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SPACECRAFT Phase B, Task D

FINAL REPORT

OCTOBER 1967

Volume 5. Science Payload Integration

Prepared for
GEORGE C. MARSHALL SPACE FLIGHT CENTER
Huntsville, Alabama

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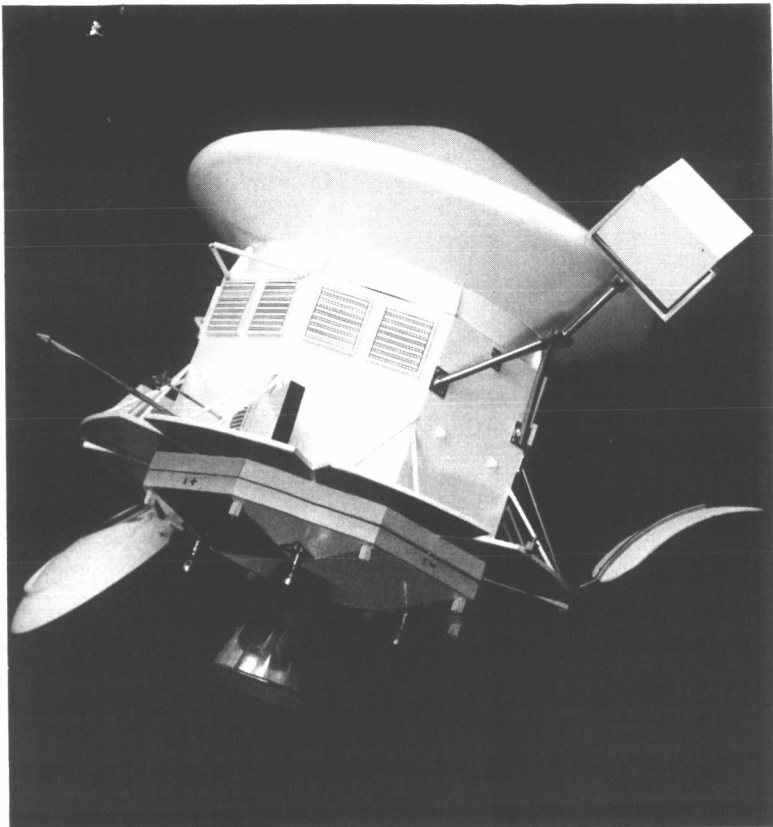
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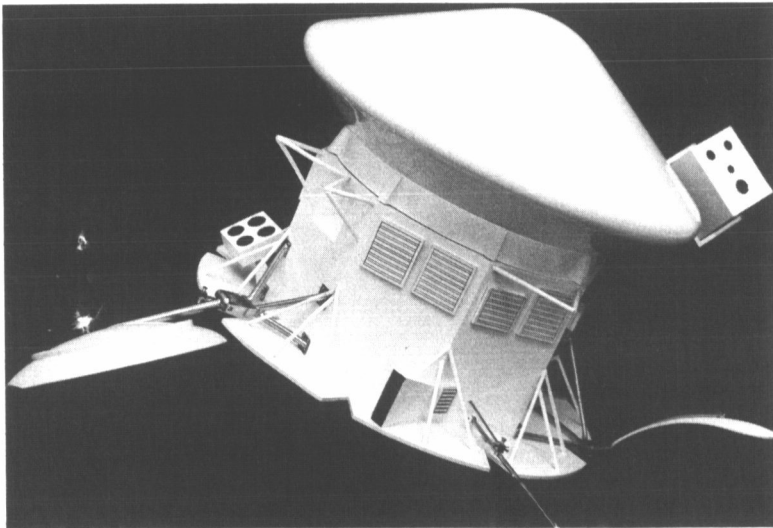
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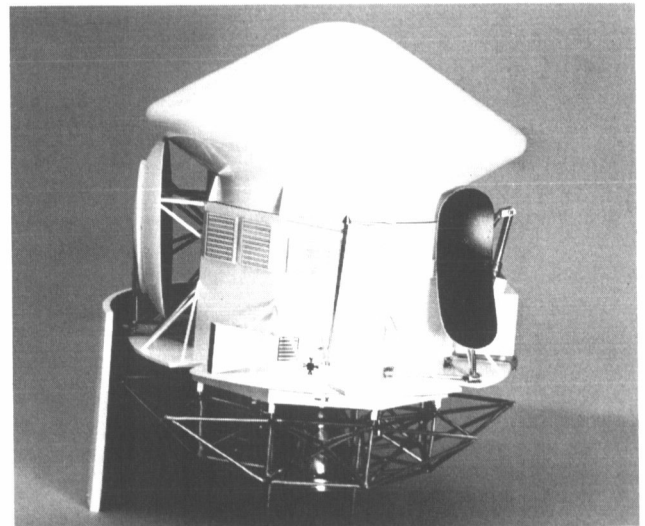


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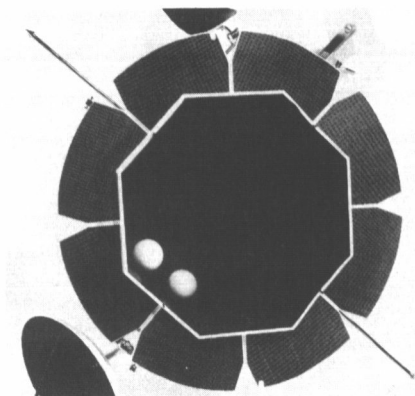
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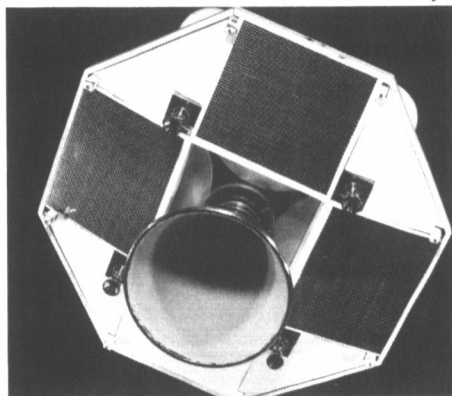
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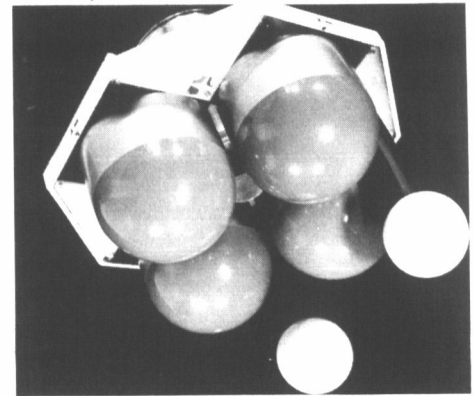
Stowed Configuration with Section of Shroud and Planetary Vehicle Adapter



Propulsion Module, Top View



Propulsion Module, Bottom View



Equipment Module, Bottom View

VOYAGER

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Prepared for
GEORGE C. MARSHALL SPACE FLIGHT CENTER
Huntsville, Alabama

TRW
SYSTEMS GROUP

*Voyager Operations
Space Vehicles Division*

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1. SUMMARY

1.1 INTRODUCTION

The purpose of this volume is threefold. First, it provides a brief summary of science objectives for Mars exploration with emphasis on those objectives which can be satisfied by experiments and equipments supported by the Voyager spacecraft.

Second, it provides a description of groups of scientific experiments and photo-imaging equipment which can be used to satisfy these objectives. Since the specific number and type of experiments to be supported by the spacecraft will be specified by NASA during the early part of the spacecraft design phase, the experiment descriptions in this volume are intended to allow determination of the type of interfaces for typical science payloads.

Third, it provides definition of spacecraft subsystems configured and sized to provide the required power, switching and sequencing, data handling, thermal control, structural support, and pointing, for these typical science payloads.

The main emphasis of this volume will be on the last item, the integration of the typical experiments with the spacecraft.

Although there is no science subsystem as such, integration of the science experiments requires consideration of the following tasks, equipments, and spacecraft subsystem interfaces:

- Science experiments and requirements
- Photo-imaging equipments and requirements
- Spacecraft configuration

Planetary scan platform mounted equipment

Body mounted experiments

Boom mounted experiments

- Mechanical and thermal interfaces
- Science equipment weight and power
- Data handling and sequencing interfaces
- Science equipment integration and test activities (see Appendix F)

1.2 SCIENCE OBJECTIVES

Guidelines for the scientific exploration of Mars were given by the Space Science Board of the National Academy of Science in its December 1965 report* and have been elaborated and applied in subsequent Voyager studies, e. g., Voyager Support Study, Advanced Mission Definition, Final Report, November 1966 prepared by TRW Systems Group for the Jet Propulsion Laboratory. This discussion uses these previous analyses to summarize the scientific objectives which can be realized by a Mars orbiter.

