHECKOUT
1139	.9084	63.95	.9424	.9383	.8122	STANDBY REDUNDANT BATTERY FLOAT CHARGERS
1264	.9141	69.54	.9483	.9441	.8173	ACTIVE REDUNDANT RECEIVERS AND TRACKERS IN RADAR ALTIMETER
1406	.9146	70.09	.9484	.9441	.8191	ACTIVE REDUNDANT BATTERY CHARGER RELAY
1406	.9151	70.63	.9485	.9441	.8209	ACTIVE REDUNDANT BATTERY CHARGER RELAY
1406	.9155	71.17	.9485	.9441	.8227	ACTIVE REDUNDANT BATTERY CHARGER RELAY
1406	.9160	71.72	.9486	.9441	.8246	ACTIVE REDUNDANT BATTERY CHARGER RELAY
1548	.9177	73.67	.9503	.9458	.8260	ACTIVE REDUNDANT THERMOSTATS
1566	.9179	73.99	.9509	.9458	.8260	INTERLEAVE LOW RATE CBS DATA ON ESP RADIO LINK
1745	.9201	76.97	.9532	.9481	.8280	DUAL CARTRIDGE EXPLOSIVE BOLTS – CAPSULE BUS/ADAPTER SEPARATION
1767	.9209	77.95	.9533	.9502	.8280	ACTIVE REDUNDANT BATTERY CHARGER RELAYS
1771	.9236	81.59	.9561	.9529	.8304	ONE OF TWO ACCELEROMETERS AND ELECTRONICS SELECTED DURING CHECKOUT
1869	.9236	81.72	.9562	.9530	.8305	ACTIVE REDUNDANT CRYSTAL CONTROLLED OSCILLATORS
1914	.9260	84.97	.9564	.9595	.8305	ACTIVE REDUNDANT ESP TV BUFFER
1958	.9271	86.64	.9576	.9607	.8315	DUAL CARTRIDGE EXPLOSIVE BOLTS – DEORBIT MOTOR RELEASE
1994	.9283	88.35	.9587	.9618	.8325	DUAL CARTRIDGE EXPLOSIVE BOLTS – AEROSHELL RELEASE
2022	.9288	89.21	.9593	.9624	.8330	REDUNDANT CARTRIDGE IN EACH OF 3 N.C. PYRO VALVES (RCS)
2022	.9290	89.49	.9595	.9626	.8331	REDUNDANT INITIATORS IN PARACHUTE CATAPULT
2058	.9310	92.66	.9616	.9647	.8349	DUPLEX MEMORIES AND MEMORY BUFFER REGISTORS WITH ERROR DETECTION
2106	.9323	94.61	.9617	.9683	.8349	ACTIVE REDUNDANT ESP TV DATA PROCESS ELECTRONICS
2147	.9328	95.47	.9623	.9688	.8354	TRIPLE REDUNDANT FREQUENCY DIVIDERS WITH MAJORITY VOTERS
2258	.9340	97.39	.9634	.9700	.8364	DUAL CARTRIDGE EXPLOSIVE BOLTS – PARACHUTE RELEASE
2602	.9345	98.43	.9635	.9715	.8364	ACTIVE REDUNDANT ESP SCIENCE DATA REMOTE INTERFACE ELECTRONICS
2646	.9361	101.30	.9636	.9758	.8364	STANDBY REDUNDANT ESP COMMUTATOR AND ENCODER
2919	.9374	104.28	.9650	.9772	.8376	SERIES REDUNDANT PRESSURE REGULATOR (RCS)
2960	.9379	105.19	.9651	.9784	.8376	STANDBY REDUNDANT ESP PROGRAMMER
3034	.9387	107.07	.9659	.9792	.8384	ACTIVE REDUNDANT TRANSMITTER TUBES IN RADAR ALTIMETER
3096	.9395	109.02	.9668	.9801	.8391	ACTIVE REDUNDANT RESISTANCE HEATERS
3430	.9396	109.15	.9668	.9802	.8391	INCORPORATE TRIPLE REDUNDANT DECREMENTERS AND ZERO DETECTORS
3493	.9409	112.78	.9671	.9802	.8441	STANDBY REDUNDANT SLS TV DATA PROCESSER
3572	.9450	123.86	.9713	.9844	.8478	VELOCITY AND RANGE SENSOR REDUNDANCIES IN LANDING RADAR
3966	.9455	125.34	.9718	.9850	.8483	ACTIVE REDUNDANT DISCRETE OUTPUT LINE DRIVERS
4240	.9456	125.69	.9718	.9853	.8483	ACTIVE REDUNDANT THERMOSTATS
4663	.9481	134.56	.9722	.9853	.8575	STANDBY REDUNDANT SLS COMMUTATOR AND ENCODER

TS

PSULE LANDING

:RIMENTS

'ERIMENTS

design constraints. These included four items in the Surface Laboratory. Functionally redundant high gain antenna pointing and steering equipment (monopulse tracking) and a sun sensor will enhance the probability of successful high rate data transmission. An S-band low data rate radio link with a solid state transmitter is less vulnerable to environmental and operational hazards than the high rate link; it will provide engineering and science data. Dual cartridge pyrotechnic devices are provided for deployment of Surface Laboratory experimental equipment.

The systems effectiveness analysis was used as a guide in selecting redundancies for the preferred design. The first 45 redundancies listed in Figure 4.6-4 were considered for the preferred Flight Capsule design, and 21 were selected. Our analysis showed that the reliability for performing Surface Laboratory experiments was substantially improved by improving the reliability for achieving landing. Accordingly, although 18 redundancies were incorporated to improve the reliability for landed experiments, 16 of them are components of the Capsule Bus. They increase the reliability for landed experiments to 78%.

Addition of redundant components increases both vehicle weight and reliability, as illustrated in Figure 4.6-5. Point A on this figure represents a design without redundancy and point B a design including the six items necessary to meet the constraints. Our preferred design, with a reliability of all equipment of .71, is represented by point C. The potential reliability if all 45 of the recommended redundancies could be incorporated is indicated by point D. The line from A to B to D represents the maximum reliability available when the design includes the six constraint-required items.

The parallel paths represented by use of two Planetary Vehicles improves the probability of successful landing. For our preferred design, the reliability of all equipment (excluding experiments) for performing successful Surface Laboratory experiments with at least one of the two capsules is 95%. All of these estimates are conditional upon the successful operation of the Flight Spacecraft, the Launch Vehicle, and other VOYAGER systems which support the Flight Capsule.

RELIABILITY IMPROVEMENT THRU EFFECTIVENESS ANALYSIS

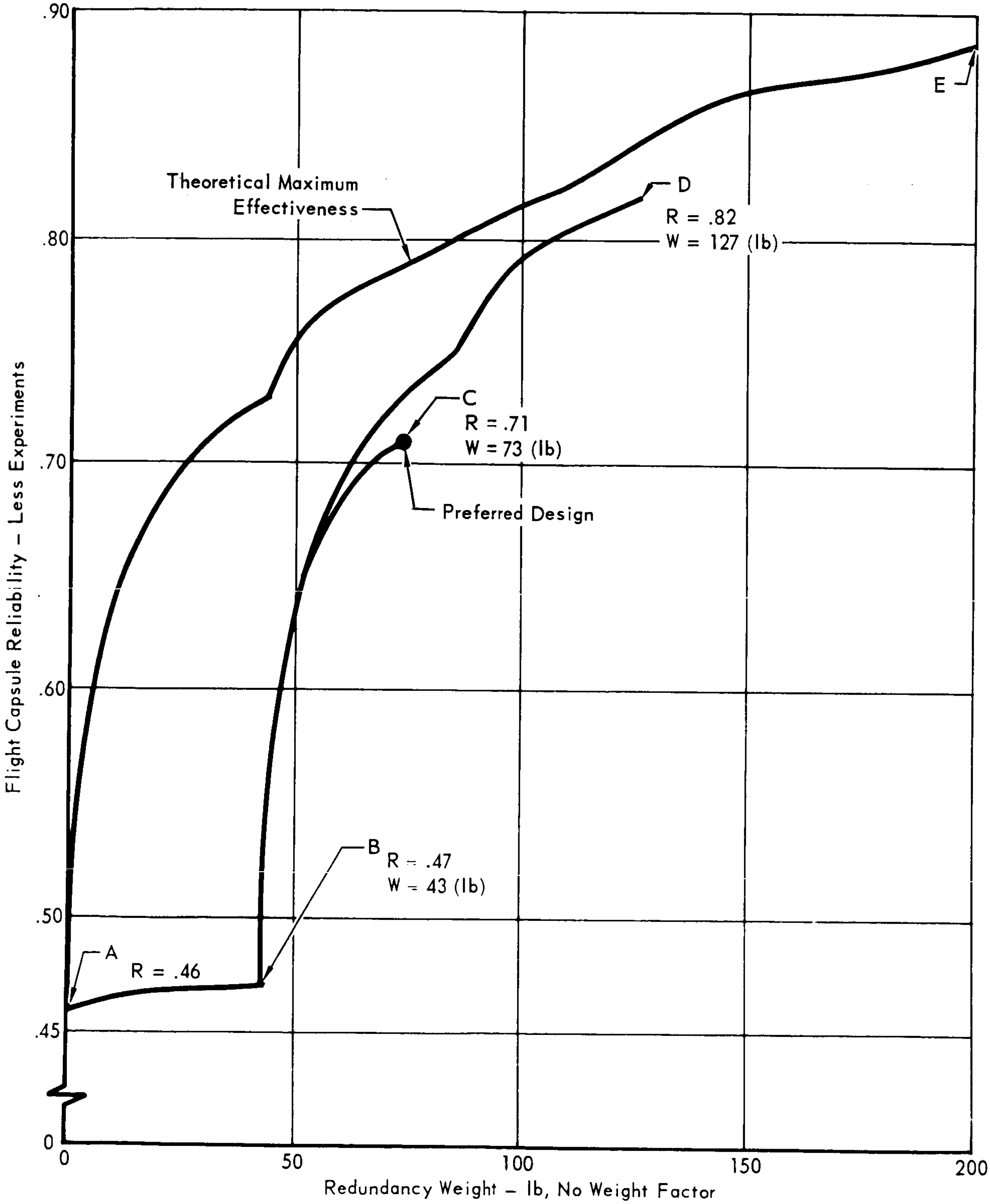


Figure 4.6–5

SECTION 4

REFERENCES

4.6-1 VOYAGER Reliability/Effectiveness Study Methods, McDonnell Report F534, dated 10 May 1967

4.6-2 General Specification for Performance and Design Requirements for the 1973 VOYAGER Mission, Jet Propulsion Laboratory SE 002 BB 001-1B21, dated 1 January 1967

4.7 EXTENDED MISSION STUDY - A subsystem study was conducted to assess the desirability of an extended mission duration for the 1973 VOYAGER Surface Laboratory. The power/thermal energy sources that were evaluated to determine their effect on mission life include batteries, isotope, and chemical heat sources in varying combinations with solar cell, RTG, and fuel cell power generation techniques. Science sequences were generated to match the additional power available and a scientific value was established for each power/thermal technique. The value of an extended mission was assessed by comparing the weight penalty, reliability, and system constraints incurred by these methods with the scientific value achievable as a function of the additional time on Mars.

The study shows that an RTG/Battery subsystem can extend the mission to 30 days with the least penalty. The 30-day extension doubles the scientific value of the basic one-diurnal-cycle mission. A solar array/battery/isotope heater system has the next lowest weight for a 30-day mission; however, it is limited to a small range of landing site latitudes. In addition, the effect on power output of long duration cloud coverage has not been assessed. If sterilizable and available for the 1973 mission, fuel cells can provide a competitive weight subsystem for durations up to about 12 days. The scientific value of the 12 day mission is 50% greater than the basic one-diurnal-cycle mission.

4.7.1 Objectives - The extended mission study objectives are to:

a. Determine scientific value of an extended mission.
b. Assess the power/thermal combined subsystem approaches to define energy sources capable of supporting an extended science program.
c. Determine the mission value of the increased quantity of science information obtained, and the effect on Surface Laboratory (SL) performance.

4.7.2 Constraints - One major study goal was to achieve a mission value as high as possible from extended operations without jeopardizing the basic one-diurnal-cycle mission. Thus, the extended life study was constrained by the SL design requirements and constraints noted in Section A2. The specific subsystem requirements and constraints directly applicable to the study include:

a. Power/thermal subsystem weight limited to 300 lb for the basic mission
b. Landing site between 40^{o}S and 10^{o}N latitudes, spring-time landing
c. Internal temperature maintained between 40 and 120^{o}F
d. Evening landing
e. Hardware state-of-the-art by 1969

These requirements must be satisfied within the following environmental constraints:

a. Surface temperature extremes, $150^{o}K$ and $320^{o}K$

b. Constant low temperature surface environment, $150^{o}K$

c. Solar constant, 160 to 230 Btu/hr-ft^2

d. Normal transmission of solar energy, 0.7

e. Wind velocity, 110 to 220 ft/sec (per VM-1 through 10)

f. Subsystem dry heat sterilization at $135^{o}C$

g. Decontamination using 12% ethelyene oxide and 88% Freon-12 at $50^{o}C$

h. Terminal sterilization cycle at $125^{o}C$ for 24.5 hrs.

4.7.3 Science Consideration - The "science value" of a mission is defined relative to the mission goals. Techniques are available to numerically determine the relative merits of science payload configurations based on assigned priority values for the specified mission goals. These same techniques can be extended to select the payloads which will optimize the mission science value as a function of mission duration. It was not the purpose of this effort to consider all possible experiments but to apply the technique to the baseline experiments and to assess their extended mission value.

Study ground rules include: the science payload is the 1973 mission baseline; the maximum extension to be considered is 30 days; the Surface Laboratory is immobile; each of the experiments, when gathering essentially new information, contributes equally to the collective science value of the mission (a more thorough analysis would take this assumption to task). The problem confronted here is to determine what constitutes essentially new information.

The experiment is judged to have half the value of an hourly variation determination after a half hour of operation. The complete value of an hourly variation is accomplished at the end of 3 hours. Based on the new information theory described earlier, no significant increase in value of this experiment occurs until the next phenomenological change, which in this case is 1 day. The diurnal variation is not totally confirmed until the second measurement set completes about 3 days of cycling. In a like manner, the total value of the experiment as applied to a weekly variation is not accomplished until a 30 day period of cycling is completed. However, partial value is obtained on a weekly basis. Thus, an experiment that has phenomenological changes over large periods can be represented by a higher overall science value than an experiment that does not collect new information as a function of time.

A value of a science program, as a function of extended mission period, can now be derived from the information in Figure 4.7-1. The programs selected for evaluation are:

Program 1 (1973 preferred)	Program 2	Program 3	Program 4
Imaging Atmospheric Humidity Pressure, Temperature, and Wind Spectro Radiometer Subsurface Probe Life Detection Element Spectrometer	Delete Imaging	Delete Imaging and Element Spectrometer	Delete Imaging, Element Spectrometer, and Sub-Surface Experiment

The value of these programs, normalized to the baseline program value for the basic 1-day operation, are shown in Figure 4.7-2. It must be repeated that for this analysis each payload contributes equally to the collection of scientific value of the mission. The actual "science value" of an experiment when taken into the overall context of the mission goal can only be defined by NASA.

4.7.4 Thermal and Power Subsystem Characteristics - The 1973 mission as presently envisioned utilizes primary batteries as the power and thermal source for the basic one-diurnal-cycle mission. Approximately 40% of the installed battery capacity is required by the thermal control design point of continuous $-190^{\circ}F$ temperatures. About 8% of the battery capacity is used each night to operate electrical heaters for temperature control in a cyclic environment. Auxiliary heaters (isotope or chemical) permit battery capacity to be utilized for either additional science or mission life extension. However, the available battery power probably does not extend the mission beyond a second day, so other techniques were pursued.

Photo-voltaics are a source of limited primary power and of indeterminant energy to recharge the batteries. This technique would only be used to complement the battery system for the primary mission. Isotope heaters would be required (chemical heaters are too expensive in weight) to maintain temperature control during the night or continuous shade periods whether the batteries are operating or not.

A Radioisotope Thermoelectric Generator (RTG) electrical power source is attractive for space missions of long duration which preclude the use of batteries or fuel cells, and for missions where an appreciable portion of the mission cycle must be conducted in areas of weak or no solar flux, compromising the use of solar cell arrays.

INSTRUMENT MATRIX

INSTRUMENT	SAMPLING PHILOSOPHY NEW SAMPLE TAKEN AT	VALUE vs TIME	
		TIME	VALUE
Imaging	2 per min + 4 – 5 hrs	4 hrs	1.0
	+ 15 – 17 hrs	16 hrs	1.5
	+ 20 – 30 days	26 days	3.0
Atmospheric Composition	2 per min + 16 hrs	16 hrs	0.5
	+ 22 hrs	22 hrs	1.0
	+ 40 hrs	40 hrs	1.15
	+ 46 hrs	48 hrs	1.25
	+ 64 hrs	64 hrs	1.4
	+ 70 hrs	70 hrs	1.5
	+ 30 day	30 days	2.25
Humidity	2 per min + 1 hr	1 hr	0.25
	+ 6 hrs	6 hrs	0.50
	+ 21 hrs	21 hrs	0.75
	+ 24 hrs	24 hrs	1.0
	+ 3 days	3 days	1.25
	+ 15 days	15 days	1.35
	+ 30 days	30 days	2.0
Pressure, Temperature and Wind Velocity	96 samples on 1st day, 36 samples on 2nd & 3rd days. Repeat this cycle again starting on 7th, 14th, 21st & 28th days.	3 hrs	0.5
		12 hrs	1.0
		24 hrs	1.5
		3 days	2.0
		10 days	2.25
		17 days	2.5
		24 days	2.75
Spectro-Radiometer (U.V. Spectrometer)	12 samples around terminator for 1st three occurrences of terminator. 6 samples around noon of 1st, 3rd, 15th & 30th days. 2 samples around terminators of 3rd, 15th & 30 days.	1 hr	0.5
		4 hrs	1.0
		18 hrs	1.25
		25 hrs	2.0
		29 hrs	2.25
		48 hrs	2.65
		3 days	3.0
		15 days	3.65
		30 days	4.75
Element Spectrometer	2 per min + 6 hrs	6 hrs	1.0
	+ 18 hrs	18 hrs	2.0
	+ 30 hrs	30 hrs	3.0
	+ 9 days	9 days	4.0
	+ 17 days	17 days	5.0
Surface Probe (Temperature sensor soil heat conductivity)	1 sample per hour for 24 hrs, 1 sample per 3 hours for 48, repeat 1 sample per 3 hours on 15th and 30th day	6 hrs	0.5
		18 hrs	1.5
		24 hrs	1.75
		48 hrs	2.0
		3 days	2.0
		9 days	2.5
		17 days	13.0
Life Detection	1 sample every 15 min for 3 hours to 1 day, 1 sample every 30 min. for 5 hours every day after	8 hrs	1.0
		22 hrs	3.0
		3 days	5.0
		13 days	8.0
		22 days	9.0
		30 days	10.0