The Space Science Board believes that the biological exploration of Mars should receive the highest priority, but that "specifically biological questions should be asked in an ordered sequence of exploration whose purpose is to understand the overall evolution of the planet's crust and atmosphere." In addition to this broad scientific coverage, the exploration program must realize environmental data for use in the development, design, and operation of later advanced missions; experiments contributing to this end should receive the highest priority.

1.2.1 The Atmosphere and its Interaction with the Surface

Questions that can be answered, in part, by experiments operating from an orbiting spacecraft are:

a) Meteorology

- What are the spatial and temporal variations in atmospheric pressure and temperature near the surface?

* Space Research, "Directions for the Future, Part One, Planetary and Lunar Exploration," Report of a Study by the Space Science Board, National Academy of Science, December 1965.



- What are the atmospheric composition and its state parameters as a function of altitude? What are the absorption characteristics of the atmosphere over the solar electromagnetic spectrum?
- Are there surface winds and if so, what are their characteristics?
- Are there aerosols in the atmosphere? If so, what is their vertical distribution?
- What are the white clouds, the yellow clouds or haze, and the blue haze?
- What is the nature of the seasonal changes in surface features?
- Do electrical storms occur?

b) Ionosphere

- Over what altitude region does the ionosphere extend? (Electron concentration as a function of altitude.)
- What are the types and densities of the positive ions as a function of altitude?
- What are the temporal variations in ionospheric properties, and especially how do they depend on solar activity?
- What are the ion and electron temperatures as a function of altitude?
- Do instabilities occur?
- What are the characteristics of the night glow?
- Do auroral phenomena occur, and if so, what is their nature?

c) Planetary and Atmospheric Evolution?

- What is the nature of the interaction of the solar wind with the planet? Does the solar wind contribute to the atmosphere?
- What are the $\text{Ne}^{20}/\text{Ne}^{22}$ and A^{36}/A^{40} isotope ratios? Are other ratios such as $\text{C}^{12}/\text{C}^{14}$ modified from the terrestrial values as a result of cosmic ray bombardment?

The composition of the Martian atmosphere has been investigated spectroscopically from the earth; carbon dioxide has been observed and water vapor appears as a minute trace constituent. The low atmospheric density found by Mariner IV indicates that CO_2 is probably the dominant constituent with perhaps some small concentrations of the infrared-inactive elements argon and nitrogen, whose presence is inferred by analogy with earth. An orbiting spacecraft should be equipped with sensitive ultraviolet and infrared spectrometers to observe characteristic emission and absorption from minor constituents of the Martian atmosphere that cannot be detected when viewed through the earth's atmosphere. The altitude dependence of density and temperature can be determined spectroscopically by appropriate modes of observation. The ultraviolet spectrometer acting as a multi-wavelength photometer can measure Rayleigh scattering in the lower atmosphere and the ultraviolet reflectivity of the planet's surface. The presence of aerosols may be studied by photometric and polarimetric measurements of scattered light.

Additionally, the infrared is useful for temperature measurement, and an infrared radiometer should be carried to exploit this type of measurement, as well as to provide backup over wide energy bands for the spectrometry measurements. The chief functions of the radiometer are to obtain overall atmospheric temperature measurements and to carry out detailed temperature mapping of the planet's surface, which can be correlated with the photo-imaging results.

An orbiting spacecraft should also contain an atmospheric mass spectrometer as a backup to similar instruments on entry/lander vehicles, in order to obtain direct measurement of atmospheric species found at the orbit altitudes. Obviously as low a periapsis as possible is desirable for this purpose, and a minimum altitude of 400-500 km, rather than the present nominal value of 1000 km, would be of great importance. If the periapsis is sufficiently low, the mass spectrometer may give direct measurement of one or more of the significant isotopic abundance ratios.

There can be little doubt that photo-ionization by sunlight establishes a Martian ionosphere. The Mariner IV radio occultation measurements indicated a maximum density of approximately 10^5 electrons/cm³ on the sun-lit region. A large diurnal effect was also indicated. The ionosphere can be characterized by identification of ion species, number, densities, and temperature. Measurement of the charge density can be made from orbit by any of several types of radio wave probes. Identification of the positive ion species in the ionosphere and direct measurement of their densities can be done with the atmospheric mass spectrometer and, of course optical spectroscopy also leads to the determination of some species. To detect auroras and measure the characteristics of the night glow an ultraviolet spectrometer may be used to look at the night side of the planet, especially during an eclipse.

1.2.2 Surface, Interior and Solar System Interactions

In these areas the chief scientific questions to which an orbiting spacecraft can contribute information are:

- What are the equatorial and polar radii of Mars, what are the higher order terms in its gravitational potential, and what is its mass?
- What are the details of the visual surface features?
- What are the abundances of radioactive elements, both natural and induced?
- What is the surface composition and what is the abundance of bound water?
- Can a magnetic field be detected, and if so, what are its properties – configuration, magnitude, fluctuations and their correlation with solar activity?
- Are there detectable trapped particle belts?
- What is the nature of the interaction of the solar wind with the planet?

- What is the magnitude and energy spectrum of the charged particle environment near and at the planet's surface, due to cosmic rays, the solar wind, and planetary interactions?
- What is the meteoroid flux distribution near Mars and what are the mass and velocity spectra of the meteoroids?

More accurate information about the mass of Mars and higher order terms in its gravitational potential can be obtained by accurate tracking of the orbiter spacecraft. The photo-imaging equipment, in turn, can provide data leading to more accurate determinations of the equatorial and polar radii of Mars.

Mapping of the surface of Mars by remote imaging devices is an important first step in a comprehensive exploration program, to include the identification and classification of major physiographic features. Color and polarization reconnaissance are needed along with infrared surveys. Those areas having distinctive features such as color, albedo, thermal emission, erosional effects, or small scale geological features should have higher resolution for detailed study.

About three-quarters of the surface of Mars consists of the bright areas, orange colored, called deserts, now thought to be made up largely of hydrated iron oxides. Optical polarization and diffusion measurements suggest a powdery substance consisting of fine, opaque grains; the substance most nearly resembling this material is the hydrated iron oxide called limonite. The steep slope of the Martian albedo in the visible and near infrared wavelengths is a significant composition index; it is characteristic almost uniquely of ferric oxides. If the surface is indeed largely a mixture of hydrated iron oxides, the high oxidation state and water content point to environments with relatively high humidity, high temperatures and free oxygen during their formation. Moreover, given limonite, the Martian surface may be composed of 5 to 10 percent water by mass, bound to the ferric oxide by weak crystallization bonds.



The polar caps of Mars are thought to be composed of either frozen water or carbon dioxide, with earth-based polarimetry suggesting that water is a constituent. Better polarization measurements from an orbiter will extend present knowledge of both the polar caps and the deserts. These measurements should be taken from all angles in reflected sunlight. In addition, a continuing high gain photographic and radiometric survey of the morphology of the polar caps should be made as they advance and retreat.

It would seem desirable that an orbiter spacecraft carry a magnetometer of maximum sensitivity in order to determine once and for all if a detectable magnetic field exists, although this must be considered a backup to a lander experiment. However, in view of the Mariner IV finding that any magnetic field must be at most 1/1000th that of the earth it seems very unlikely that significant belts of trapped particles can exist. Yet the solar wind must in any case interact in some fashion with the planet, even if it is directly incident on the surface. Thus, on the one hand the interplanetary magnetic field may extend with little or no change nearly to the surface itself, or on the other hand perturbations and turbulence due to the interaction of the solar wind with the planet may be found in the near Mars environment. Both questions relate immediately to the nature of the charged particle flux at the planet's surface and hence, to the conditions under which indigenous life may exist as well as the conditions which must be sustained in any manned exploration of the planet.

At the present time there exists little direct data concerning the meteor environment near Mars. Several arguments point towards relatively high meteor impact rates in the vicinity of the Martian orbit. The proximity of the asteroid belt to the Martian orbit is one; the disturbing effect of Jupiter on the asteroids motion is another. The argument of increased meteor activity near Mars has been lent particular weight by the results from Mariner IV, which indicate increased count

rates of micrometeoroids as it approached the Mars orbit. Because of its significance for future Mars missions, it is important to investigate this question thoroughly by means of the orbiters. Both count rates to determine the overall meteor flux, and mass and velocity spectra should be obtained. Although several types of impact detectors have been proposed and flown on various space missions, confidence in their performance is still lacking. Furthermore, the accessible mass range of perhaps 10^{-5} gm to 10^{-11} gm constitutes the cosmic dust responsible for the zodiacal light; although of considerable astrophysical interest, these masses are less than those which provide greatest potential danger to man and equipment. It would be desirable to fly a meteor detector sensitive through a range of larger masses as well as a more conventional cosmic dust detector. Recently, studies have led to the possibility of meteor flash detectors, using the light emitted upon a meteor's encounter with the atmosphere as the basis not only for detection, but also for determination of mass and velocity. According to theoretical analysis the flash detector is useful for appreciably larger masses than are present impact detectors. The useful mass range depends on the velocities, but for typical expected velocities the masses are of the order of 10^{-1} gm. This would provide data concerning a regime of meteoric particles hitherto unexplored, between present laboratory data and space data, and presenting the greatest hazard to man and equipment in future space missions. Such an instrument is further attractive in that it can continuously measure the night glow while trained on the dark side of the planet and during eclipses.