Figure 4.7–1

NORMALIZED SCIENCE VALUE FOR FOUR SCIENCE PROGRAMS AS A FUNCTION OF EXTENDED MISSION STUDY

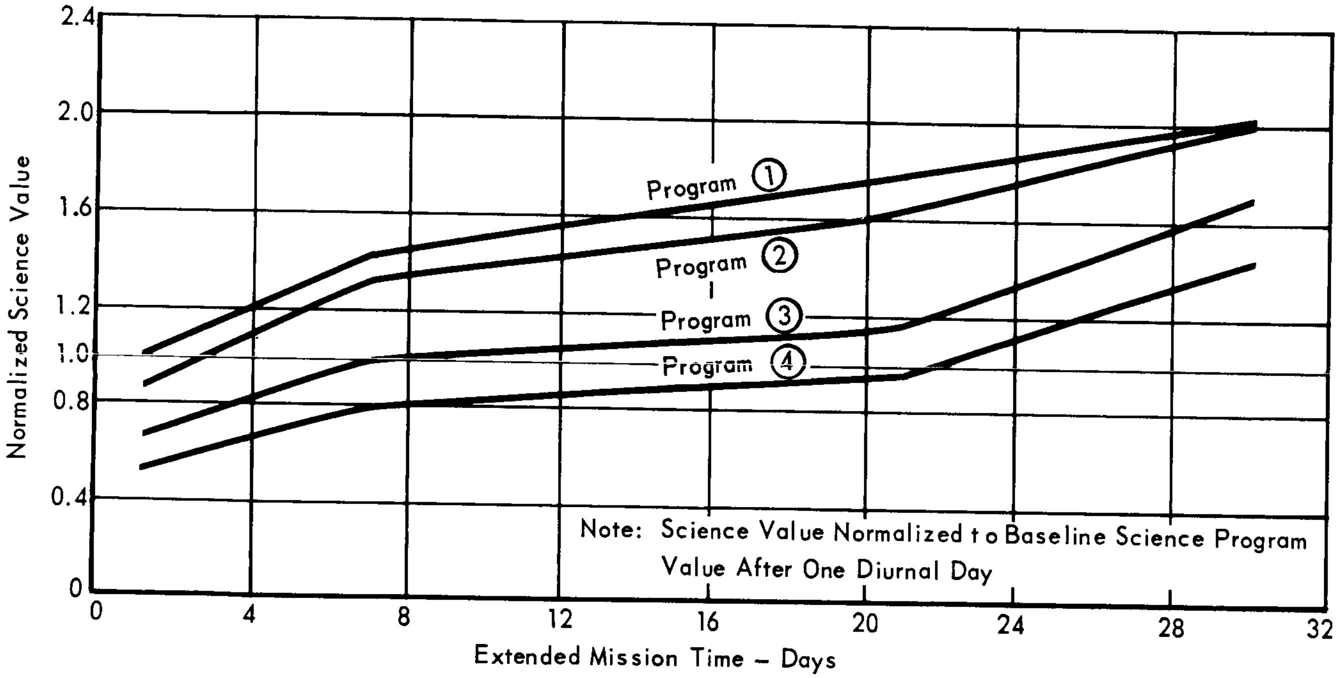


Figure 4.7–2

4.7.4.1 Thermal Control - The thermal control analysis and design is discussed in more detail in Section 5.8, so only those aspects pertinent to the design will be presented here. The thermal design requires about 320 Watts (thermal) of internal heat generation to maintain a minimum temperature of 50°F within the Surface Laboratory during the continuous -190°F temperature condition. Approximately 4 in. of insulation is used. The internal heat is obtained from battery powered equipment and battery operated electrical heaters, the proportion depending on equipment operations.

To prevent internal temperature rise above 120°F during hot day periods, a radiator and a variable heat path device is used. During hot periods this device will short circuit the insulation, allowing rejection of equipment heat. During cold periods the heat path will be turned off, thus reverting to the insulated design.

The major result of the analysis is that, for the continuous -190°F environment, about 4000 Wh of thermal energy is required for thermal control, or close to 40% of the battery capacity. To reduce the battery energy requirements for thermal control, other sources of thermal energy with much higher specific thermal weight were examined, in particular, isotope heaters.

It is well established that isotope heat sources offer the potential of extremely low specific weight thermal energy sources. Typically, as little as a few grams of compounded isotope material provides thermal energy equivalent to one or more pounds of primary batteries and electrical heaters. Utilization of isotopic sources is frequently restricted by technical, economic, safety, and integration problems. These become most pronounced when relatively large quantities of isotope material are required such as the case with RTG's. Conversely, when relatively small quantities of isotope material are involved, most of these problems are minimized. A specific example is the 5 W to 10 W, Promethium-147 isotope heaters manufactured by Atomics Internation, for which essentially all aerospace nuclear safety standards and resultant design requirements are met. Virtually all development requirements have been satisfied and the residual problem is the specification of design techniques which accommodate the continuous heat generation characteristics of the isotope. Where isotope heater weight would be essentially constant for an extended mission to, say, 30 days, heater battery weight becomes prohibitive after several days.

The problem of handling the isotope continuous heat dissipation, though definitely not insolvable, led to the investigation of chemical heaters that provide heat only when required. The use of chemical reactions to provide thermal energy has several advantages. These include: (1) direct chemical combination of materials in exothermic reaction provides high energy densities; (2) temperature control can be achieved by varying the rate at which reactants are brought together, thereby meeting a variety of load requirements; (3) chemical systems can be devised which will produce little or no gas; and (4) suitable chemical systems are envisioned which can undergo thermal sterilization with little or no degradation of performance. An example of a system that meets this criterion is one consisting of pressurized liquid chlorine mixed on demand with magnesium. The estimated energy density, including constituents, containers, control values, and sensors, is about 375 Wh/lb. The disadvantages of this system lie in the reliability problem associated with the extreme corrosiveness of chlorine. The system is also specific weight limited for missions beyond 4-5 days.

Another source of heat is the waste heat of an RTG should this approach prove feasible for the 1973 mission. The "variable resistance device" could be attached to the radiating fins to allow the heat to enter the Surface Laboratory. The device can shut off the heat supply during day operation. Another approach would allow the RTG to radiate to the interior of the Surface Laboratory through a set of louvers. The louvers would close or open in a direction that reflects the RTG heat load away from the Surface Laboratory while allowing internal heat to radiate away during the hotter phase of the mission. The inherent problems associated with the RTG, as discussed previously, and the complex design technique required to utilize the excess heat make this approach questionable for the 1973 mission.

4.7.4.2 Electric Power

Battery - A detailed assessment of the battery is performed in Volume III Part B. The salient feature of that discussion is the estimated energy density of sterilizable silver-zinc batteries as reported by General Electric Co., Electric Storage Battery Co., and the Douglas Astropower Division. The results range from 20 to 40 Wh/lb depending upon the discharge rate and capacity. A value of 26 Wh/lb was used for the Capsule Bus because the discharge rate of about C/4 was used. The Surface Laboratory discharge is at a 30 hr rate (C/30); thus a higher figure of 35 Wh/lb was used. This is a reasonable assumption since the Eagle Pitcher batteries tested by General Electric Co. survived cycling (11 cycles to date) after sterilization and a 10 month wet stand period, and still maintained 98% of their initial capacity. For this study an energy density of 35 Wh/lb is used.

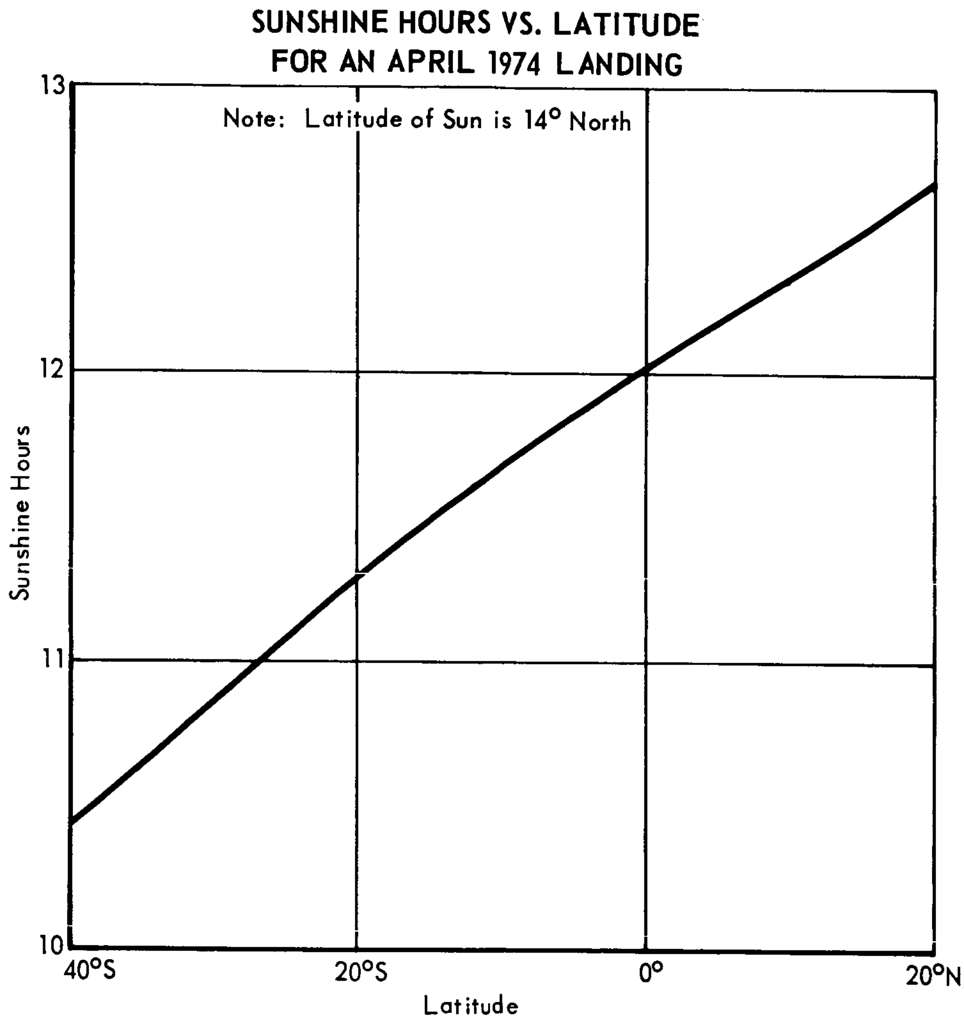

SUNSHINE HOURS VS. LATITUDE
FOR AN APRIL 1974 LANDING
Note: Latitude of Sun is 14° North
13
12
11
10
Sunshine Hours
40°S
20°S
0°
20°N
Latitude

Figure 4.7-3

Solar Arrays - The solar array investigations were predicated on well established sizing factors, and estimated degradation factors, with a resulting high uncertainty.

The sizing loss factors are:

o Packing Density	0.92
o Temperature Extreme ($50^{o}C$)	0.96
o Blue Filter, Adhesive, and AR Coatings	0.96
o Maximum Mars Orbit Ellipticity Factor	0.84
o Series and Parallel Wiring	0.99
o Diode Losses	0.98
o Average Mars-Sun/Earth-Sun Intensity Radio	0.43
o Cell Efficiency @ $20^{o}C$	0.12
o Cell Mismatch	0.95
Product:	0.034

Thus, the array power output is 0.034 times the mean solar flux at 1.0 Astronomical Unit (130 W/ft^2) or 4.4 W/ft^2 of array area. This must be further reduced by the following environmental factors:

o Martian Particulate Radiation & U.V.	0.99
o Martian Atmosphere Transmission (Ref. 4.7-1)	0.70
o Martian Dust Erosion	0.98
o Yellow Dust Cloud Transmission	0.77
Product:	0.52

Including the environmental factors reduces the power output to about 2.4 W/ft^2 of array. The number of sunlight hours when solar energy can be collected is plotted in Figure 4.7-3 for an April landing in 1974. (Sun/Mars zenith at $14^{o}N$ latitude). The energy output for panel type and windowshade type arrays as a function of latitude for various orientation conditions is given in Figure 4.7-4. It is interesting to note that under the worst landing orientation condition of 34^{o} slope away from the Sun, a fixed array has no sun impingement at a landing site of $40^{o}S$. The fixed array is drastically reduced in performance for latitudes lower than $10^{o}S$ under this worst landing orientation condition.

The specific weight and volumetric efficiencies of typical deployable arrays are given in Figure 4.7-5. This is a compilation of information from contractors (Boeing, Hughes, Ryan, and Goodyear) who are presently doing research and development in the field of large deployable solar arrays.

ORIENTATION CONDITION	POWER OUTPUT (WATT-HR/FT^2/DAY)							
	PANEL ARRAY*				WINDOW SHADE ARRAY			
LATITUDE –	40°S	20°S	0°	20°N	40°S	20°S	0°	20°N
1. – Fixed Array, Perpendicular to Suns Rays at Hi Noon	13.4	14.5	15.4	16.2	15.4	16.7	17.7	18.6
2. – Fixed Array, Perpendicular to Local Vertical	7.9	12.0	14.9	16.1	9.1	13.8	17.1	18.5
3. – Fixed Array, Parallel to a 34° Slope Away from Sun	0	5.4	10.3	12.8	0	6.2	12.0	14.7
4. – Tracking Array	21.9	23.3	24.2	26.6	25.2	26.8	27.8	30.6

*Packing factor for fold-out panel 0.8, for windowshade 0.92

Figure 4.7–4

CONTRACTOR	TYPE	ARRAY CHARACTERISTICS		COMPLETION DATE
		LB/FT^2	CU FT/FT^2	
Boeing	Folding Modular Solar Array and Associated Equipment	0.6	.06–.08	1969
Hughes	Flexible Arrays Using Dendritic Cells	0.4	.03–.06	1968
Ryan	"Window Shade" Large Area Array	0.3	0.03	1968
Goodyear	Circular Petal Array; "Lazy-Tong" Sandwich Panel; Inflatable Substrates	0.4–0.6	—	1969

Figure 4.7–5

The volume available for stowage on the Capsule Lander is approximately 15 ft^3 for a deploying planar array (40" x 30" x 20") and 11 ft^3 for a "windowshade" cylinder (25" diameter by 40" long). Thus the available array area for the planar array (Boeing design) is 233 ft^2 and weighs about 117 lb. The power output for this approach is given below.

Orientation Condition	Power Output (Watt Hrs/Day)			
Latitude	(40°S)	(20°S)	(0°)	(20°N)
Fixed Array, normal to Sun	3120	3380	3600	3780
Fixed Array, normal to local vertical	1850	2800	3480	3760
Fixed Array, parallel to 34° slope	0	1300	2400	2980
Tracking Array	5100	5440	5650	6200

At the worst design condition (34° slope) a fixed array could only support a very limited program with most of the energy being used to recharge the battery.

In contrast, the windowshade design would have about 370 ft^2 of array packaged into the 11 ft^3 and weigh about the same as the planar array, 111 lb. The power output for this approach is summarized below:

Orientation Condition	Power Output (Watt Hrs/Day)			
Latitude	(40°S)	(20°S)	(0°)	(20°N)
Fixed Array, normal to Sun	5700	6300	6550	7100
Fixed Array, normal to local vertical	3370	5100	6320	6850
Fixed Array, parallel to 34° slope	0	2400	4440	5430
Tracking Array	9300	9900	10300	11400

Though the power output and array weight in this approach are reasonable, the method of achieving 370 ft^2 of area poses a major design problem. Windowshade design studies to date have investigated only panels in the order of 25 to 30 ft^2. A design technique must be developed to extend the shade by approximately 90 ft^2. Also of major importance is the terrain effect on the solar cell/Sun angle over the complete array, and its ability to deploy the complete length.

A solar array approach to achieve additional power for an extended mission will still require a battery system since only 12.7 hours (maximum) of daylight are available and continuous power must be supplied to specific experiments and engineering equipment.

Radioisotope Thermoelectric Generator (RTG) - An RTG may be the only feasible low-weight power generator for continuous long duration missions since its power output is relatively unaffected by the anticipated Martian environments. To achieve a low system weight, a battery is included to handle the peak loads while the RTG is employed to supply the constant load and the energy to recharge the batteries. The weight savings may not be as large as one would expect since the battery specific energy and life is drastically reduced by the sterilization requirement and further compounded by the required number of charge and discharge cycles. Another factor to be considered is the increase in failure probability when using a battery and associated charger, particularly as the mission duration increases.

A rough calculation of required RTG electrical power yields 169 watts required to support the full science program defined in paragraph 4.7.3. One RTG system that could achieve that output is a modified SNAP-27. The SNAP-27 was designed for the Advanced Lunar Science Experimental Package (ALSEP). The specific energy output 1.09 watts (electrical) per pound of generator with a thermal heat output of about 1500 watts (t). Thus, three modified SNAP-27 units would be required resulting in an overall system weight of about 237 lb (82 lb of batteries and 155 lb of RTG).