1.3 SCIENCE PAYLOAD DEFINITION

1.3.1 Hypothetical Science Payload

Detailed integration of scientific experiments with the spacecraft can be performed only after the particular instrument interfaces and their support requirements are defined. However, in order to define the spacecraft supporting subsystems, TRW has elected to utilize,



as an initial science payload, a "Hypothetical Science Payload" suggested in part by MSFC. This hypothetical payload consists of the following equipment:

- Photo-imaging equipment — all television system
 - a) One low-resolution color television camera (1000 m resolution)
 - b) Three medium-resolution TV cameras (100 m resolution)
 - c) One high-resolution television camera (10 m resolution)
- "Core" instruments for measurements of Mars surface and atmospheric constituents
 - a) High-resolution infrared spectrometer
 - b) Broadband infrared spectrometer
 - c) Infrared radiometer
 - d) Ultraviolet spectrometer

Additionally, measurements of the Martian atmospheric properties (ionosphere) can be made using the shifts in signals from the spacecraft's communication system during earth occultation periods.

A change was made by TRW in the MSFC list of equipment in the area of photo-imaging equipment. The MSFC list contained one high-resolution TV camera and two medium-resolution cameras. As explained in detail in Volume 11, the analysis of photo-imaging objectives and ground coverage desired in conjunction with data handling system constraints showed that a third medium-resolution TV camera should be added to allow continuous ground track coverage at medium-resolution. The low-resolution color TV camera was added to allow overall mapping of the complete planet in color early in the mission.

The hypothetical payload is defined in more detail in Section 2 below. The hypothetical photo-imaging equipments, as well as other photo-imaging alternates, are discussed in detail in Volume 11.

1. 3. 2 Additional Experiments

In addition to the "core" group of science experiments (listed in 1. 3. 1), MSFC suggestions also contained a group of "additional experiments" under consideration for the Voyager mission. These plus other experiments that would contribute to Voyager objectives are listed below:

- a) Atmospheric mass spectrometer
- b) Polarimeter
- c) Solar occultation
- d) Gamma ray spectrometer
- e) Neutron albedo sensor
- f) Cosmic ray detector
- g) Magnetometer
- h) Micrometeor impact detector
- i) Meteor flash detector
- j) Solar plasma detector
- k) Solar x-ray detector
- l) Celestial x-ray detector

The detailed descriptions and objectives of these experiments is given in Section 5. An analysis was made of the requirements imposed on the spacecraft by the addition of these experiments, and the spacecraft subsystems have been designed to accommodate the addition of any or all of these experiments as desired.

1. 3. 3 Film/Television Photo-imaging System

As a result of the present Task D Voyager study on photo-imaging systems, TRW is recommending a combination of a high- and medium-resolution Lunar-orbiter-type film system and a low-resolution color TV photo-imaging system. At the nominal periapsis altitude of 1000 km the resolutions proposed for the film system are 100 meters and 10 meters and the resolution of the color TV system is 1000 meters. Detailed descriptions of the combination film/TV systems are given in Volume 11 and a summary description is provided in Section 4. 2.



Since the size and shape of the film/TV system is considerably different from the all-TV "hypothetical" system, an additional configuration layout of this photo-imaging system has been made to investigate the feasibility of integration into the Voyager spacecraft. This layout is shown in Section 4.3 below. In addition to the physical differences, the film system has a different data interface with the spacecraft since it contains its own data storage and readout systems. The detailed integration of this film system with the spacecraft is also discussed in Section 4.3.

The main advantages of the recommended film system over the all-TV system are, 1) greater data storage capabilities of the film versus tape recorders for the TV and, 2) much wider field-of-view for the high-resolution film lens (because of the film format size) versus a narrow field-of-view for the high-resolution TV lens, requiring less stringent pointing for the film lens during photo-imaging operations. A disadvantage of the high-resolution film system is that image motion compensation (IMC) is required to prevent "smear" of the picture.

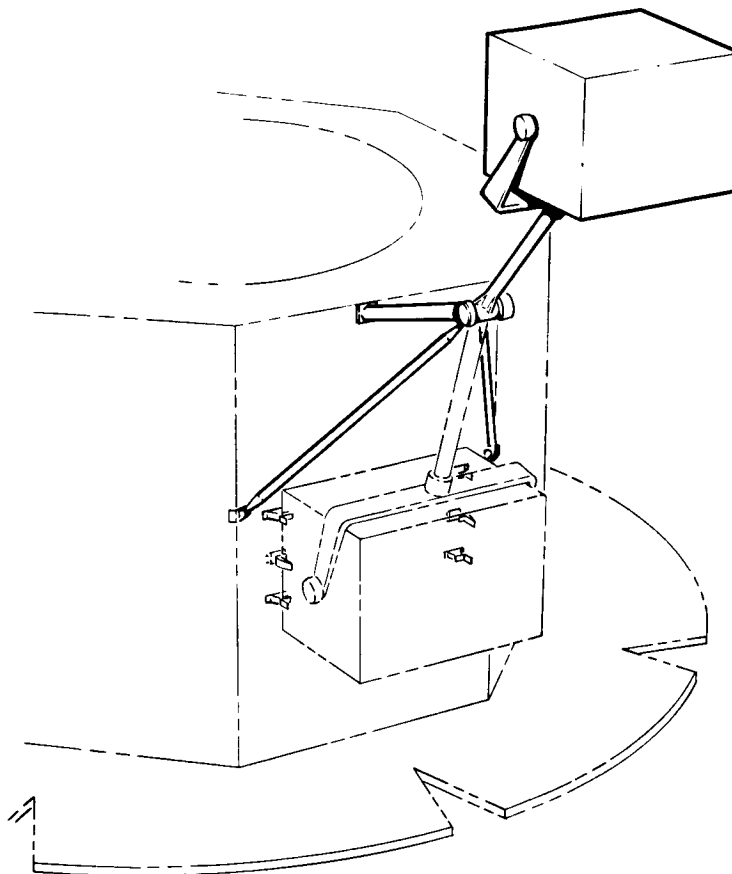
1.4 SCIENCE SYSTEM INTEGRATION DEFINITION

1.4.1 Mechanical and Thermal

The mechanical integration of the hypothetical science payload on the spacecraft consists of determining specific requirements for each experiment such as view angles, pointing directions and accuracies, and allowable tolerances on mounting alignment; and then satisfying the specific requirements in the design of the spacecraft configuration. The core group of hypothetical experiments listed above all require pointing at the planet; a gimballed planetary scan platform (PSP) is provided on the spacecraft for mounting these experiments. (See Figure 1-1.) The planetary scan platform is hinged from the side of the spacecraft through a series of gimbal drives to allow movement of the platform face (boresight) by the drives to allow continuous viewing of the planet throughout the orbit. Three drives are provided for the planetary scan

PRELIMINARY SPECIFICATION

Hypothetical Science Payload



STRUCTURE

Weight: 183 lb
 Strength: Designed to withstand Mars orbit insertion G loads with PSP deployed
 Front face area: 45 in. x 37 in.
 Volume: 34 cu ft
 Deployment: electric hinge axis drive, explosive release from stowed position

THERMAL CONTROL

Insulation: Foam within meteoroid protection honeycomb panels
 Thermostats and heaters: For experiments and drive mechanisms
 Coatings: Black Cat-a-lac paint except white IIT RI Z-93 on cooling radiators
 Radiation: 3/16 aluminum plates for IR Spectrometers
 Cryogenic refrigerator: For broadband IR Spectrometer

POINTING AND CONTROL

Pointing accuracy: 0.3° (Mars sensors - closed loop mode)
 Control rates: ± 10 mr/sec ± 0.17 mr/sec
 Reliability: 0.955

EXPERIMENTS

High resolution IR Spectrometer
 Broadband IR Spectrometer
 IR Radiometer
 UV Spectrometer
 Photo-imaging System

INTERFACES

Spacecraft structure: 3 support struts
 Data handling: Through remote multiplexers in PSP
 Computer and sequencer: Through remote decoders in PSP
 Power: Through remote switching assembly in Planetary Scan Platform
 34.6 watts (average) for noneclipse orbit
 55.43 watts for eclipse orbit

Figure 1-1



platform. One drive is mounted at the main shaft/spacecraft hinge axis. This drive is used for deploying the planetary scan platform away from the spacecraft (stowed position) to the desired viewing positions. The second drive rotates the scan platform about its main shaft and the third drive rotates the actual platform box within a yoke attached to the main shaft. Power and signals are provided through cable wrap-ups around the gimbal axes.

The gimbal system for the planetary scan platform allows three degrees of freedom viewing of the planet for those angles not restricted by the gimbal locks or by the spacecraft or capsule bodies. An additional mode of viewing requiring a combination spacecraft roll maneuver and planetary scan platform shaft positioning is described in Section 3.5 below.

The planetary scan platform structural supports, hinge drive, shaft and yoke are designed to allow the platform to be left in a deployed position during orbit insertion maneuvers. The box construction also provides meteoroid protection for the experiments.

The design of the planetary scan platform is such as to provide the required thermal control for each experiment. For example, the broadband and high-resolution spectrometers require special cooling for the detectors, and radiating plates are installed in the surface of the planetary scan platform box to provide cooling for these devices. Other instrument temperatures are controlled by the use of insulation, coatings, and thermostatically controlled heaters.

1.4.2 Science Data Handling and Sequencing

Figure 1-2 is a science interface block diagram showing the electrical interfaces with other spacecraft systems. In the hypothetical science payload all of the experiments are mounted in the planetary scan platform and the electrical integration assembly shown on the block

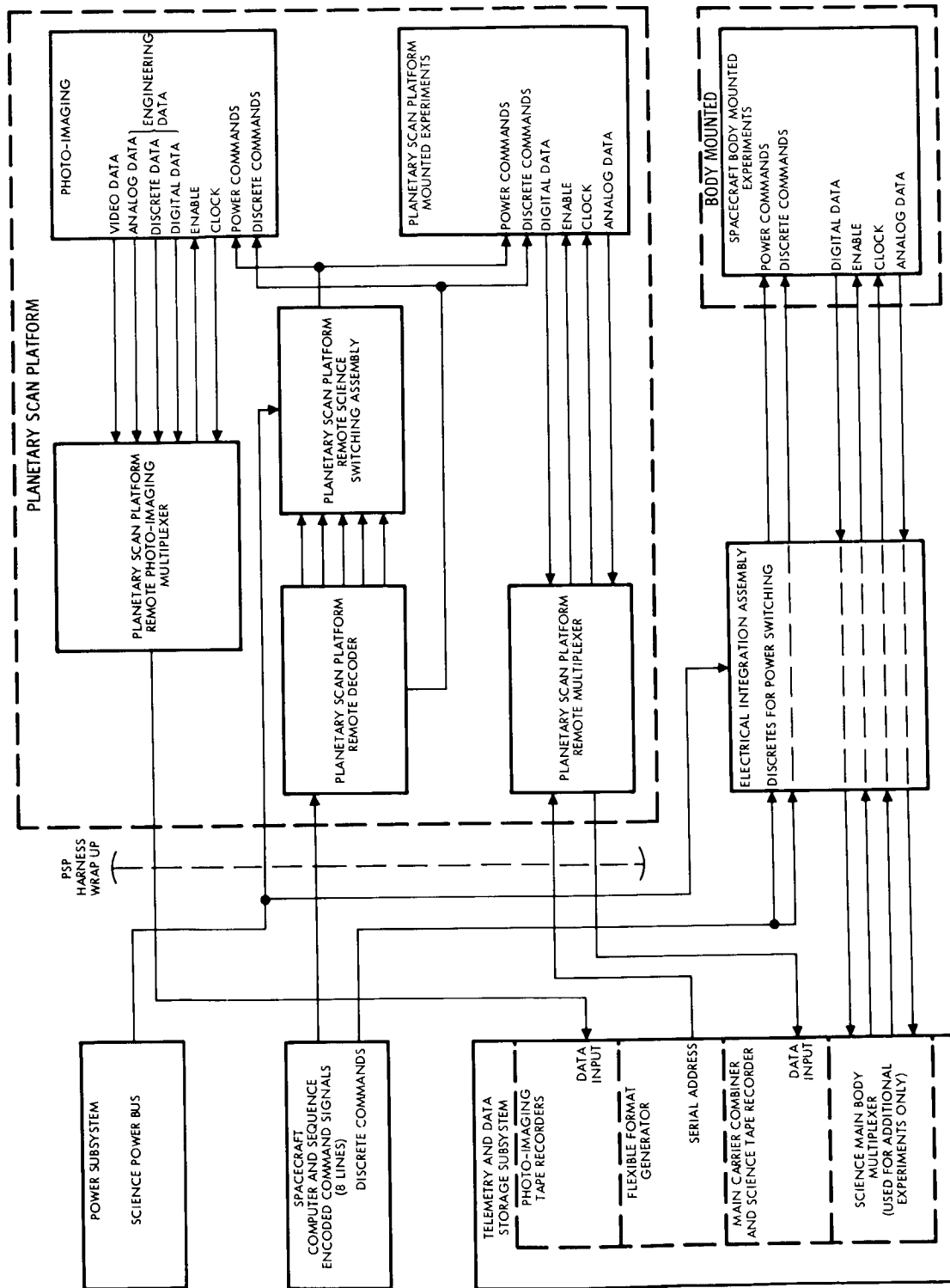


Figure 1-2
SCIENCE EQUIPMENT INTERFACE BLOCK DIAGRAM shows how installation of remote command decoders, switching assemblies and multiplexers on the Planetary Scan Platform reduces wiring between instruments and spacecraft; body-mounted experiments interface with the spacecraft through a science electrical integration assembly.



diagram for body mounted experiments is not required. Remote multiplexer, data handling, command decoders and science switching assemblies provided on the planetary scan platform allow an order of magnitude reduction in the number of wires required in the planetary scan platform cable harness. Because of the significantly higher data rates for the photo-imaging systems, a separate planetary scan platform remote multiplexer is provided to handle photo-imaging data.

The data handling system proposed allows maximum flexibility in the choice of experiments and data format and allows changes to experiments subsequent to initial design with only minor changes to the spacecraft system interfaces. Individual experiment data requirements can be satisfied by selectively assigning experiment input data channels to the data handling subsystem and by the capability to change the telemetry format by reprogramming a "flexible format telemetry generator." The total data to be handled for the hypothetical science payload is 1.6×10^7 bits per orbit for the core experiments and 29.8×10^8 bits per orbit for the photo-imaging system. The total average power required for the hypothetical science payload is 55.4 watts for an eclipse orbit and 34.6 watts for a noneclipse orbit.

The spacecraft also provides the following command and sequencing functions for the experiments:

- Signals to switch experiment power
- Signals to switch power and execute signals for science pyrotechnic devices (if required)
- Signals to initiate calibration and data taking sequences

All science commands will be sequenced by the computing and sequencing subsystem and individual stored program sequences and commands can be changed by ground command if required.

As noted above, there are no body mounted experiments in the hypothetical payload, but an electrical integration assembly is shown on Figure 1-2 to allow interfacing the spacecraft subsystems in a manner similar to that used for the planetary scan platform experiments. The same data handling interfaces would then be presented to the body mounted experiments that are presented for the hypothetical science payload.



2. HYPOTHETICAL SCIENCE PAYLOAD DEFINITION

2.1 CORE EXPERIMENTS

The list of five planetary surface and atmosphere experiments in the hypothetical science payload has been provided by Marshall Space Flight Center, together with general schematic diagrams and some information about instrument parameters and requirements. It is the contractor's function to define these experiments, not in complete detail, for the experimenters will introduce their own designs and modifications as development proceeds, but only in sufficient detail to permit design of the other spacecraft systems, to demonstrate capability and flexibility of the overall spacecraft design to accommodate such a payload, including such modifications as may reasonably be expected, and to identify significant problem areas which may require further analysis.

Thus, no comparative evaluation of the experiments, their effectiveness in achieving the stated objectives, or their relative priority within the mission has been given. Rather, in this section each experiment is discussed in the following way; first, the scientific objectives are stated as a basis for understanding the general design of the instrument and its necessary or desirable operating modes. Then the instrument is described in rather general terms, but with its critical performance parameters indicated and, where desirable, discussed. It is here that problem areas, if any, are noted. In general, the instrument electronics have been developed in considerable detail as a basis for realistic assessment of the power, data handling and engineering (housekeeping) requirements, and these are discussed in order to present significant aspects.

Following the discussion of each experiment a schematic block diagram and an outline-summary of the operating characteristics and the interface data for the instrument are given. These sheets are in the form of preliminary instrument specifications, in as much detail as is required to satisfy the need by other subsystem designers for experiment interface and integration data.