The RTG waste energy is also available for Surface Laboratory thermal control. However, a major heat dissipation problem arises during the launch and cruise portions of the mission when close to 4.5 kW of thermal energy must be dissipated. The problem is further compounded since this heat must be removed through the Capsule Bus Sterilization Canister and Launch Vehicle shroud. Though the problem is not insurmountable, it does impose a severe design problem at many interfaces.

Additional drawbacks to this approach which also result in weight penalties and/or complex design requirements include the radiation emitted by the isotope fuel and the related nuclear safety aspects. Though dose rates can be minimized by shielding and/or separation, an accumulated dose may have significant effect on both electronics and scientific equipments as the mission is extended. The neutron dose rates of three 1.5 kW(t) RTG's is shown in Figure 4.7-6.

The safety problem manifests itself in the following categories: handling difficulties on the ground; accidental impact on Earth; and abort in the Earth's atmosphere. These problems have, in most cases, been resolved on the ALSEP SNAP-27 program but must be contended with for each new mission.

VOYAGER CAPSULE NEUTRON DOSE RATE ISOPLETHS

(MILLIREM/HR)

4500 Watts Total Installed Thermal Power (3 RTG Units)

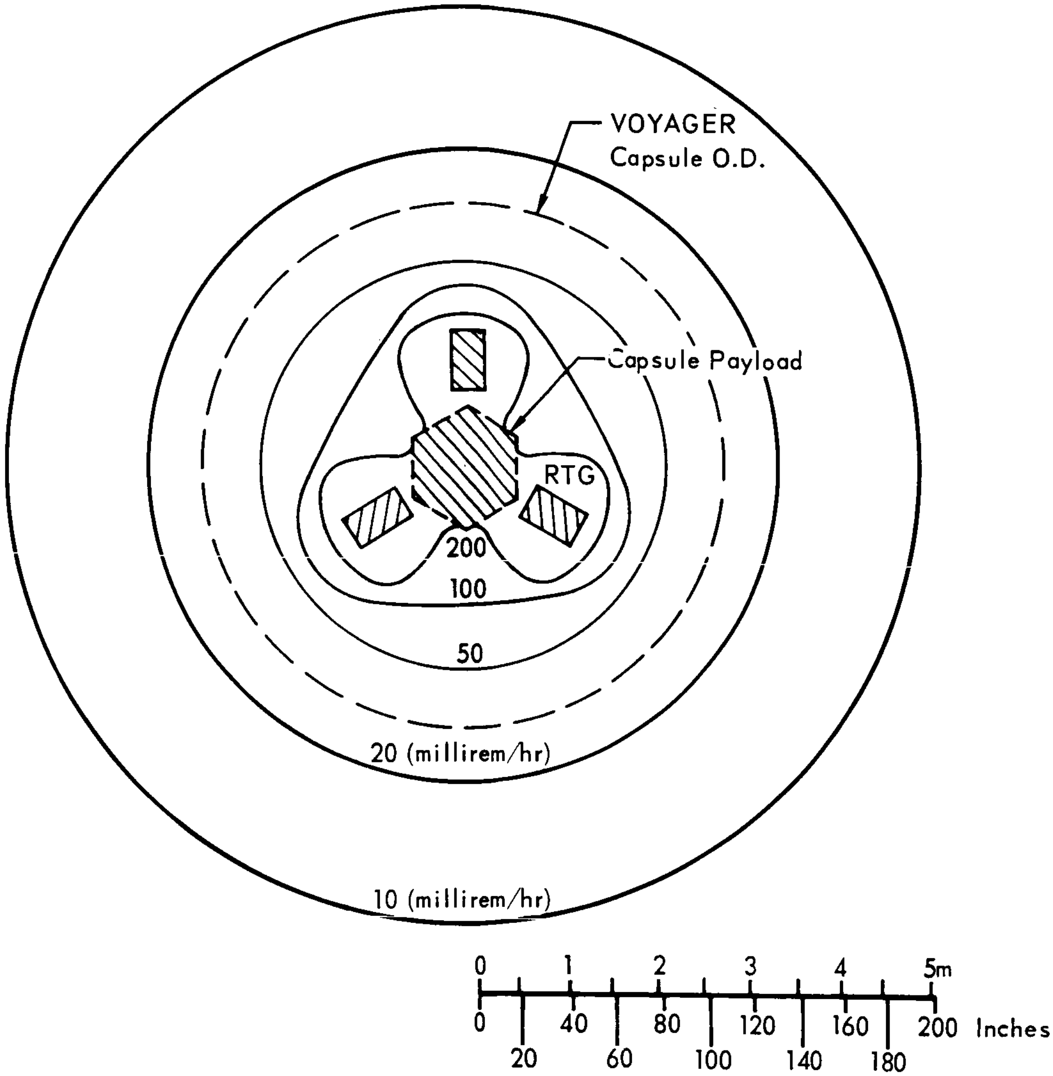


Figure 4.7-6

In view of the thermal dissipation problem, the complexity in design to minimize radiation dosage to equipment, the overall handling problems and the relatively small weight advantage when compared with a battery alone, the use of an RTG for short missions appears impractical. For missions up to 30 days, it is very attractive when compared on a weight basis to battery and solar cell sources of power.

Fuel Cells - Two types of fuel cells where investigated, the H_2-0_2 fuel cells made by the General Electric Co., Allis-Chalmers, Union Carbide, and Electro Optical Systems, and the Lithium-Chlorine fuel cell developed by the General Motors Defense Research Laboratory. Survey of the various H_2-0_2 fuel cells indicates that energy density ranges from 100 to 200 Wh/lb. The weight as a function of mission time for the two power profiles is shown in Figure 4.7-7 for the most optimistic energy density.

The Lithium-Chlorine fuel cell is a relatively new technology. The constituents can be stored as liquids and the heat sterilization temperature of 145°C does not cause excessive tank pressures. The power density is being quoted as at least double that of the H_2-0_2 fuel cell. Thus a Li-Cl fuel cell system weighing 300 lb could extend the full science mission out to 22 days.

The major limitations of the fuel cell approach are: (1) they are specific energy limited; (2) they may require development to withstand the sterilization requirement; and (3) the Li-Cl fuel cell requires a heat up period to start the reaction, thus an auxiliary heating system is initially required.

4.7.5 Subsystem Analysis - Examination of the science program power requirements leads to the two extreme power profiles shown in Figure 4.7-8. The maximum power profile includes the baseline science instrumentation and engineering equipment. The minimum power profile minimizes the high rate telecommunication operation and deletes the imaging system. The deletion of the elemental spectrometer and the gas chromatograph does not have an appreciable effect on the overall power requirements.

Battery Subsystems - The thermal heat requirements for temperature control in a -190°F environment is 320 Watts. This condition is the same for the minimum data transmission case as well as the full science program case. Thus, considering only batteries or batteries plus auxiliary heaters (isotope or chemical) as the power and thermal control source results in the additional system weight as a function of time beyond initial one diurnal cycle as shown in Figure 4.7-9.