2. 1. 1 High Resolution Infrared Spectrometer

Purpose. The scientific objectives of the high resolution infrared spectrometer are to determine, as a function of location on the planet:

- a) Vertical variation of temperature in the atmosphere
- b) Abundance of water vapor
- c) The reflectivity of the planet's surface in the 1.5 to 2.0 and the 3.0 to 4.0 micron ranges
- d) The presence of high clouds
- e) The surface and/or cloud-top temperatures.

Instrument. High resolution infrared spectrometers are generally of an interferometer type or a diffraction grating type. An interferometer instrument can combine a wide spectral range (for example, to 20μ) with high sensitivity; a disadvantage is that extensive data processing is required to derive spectral information from the pattern output.

The spectrometer described here is a fixed grating instrument with folded (Ebert) optics to save space and with a large number of fixed-position detectors. The instrument is thus designed to make measurements at definite selected wavelengths and sacrifices flexibility and survey potentiality to simplicity, reliability and a high certainty of achieving success in the chosen objectives. The spectral range and the type of detectors depend on the experimenters' choice of specific objectives. A nominal range of 1.2 to 5.0μ has been stated here and maximum sensitivity can then be obtained with PbS detectors in the 1.2 to 3.5μ range and PbSe detectors in the 3.0 to 5.0μ range. Both types of detectors should be cooled to approximately -60°C to -80°C when operating. In the case of PbS detectors maximum sensitivity is obtained by cooling to -80°C , but the sensitivity in the neighborhood of $+20^{\circ}\text{C}$, while an order of magnitude less, is still greater in the 1.2 to 3.0μ range than cooled PbSe in the 3.0 to 5.0μ range. Hence, it may be deemed unnecessary to cool the PbS detectors. One detector is shown as representative of each type in the schematic block diagram (Figure 2-1).

The instrument should view the planet along the vertical line-of-sight toward both the day and night sides including terminator crossings



PRELIMINARY SPECIFICATION

High Resolution Infrared Spectrometer

Performance Characteristics

Spectral Range

Nominal - $1.2\mu \rightarrow 5.0\mu$ (at discretion of experimenter)

Spectral Resolution

Determined by grating and detector geometry.

Dynamic Range

10^6 at signal noise > 10 - for each detector

Accuracy

1 percent for each detector output

Field of View

1.0×4.0 deg - for telescope

Aperature. π steradians aperture for radiative cooling of PbSe detectors and (optionally) Pb Se detectors

Pointing - Positioned to view along vertical line-of-sight, referenced to visual imaging equipment

Alignment - Shall be known to ± 0.5 deg

Interfaces

ELECTRICAL

Commands

Power - ON-OFF

Bias level

Calibrate

Telemetry

Data

1 - 8-Bit word

Diagnostics

PbSe Temperature monitor

AGC status monitor

Internal multiplexer status

Electronics package temperature

Power supply voltage monitor

Bias level monitor

Calibration monitor

Power

20 watts

MECHANICAL

Weight

30 lb

Dimensions

$10.00 \times 12.00 \times 20.00$

Volume

2400 In.^3

THERMAL

Operating Temperatures

PbSe detectors -60°C

Spectrometer package

$-60^\circ\text{C} \rightarrow +10^\circ\text{C}$

Electronics package

$-30^\circ\text{C} \rightarrow +50^\circ\text{C}$

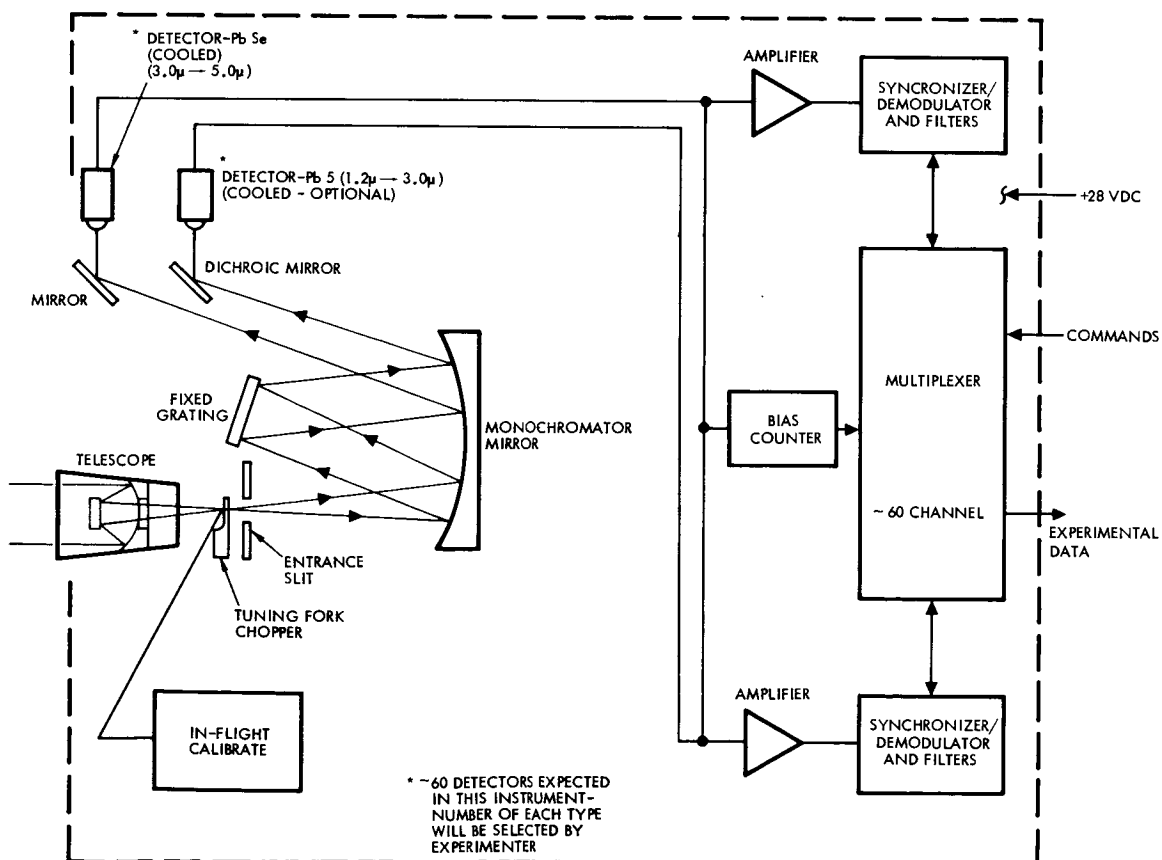


Figure 2-1

whenever possible. It is desirable to look at the entire disc of the planet, correlated with the visual imaging equipment, and to 10 degrees beyond the edge of the disc.

2. 1. 2 Broadband Infrared Spectrometer (See Figure 2-2)

Purpose. The scientific objectives of the experiment are:

- a) Detect the presence of an oxidizing or a reducing atmosphere (positive or negative ions)
- b) Detect the presence of polyatomic molecules that suggest chemical processes, affect the ambient surface temperature, and limit the ultraviolet flux at the surface
- c) Detect compositional variations of atmospheric constituents relative to geographical locale

Instrument. The broadband infrared spectrometer is a two-channel instrument consisting essentially of a telescope, an optical wedge beam-splitter, and for each optical path, a rotating interference filter, a rotating calibration disc, a tuning-fork beam chopper and a semiconductor detector. One of the most important features of the instrument is that it covers the entire desirable range of 1.5 to 15μ with a resolution which is approximately 1 percent of the instantaneous central wavelength, and as a result the spectrometer is especially suited for detecting many weak lines and bands due to minor constituents of the Martian atmosphere. The 8- 12μ range is an atmospheric window (relative absence of absorption bands) which may be convenient for observation of the planet's surface.

For the lower wavelength range, from 1.5 to 6.0μ , a PbSe detector is used. To retain sensitivity above 4μ and to have increased sensitivity at all wavelengths, the detector should be cooled to -80°C ; for maximum sensitivity it should be cooled to approximately -143°C . For the higher wavelength range from 4 or 5 to 15μ a HgGe detector is the most desirable type from the standpoint of sensitivity and wavelength coverage. Unfortunately, it must be cooled to at least -228°C and this is a severe problem which requires thorough analysis.

Operation of the spectrometer with the desired dynamic range and signal to noise ratio over its full wavelength band is also a difficult



PRELIMINARY SPECIFICATION

Broadband Infrared Spectrometer

Performance Characteristics

Spectral Range
Channel 1 - 4.0μ — 15.0μ
Channel 2 - 1.5μ — 6.0μ

Spectral Resolution
1.0 percent of instantaneous
Central λ from 1.5μ to 15.0μ

Spectral Scan
Complete spectral scan every 10 second interval

Dynamic Range
 10^4 at signal noise >10 over range from 1.5μ to 15.0μ

Accuracy - 1.0 percent for output data from each channel

Field-of-View
Offset optical axis from visual imaging equipment reference axis as follows:
 $+1.0 \pm 0.3$ deg in clock
 -4.0 ± 0.3 deg in cone

Alignment
Pointing direction specified under field-of-view shall be known to within ± 0.1 deg.