H_2-O_2 FUEL CELL SYSTEM WEIGHT AS A FUNCTION OF MISSION EXTENSION AND POWER PROFILE

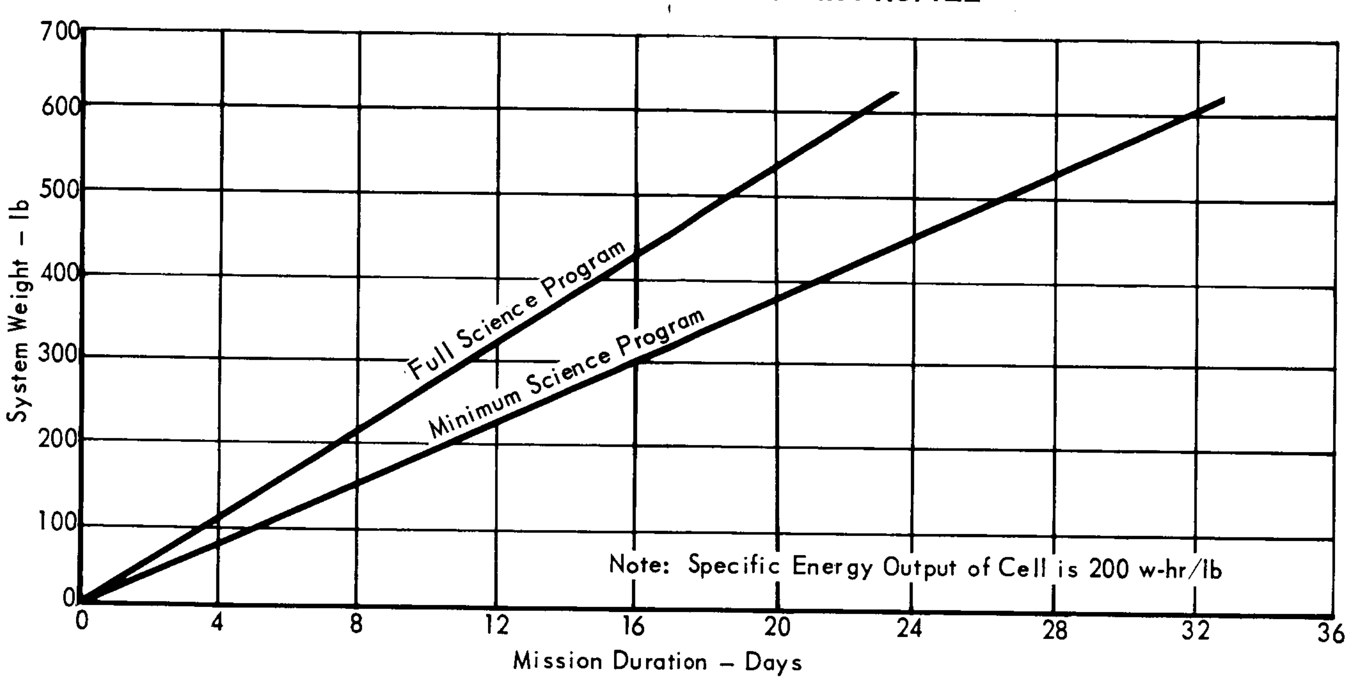


Figure 4.7–7

MISSION EXTENDED POWER PROFILE

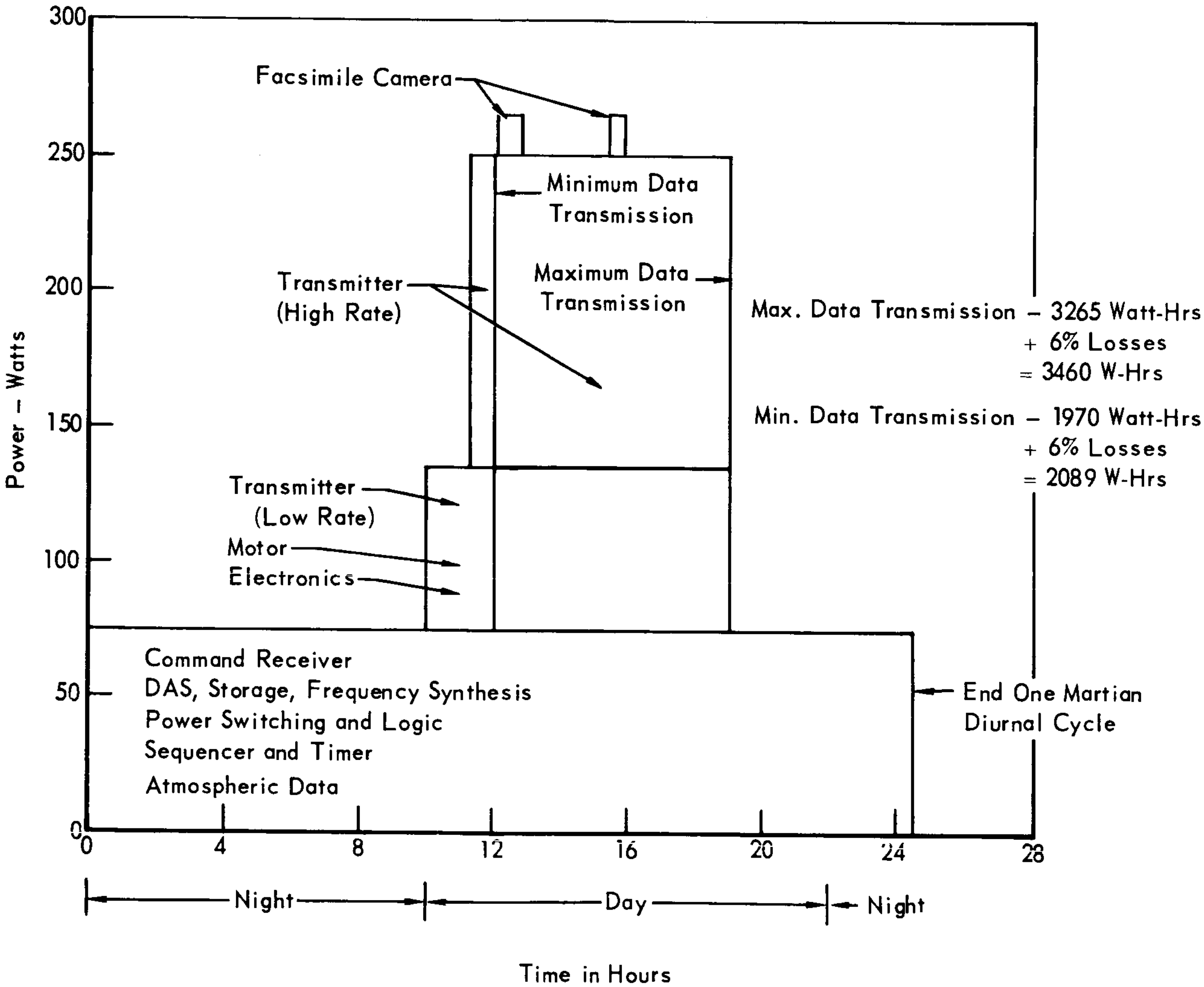


Figure 4.7–8

BATTERY SYSTEM WEIGHTS AS A FUNCTION OF TIME
FOR MAXIMUM AND MINIMUM SCIENCE PROGRAMS
(HEATING BY BATTERIES, CHEMICAL HEATERS, AND ISOTOPES)

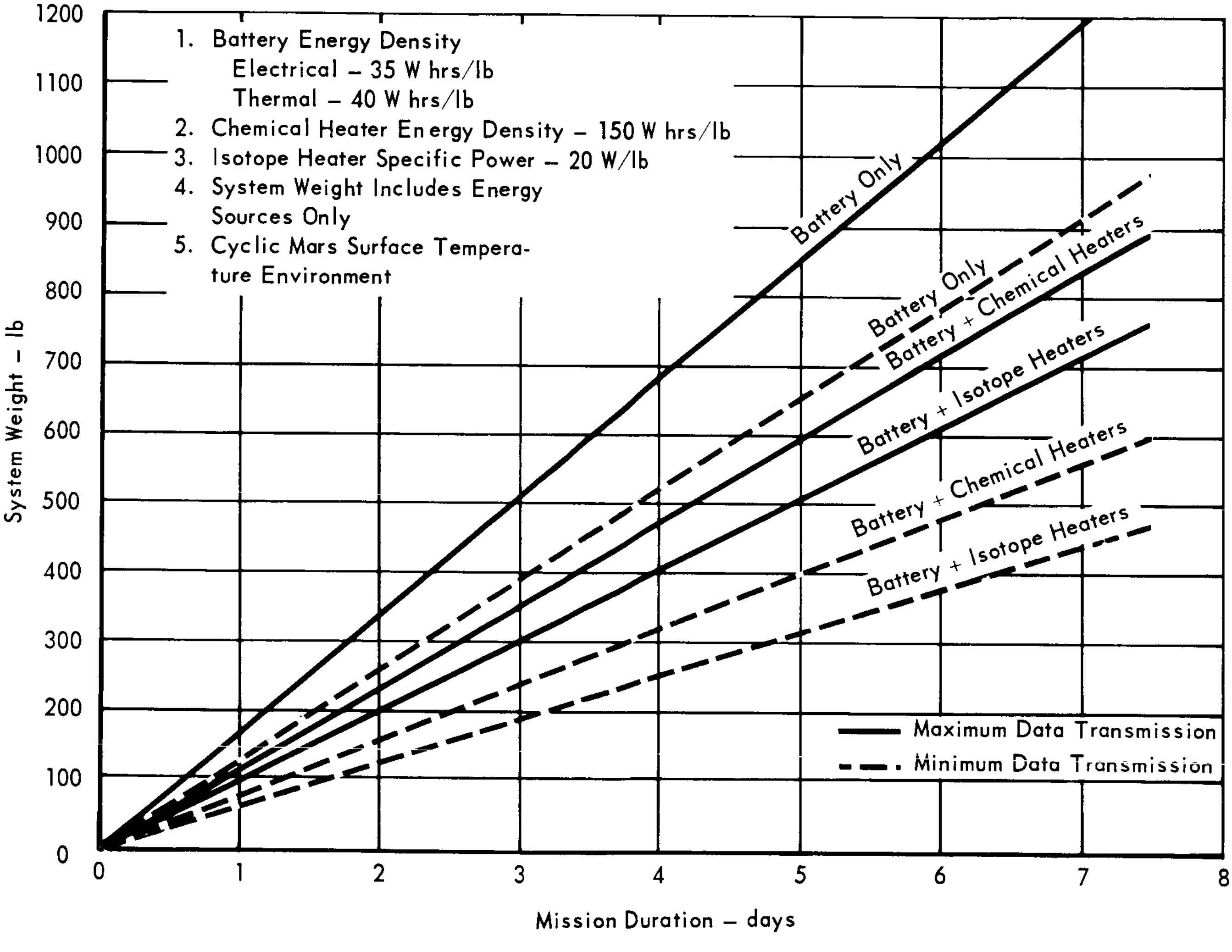


Figure 4.7-9

The following summary data (from Figures 4.7-9 and 4.7-2) compares the mission extension and associated changed science value (relative to the baseline) achievable with a 300 lb weight addition using three battery system approaches and both power profiles. The increased science value is small (and in one case negative) and does not appear to warrant the additional weight and design complexity.