Interfaces

ELECTRICAL

Commands

Power - ON-OFF
Calibrate
Cryogenic cooler - ON

Telemetry

Data

Multiplexed - 7 - 8-Bit words
Total BPS - 1400

Diagnostics

Channel 1 detector temperature
Channel 1 chopper temperature
Channel 2 detector temperature
Channel 2 chopper temperature
Telescope and monochromator temperature
Electronics package temperature
Internal multiplexer status

Power

20 watts

MECHANICAL

Weight

20 lb

THERMAL

Operating Temperatures

Electronics package - $40.0^\circ\text{C} \rightarrow +40.0^\circ\text{C}$
Telescope and monochromator - $38.0^\circ \pm 5.0^\circ\text{C}$
Channel 1 detector (HgGe) - $228.0^\circ\text{C} (-246) \pm 5^\circ\text{C}$
Channel 1 chopper - $143.0^\circ \pm 5.0^\circ\text{C}$
Channel 2 detector (PbSe) - $143.0^\circ\text{C} \pm 5.0^\circ\text{C}$
Channel 2 chopper - $38.0^\circ\text{C} \pm 5.0^\circ\text{C}$

Dimension

$14.00 \times 12.00 \times 11.00$

Volume

1848 in.^3

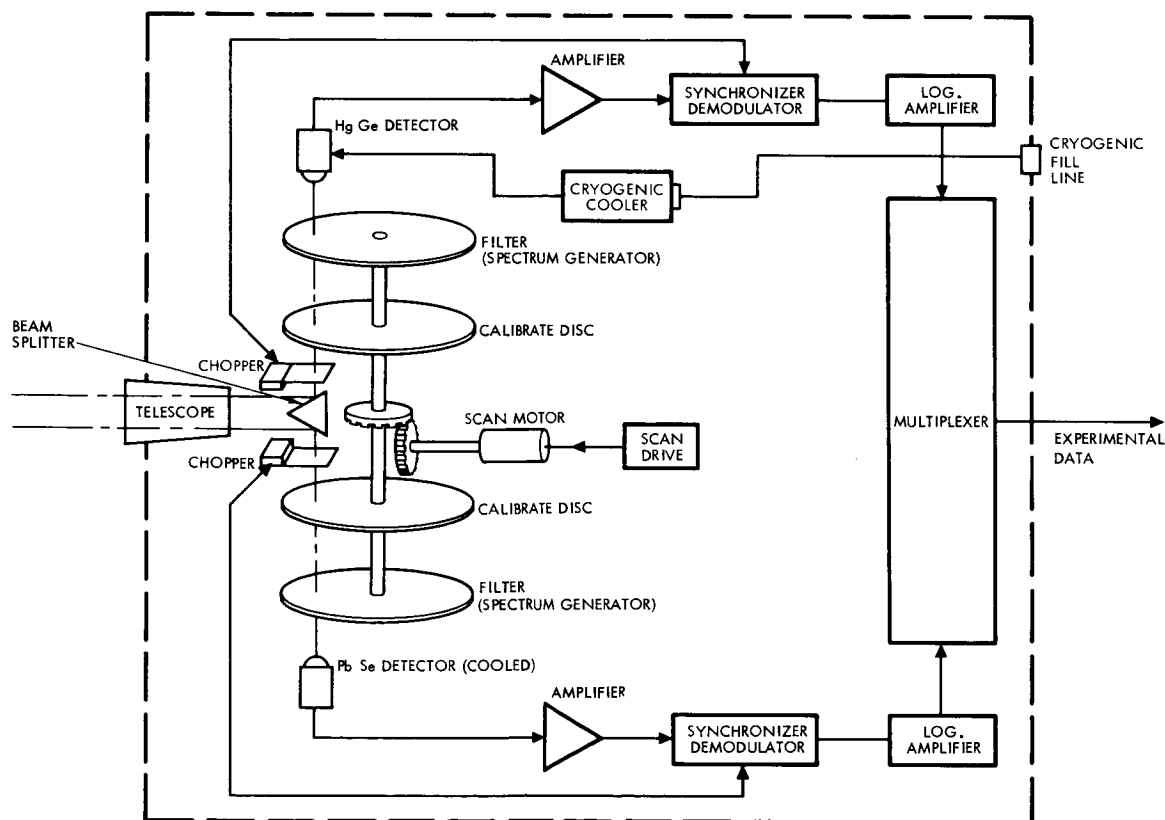


Figure 2-2

problem and requires first of all that the choppers as well as the detector should be cooled. It should be noted that laboratory calibration and test will also require the same cooling and even then may be impossible over the upper end of the range, so careful design of the instrument on a component basis for the desired signal to noise ratio is essential.

The instrument should view the planet along the vertical line-of-sight toward both the day and night sides, including terminator crossings whenever possible. It is desirable to look at the entire disc of the planet, correlated with the visual imaging equipment, and to 10 degrees beyond the edge of the disc.

2.1.3 Infrared Radiometer

Purpose

- a) Determine the temperature of the surface of Mars relative to geographical locale
- b) Aid in identifying cloud features by temperature measurements

Instrument This experiment employs a scanning two-channel radiometer. The instrument described here includes a rotating scan mirror, a mechanical chopper and two optical filter and infrared detector systems. Various spectral ranges and various types of detectors could be selected for the two channels, and electronic chopping of the detector outputs could be substituted for mechanical chopping of the incident light beam. The spectral ranges selected by way of illustration are shown in Figure 2-3; they are 0.5 to 0.75 μ , overlapping the visible, and 8 to 13 μ .

An alternate possibility is to substitute a longer wavelength range, instead of the visible and near-infrared region, for example 18 to 25 μ which includes strong water vapor bands. In the latter case, two identical thermistor bolometers could well be used as detectors for the two channels.

However, in the instrument chosen the 8-13 μ range employs such a bolometer detector while for the 0.5 to 0.75 μ range a silicon photodiode detector is suggested. If greater sensitivity is desired, a photomultiplier tube of appropriate spectral response could be substituted.

With the scan mirror in its center or reference position the instrument should view the planet along the vertical line-of-sight correlated



PRELIMINARY SPECIFICATION

Infrared Radiometer

Performance Characteristics

Spectral Range
Channel 1 - $0.5\mu \rightarrow 0.75\mu$
Channel 2 - $8.0 \rightarrow 13.0\mu$

Spectral Resolution
Dependent upon broadband filters (or filter-detector cut-off combinations) selected by experimenter

Spectral Scan - none

Dynamic Range - $140^\circ\text{K} \rightarrow 300^\circ\text{K}$

Sensitivity
 0.1°K at 300°K
 0.4°K at 140°K

Resolution Ground - $10 \times 10 \text{ Km}$ at 1000 Km periaopsis

Temperature - 0.5°K at 185°K

Field-of-View
Scans thru 60 deg
Instantaneous - 10 milliradians

Interfaces

ELECTRICAL

Commands
Power - ON-OFF
Calibrate

Telemetry

Data
Channel 1 - 6 - 8-Bit words
Channel 2 - 6 - 8-Bit words
Total bps - 2400

Diagnostics
Calibration monitor
High sensitivity status monitor
mirror position
Channel 1 detector temperature
Channel 2 detector temperature
Electronics package temperature

Power
20 watts

MECHANICAL

Weight
20 lb

Dimension
 $13.00 \times 8.00 \times 6.00$

Volume
 624 in.^3

Pointing
Aligned to TV to within $\pm 0.3 \text{ Deg}$

Alignment
Accuracy of alignment shall be known to within $\pm 0.1 \text{ Deg}$

Operating Temperatures
Electronics - $30^\circ\text{C} \rightarrow 50^\circ\text{C}$
Radiometer - $20^\circ\text{C} \rightarrow 50^\circ\text{C}$

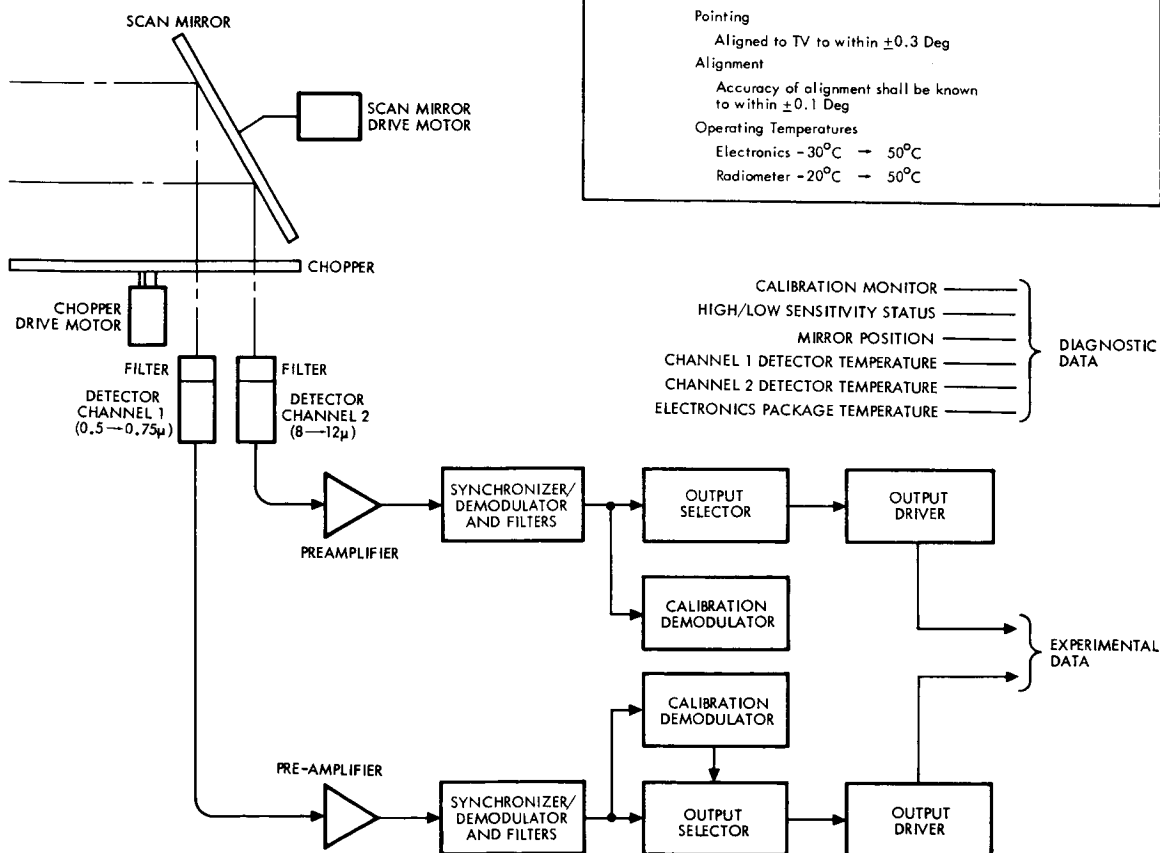
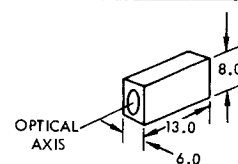


Figure 2-3

with the visual imaging equipment. Observations of both the day and the night side are desirable, and temperature measurements of the terminator crossings are especially important. To map the entire Martian surface most advantageously, the scan mirror should scan at about 0.5 rev/sec in a direction at right angles to the orbital plane of the spacecraft and to ± 60 degrees from the plane.

Three degrees of freedom are required for a linearly scanning device to scan at all times perpendicularly to the flight path. By rotation of the scanning mirror or by scanning in a circular mode two degrees of freedom are sufficient. Also, for some orbits, two degrees of freedom and a linear scan will provide acceptable data.

2.1.4 Ultraviolet Spectrometer (See Figure 2-4)

Purpose

- a) Detect the presence of as many constituents of the atmosphere of Mars as possible, and to identify the various atomic, molecular, and ionic species
- b) Measure the scale height of atmospheric constituents
- c) Measure the Rayleigh scattering from the lower atmosphere and the ultraviolet reflectivity of the planetary surface
- d) Detect, if possible, ultraviolet absorption due to atmospheric constituents such as ozone, oxygen, and nitrogen
- e) Attempt to detect atmospheric aurora and nightglow

Instrument. The instrument consists of an occulting slit telescope, a scanning Ebert (grating) spectrometer, and two photomultiplier tube detectors with associated electronics. The two photomultiplier tubes consist of one with a cesium iodide cathode, sensitive over the wavelength range from 1100 to 2200 Å and one with a bi-alkali cathode which is sensitive from 1500 to 4300 Å.

The experiment objectives listed above lead to two viewing modes for the ultraviolet spectrometer, as follows:

- a) Look at the planet's disk in order to:
 - Determine the ultra-violet reflectivity from the lighted side



PRELIMINARY SPECIFICATION

Ultraviolet Spectrometer

Performance Characteristics

Spectral Range

Channel 1 (CsI) 1100Å — 2200Å

Channel 2 (Bialkali) 1500Å — 4300Å

Spectral Resolution

Nominal 20Å for first order spectra

Spectral Scan

Grating rotates at a rate of 16 deg each 3 second interval

Dynamic Range

Channel 1 $50 \rightarrow 50 \times 10^3$ Rayleighs*/20Å interval

Channel 2 $50 \times 50 \times 10^6$ Rayleighs*/20Å interval

*1 Rayleigh = 10^6 photons/CM²/sec

Accuracy — 5 percent (for detector output)

Field-of-View

2.0 deg x 5.0' (4.0 in. x 4.0 in aperture) to view planet

Interfaces

ELECTRICAL

Commands

Power — ON-OFF

Calibration

Telemetry

Data
12 — 8-Bit word

Diagnostics

Grating position monitor

Multiplexer status

Velocity/Height power supply No. 1 voltage and temperature monitor

Velocity/Height power supply No. 2 voltage and temperature monitor

Sequencer status monitor

Electronics package temperature monitor

MECHANICAL

Weight

30 lb

Dimensions

21.50 x 11.00 x 6.00 (excluding sunshade extension)

Volume

1419 in.³

Pointing

A) At planet's disc — when observing terminator, slit shall be parallel to the terminator

B) Near planet's limb — slit shall be parallel to the horizon

Alignment Accuracy

Position of instrument optical axis with respect to visual imaging equipment shall be known to ± 0.1 Deg

THERMAL

Operating Temperature

0.0°C — +40.0°C

DASHED LINES SHOW SUN SHADE POSITION

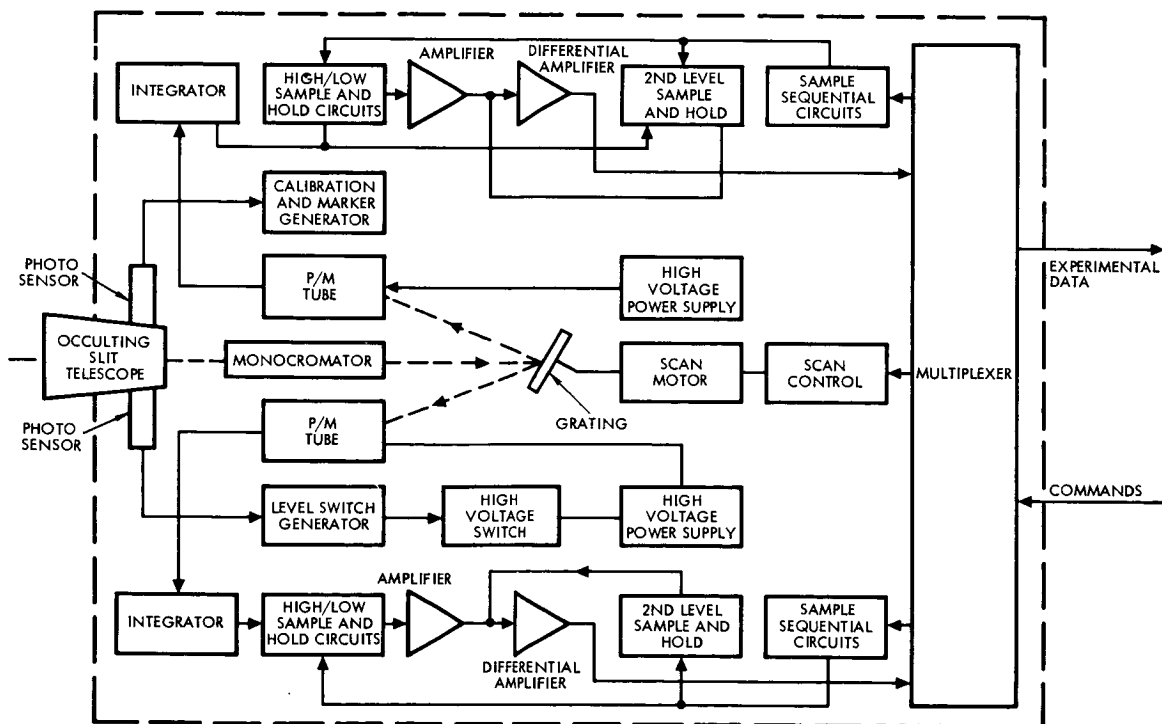
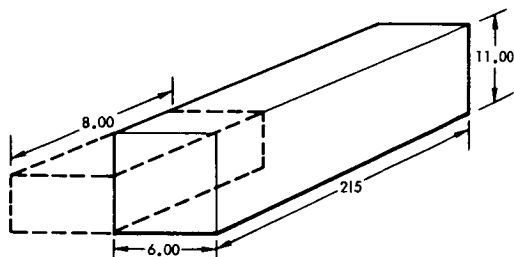


Figure 2-4

- Detect atmospheric scattering, absorption, and emission above the lighted side
 - Detect nightglow and aurora above the dark side, and especially in an eclipse
 - Observe crossings of the terminator; the spectrometer entrance slit should then be oriented parallel to the terminator
- b) To look through long atmospheric paths beyond the limb of the planet and passing at various grazing distances in order to detect the presence and determine the altitude distribution of various atmospheric constituents; for these observations the entrance slit of the spectrometer should be oriented parallel to the horizon.