Technique	Mission Duration (Days)	Science Value Change (Relative to Value of Baseline Science After One Diurnal Day)
Batteries Only, Science Program 1	1.8	+ 10%
Batteries Only, Science Program 2	2.3	- 5%
Batteries + Chemical Heaters, Program 1	2.5	+ 10%
Batteries + Chemical Heaters, Program 2	3.8	+ 10%
Batteries + Isotope Heaters, Program 1	3.0	+ 15%
Batteries + Isotope Heaters, Program 2	4.8	+ 20%

Solar Array Subsystems - The solar array analysis was conducted for four possible conditions all assuming a worst case landing on a 34° slope. The four conditions are:

a. Planar array at 20°S latitude - 5.6 $Wh/ft^2/day$

b. Planar array at 0° latitude - 10.3 $Wh/ft^2/day$

c. Windowshade array at 20°S latitude - 6.2 $Wh/ft^2/day$

d. Windowshade array at 0° latitude - 12.0 $Wh/ft^2/day$

Further assumptions are: (1) no shade period during sunlight hours, (2) battery used for night electrical load and isotope heater for thermal load, (3) solar array/battery charge efficiency of 50%, and (4) battery depth of discharge of 25%, for long life.

The battery weight is:

$$W_B = \frac{(\text{night load})\ (\text{period})}{(\text{depth of discharge})\ (\text{specific energy})}$$

$$= \frac{74\ \text{W} \times 12.3\ \text{hrs}}{0.25 \times 35\ \text{Wh/lb}}$$

$$= 104\ \text{lb}$$

The isotope heater weight to produce 235 W is 12 lb.

The solar array area just required to recharge batteries (1850 Wh) is:

Condition	Array Area (ft^2)	Array Weight (lb)
20°S latitude Planar Array	330	165
20°S latitude Windowshade Array	298	149
0° latitude Planar Array	180	90
0° latitude Windowshade Array	154	77

The solar array area and weight required to support a full science program of 2535 Wh, not including the night load, and the minimum transmission power profile (1154 Wh) for the various conditions is:

Condition	Full Science Area(ft^2)	Full Science Weight(lb)	Minimum Science Area(ft^2)	Minimum Science Weight(lb)
20°S, Planar	492	246	206	103
20°S, Windowshade	442	221	186	93
0°, Planar	268	134	112	56
0°, Windowshade	210	120	96	48

The total areas and weights of a battery + isotope + solar array for the four conditions are:

Condition	Full Science Area(ft^2)	Full Science Weight(lb)	Minimum Science Area(ft^2)	Minimum Science Weight(lb)
20°S, Planar	882	243	536	400
20°S, Windowshade	740	502	484	374
0°, Planar	448	356	292	278
0°, Windowshade	394	329	250	257

A solar array system, either planar or windowshade could support a minimum science program assuming a weight limitation of 300 lb, only at 0° latitude or higher northern latitudes with any 34° slope, or as low as 40°S with optimum or 0° slope conditions.

Another approach is to provide batteries to do one complete science mission and provide a solar array to charge batteries incrementally beyond the night-time level so that over a period of time the batteries regain capacity to accomplish another complete mission. This would result in sporadic data acquisition but when compared to the science schedule could have a high mission value.

For this approach, the days between complete science missions as a function of system weight is plotted in Figures 4.7-10 and 4.7-11 for the two extreme power profiles. Again assuming a system weight limitation of 300 lb no data can be obtained at latitudes below 0° with a maximum slope requirement of 34°. At 0° latitude with a windowshade array, the time between complete science missions is about 10 days for the full science mission. At the same conditions the time between data for the minimum science program is slightly over 2 days for the planar array and slightly under 2 days for the windowshade. These relatively short periods between data acquisition result in a high science value output. The hour variations are determined in the basic cycle and these science schedules ascertain the weekly and monthly variation. The only science desire that this approach cannot achieve directly is the diurnal variation effect.

RTG Subsystem - In comparing the various power/thermal source systems, an RTG/Battery subsystem was examined. The required RTG power to support both battery charge and the various power profiles given in Figure 4.7-8 is determined from the following expression:

$$P_r = \frac{P_p \Delta T_2 + P_m \Delta T_1 \eta_c}{\Delta T_1 \eta_c + \Delta T_2}$$

where for a full science program:

P_r = RTG power

P_p = peak power, 250 Watts

P_m = minimum power, 74 Watts

γ_c = battery charge efficiency, 50%

ΔT_1 = battery recharge period, 15.5 hours

ΔT_2 = battery discharge, 9.0 hours

P_r = 169 Watts (e)

RTG weight based on a specific subsystem weight of 1.09 W/lb is 155 lb. The battery weight, considering a 25% depth of discharge, a 35 Wh/lb energy density, and a 720 Wh output, is about 82 lb. Thus, the total system weight is 237 lb for a continuous power output. The subsystem weight for a minimum power profile

DAYS BETWEEN COMPLETE SCIENCE MISSIONS VS. SOLAR ARRAY SYSTEM WEIGHT INCLUDING BATTERIES AND ISOTOPE HEATERS

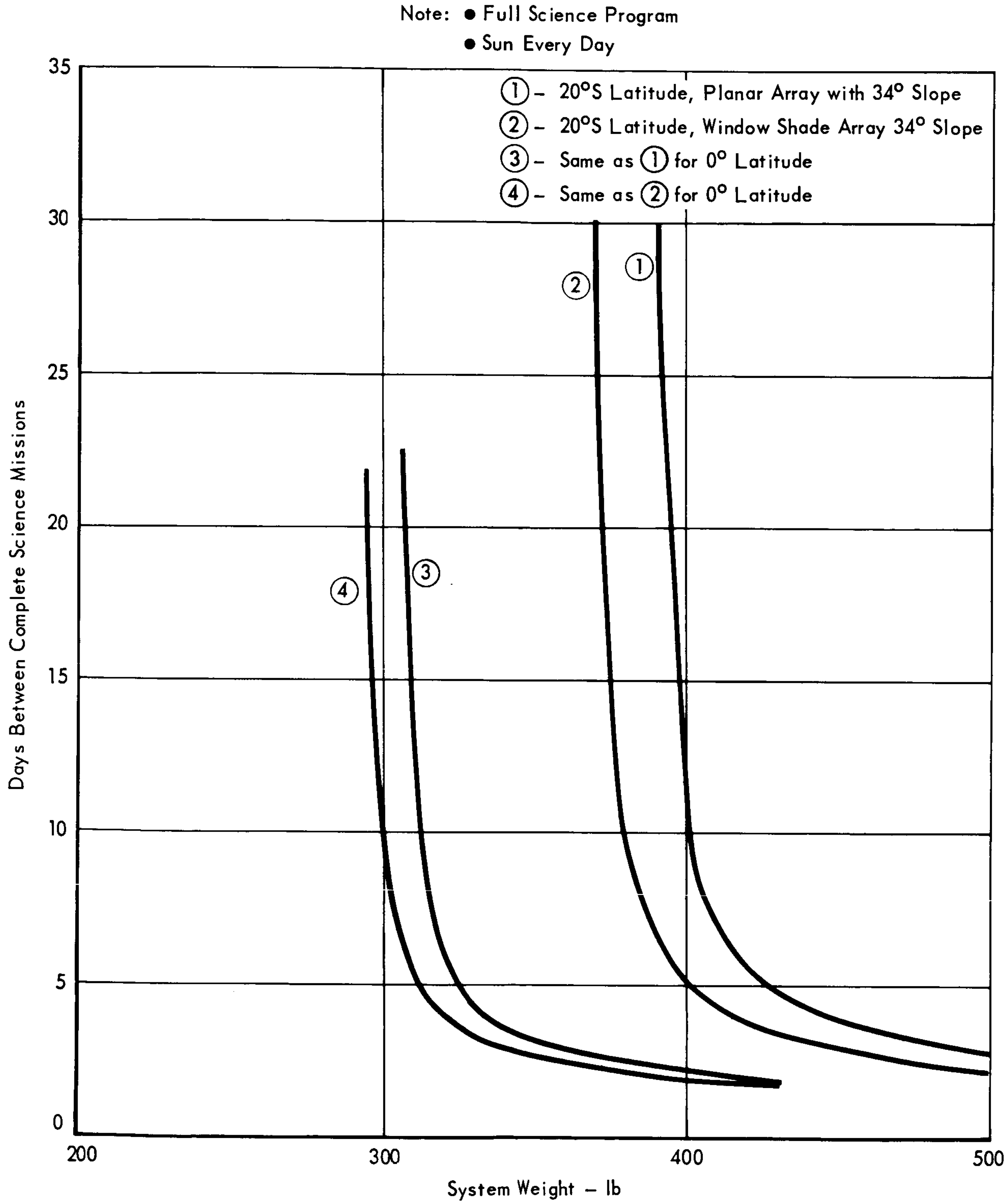


Figure 4.7–10

DAYS BETWEEN COMPLETE SCIENCE MISSIONS VS SOLAR ARRAY SYSTEM WEIGHT INCLUDING BATTERIES AND ISOTOPE HEATERS

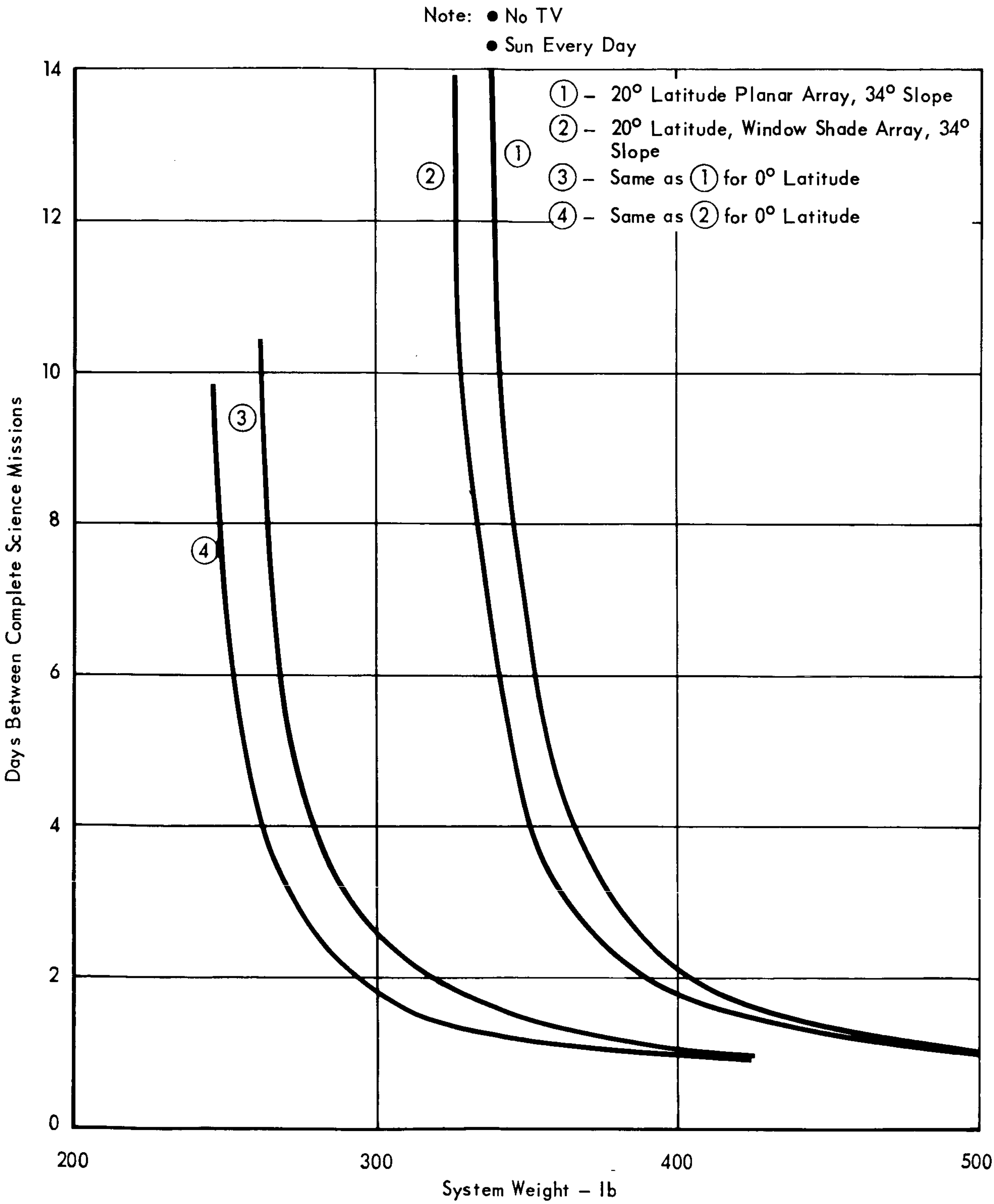


Figure 4.7-11

minimum power profile is about 107 lb (85 lb of RTG and 22 lb of batteries).

These results indicate an RTG/Battery system has a definite application as a power and thermal source for an extended mission program if the inherent problems can be solved.

4.7.6 Conclusions - The various techniques for achieving extended mission are compared in Figure 4.7-12. Superimposing the power/thermal characteristics on the science value plot in Figure 4.7-2 results in the following performance for a 300 lb limited system:

Power and Thermal Techniques	Relative Science Value*
Batteries and Heaters	+ 10%
H_2-O_2 Fuel Cell	+ 50%
Solar Cell/Isotope/Battery	+ 100%
RTG/Battery	+ 100%

*Science value above the level established for the baseline science being performed during one diurnal cycle = 1.0.

The results indicate that the RTG/battery subsystem can extend the mission to 30 days with the least weight penalty. This subsystem is also relatively unaffected by the environment and, therefore, is not limited in landing site as is the solar energy subsystem. Purely on a technical basis the Solar Cell/Battery subsystem would be the choice if the RTG approach is ruled out for system integration reasons, i.e., radiation dosage, and political problems. The second choice would be a solar array, battery, and isotope heater subsystem. However, this subsystem would require more detail study to assess the probability of cloud coverage versus latitude to predict the power output of an array. This approach would be recommended only as a secondary system. A battery subsystem is used as the primary power source for the one diurnal mission. The probability of cloud coverage for more than one day drastically limits the use of solar cells. If fuel cells can be proved to be sterilizable and available for the 1973 mission, an appreciable science value could be obtained by extending the mission to about 12 days. The battery and heater approach for a long duration mission is impractical from the standpoint of weight and science value.

POWER/THERMAL CONTROL SYSTEM WEIGHT
COMPARISON FOR A FULL SCIENCE PROGRAM

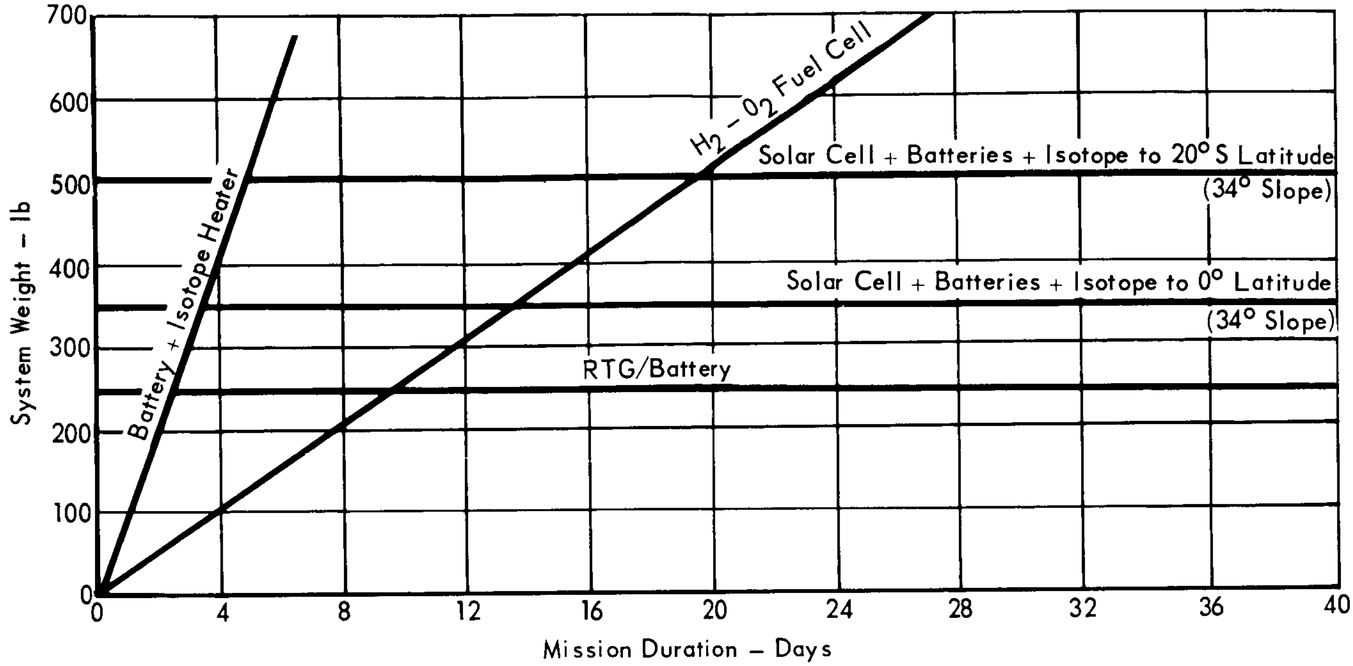


Figure 4.7–12

REFERENCES

1. Opik, E. J., Jr.: Journal of Geophysical Research, Vol. 65, No. 10, p 3057, October 1960.