The relative positions of sun, planet, and spectrometer determine which measurements are appropriate at any given time, and since the spacecraft orbit gradually changes its orientation with respect to the sun throughout the year, flexibility to select the most advantageous view-direction for the ultraviolet spectrometer can be achieved only if the spectrometer has complete freedom of orientation.

The instrument is mounted on the planetary scan platform. If the platform has only two degrees of freedom, the ultraviolet spectrometer must have the additional capability of rotation around its optic axis in order to orient the slit parallel to the terminator or alternately, to the horizon, whereas with a gimbaled three-axis platform this capability is already provided. In either case, it is desirable also to vary the viewing direction of the spectrometer with respect to the line-of-sight of the visual imaging equipment in order to maintain maximum ground overlap as well as to achieve a specified line-of-sight grazing distance from the planet's limb.

Hence, the pointing direction of the ultraviolet spectrometer should be variable by ground command and should be known in any position with reference to the line-of-sight of the visual imaging equipment to an accuracy goal of ± 0.1 degree.

2. 1. 5 Radio Occultation

Purpose

- a) To determine surface pressure and scale height of the neutral



atmosphere of Mars; to detect if possible any change in scale height in the upper atmosphere

- b) To determine electron density and scale height as a function of altitude in the Martian ionosphere
- c) To correlate variations in both the neutral and the charged particle parameters with surface location, temporal, and day-night effects

Instrumentation. In general terms this experiment is an electromagnetic probe of the Martian atmosphere. Many experimental arrangements have been proposed or used in recent radio wave soundings of the atmospheres or ionospheres of the earth, moon, Mars, and Venus and of the interplanetary plasma. The simplest of these is a single frequency radio occultation experiment similar to that successfully carried during the one-eclipse pass behind Mars by Mariner IV. The change in phase of the regular spacecraft down-link signal was measured during the 100 seconds just before the spacecraft passed behind Mars and again just after its emergence on the other side. The experiment was completely successful and yielded detailed information about the properties of both the ionosphere and the neutral atmosphere and showed the ionospheric charge density to be reduced by a factor of about 20 on the night side. It would be valuable to perform this experiment repeatedly with the first Voyager orbiter and to correlate the results with seasonal, location, and solar activity effects. Since the experiment requires no equipment beyond the normal communication down-link of the spacecraft, it is assumed that this experiment is a part of the core experiments of the hypothetical science payload.

In addition it should be noted that only the provision of a second transmitter, or receiver or alternatively, a transmitter and receiver, is required to perform somewhat more advanced experiments. As an example, the radio propagation experiment for Pioneer VI has two ground transmitters of 49.8 and 423.3 MHz respectively, and aboard the spacecraft a dual channel phase-locked-loop receiver which measures the relative phase of the modulation envelopes of the two carrier frequencies. Since the higher frequency is relatively unaffected by the presence of ionization, the phase shift of the lower frequency signal leads to a value

for the integrated electron density along the path. In principle, the transmitters could be on the spacecraft with the receiver on earth. Weight and power considerations determine the choice and in the present case of a Mars orbiter the radio probe experiment could be made a two-frequency experiment merely by the substitution of the dual channel receiver on the spacecraft together with its data processing capacity.

A third possibility is to perform either a single frequency occultation experiment or a two-frequency radio propagation and occultation experiment between the two orbiting spacecrafts of a given mission (launch). In this case one spacecraft would carry the transmitter (no extra equipment for a single frequency experiment) and the other spacecraft would have the receiver and data processing equipment as an added experiment package.

A final variation concerns the possibility that each spacecraft will, when in orbit, launch a subsatellite or orbiting experiment capsule (OEC). Clearly, either the single frequency or bi-frequency propagation and occultation experiment could be performed between such a mother-daughter satellite pair, and has been proposed as one of the three or four experiments to be carried on an OEC. The present status of the OEC proposal is not known and its potentiality for the radio occultation experiment is simply noted here in passing.

To summarize: For the present hypothetical science payload it is assumed that a single-frequency radio occultation experiment similar to the Mariner IV experiment will be performed, requiring no additional equipment. It is noted that with the addition of a receiver and/or a receiver and transmitter a bi-frequency experiment involving either one or two spacecraft can be performed. The TRW spacecraft design has sufficient capacity and flexibility to accommodate this alternative.

2.2 HYPOTHETICAL PHOTO-IMAGING SYSTEM

2.2.1 Mapping Objectives

Three levels of resolution have been selected for the photo-imaging coverage of Mars. The lowest level, 1000 meters ground resolution, is equivalent to terrestrial photography of the moon, which is limited to this

resolution by atmospheric inhomogeneities. The lunar photography from earth provided a good basis for planning and carrying out more detailed photography from lunar probes; the same usefulness can be expected of similar Martian photography. Pictures with 1000 meter resolution would reveal details on the Martian surface 200 or more times smaller than can be observed from earth. Complete coverage of the Martian surface overflown, either in color (using three exposures of each area, taken with red, green, and blue filters) or in black and white, or both, can be obtained with only 5×10^9 data bits.

A ground resolution of 10 meters is about the highest obtainable consistent with mission constraints. Capability for getting some information at this level of detail is desirable, but even if the entire transmission capacity for the mission were allotted to this, pictures of only one-half of one percent of Mars surface could be sent to earth. Thus an intermediate resolution level, providing a compromise between coverage and detail, should be included. This was chosen as 100 meters, the geometric mean between the high and low extremes. At this resolution a substantial fraction of the surface of Mars can be covered.

In summary, the photo-imaging objectives have been specified to include:

- a) Color and/or black and white coverage of all the Martian surface overflown with suitable illumination at 1000 meters resolution
- b) Monoscopic and selected stereo coverage of about 25 to 40 percent of the Martian surface at 100 meters resolution
- c) Limited high resolution coverage (0.1 to 0.25 percent of the surface) at 10 meters resolution

The subject of photo-imaging objectives and the rationale leading to them has been discussed in greater detail in Volume 11 of this report.

2.2.2 Design Implementation

The hypothetical photo-imaging system which has been configured for the Voyager spacecraft consists of five television cameras fulfilling the requirements of low and medium resolution mapping coverage and

high resolution imaging of selected areas. All three types of imaging may be obtained simultaneously or sequentially.

Low Resolution Camera. The low resolution television camera will provide either monochrome or color coverage with resolution of 1 kilometer from the altitude of 1000 kilometers at periapsis. The use of color coverage will be of benefit in the identification of vegetation on the surface of Mars and in determining the seasonal variations in cloud formations and dust storms. When used in the color mode with a field of view of 500 x 500 kilometers at periapsis, 11 color pictures (33 monochrome frames — red, green, and blue in sequence) will provide coverage of approximately $6.25 \times 10^6 \text{ km}^2$ or 4.3 percent of the total area of Mars in one orbit. The lateral coverage of the optical field of view increases to 1120 kilometers at an angle of 65 degrees from periapsis. A 1-1/2 inch RCA type 8521 electromagnetic vidicon has been chosen for this camera having adequate resolution, size of format, and moderate power requirements. In order to permit slow-scanning, an ASOS photoconductive target will be used in conjunction with a quartz storage layer. This target and storage layer combination has been developed by RCA for use in the APT (automatic picture taking) system for the Nimbus meteorological satellite program.

Medium Resolution Cameras. For medium resolution coverage, three return-beam vidicon cameras are proposed with a resolution of 100 meters from the altitude of 1000 kilometers at periapsis. The requirement for three cameras is dictated primarily by the limitation in the data acceptance rate of the associated digital tape recording equipment. The cameras are used in sequence, resulting in a continuous flow of digital video information to the tape recording equipment at the rate of 700 kb/sec. During one orbit, 27 frames of information may be obtained (9 frames per camera), permitting continuous coverage over a ground track length of 4480 kilometers. The number of frames which may be obtained in one orbit is limited by the data capacity of the digital tape recording equipment. The lateral ground coverage at periapsis is 150 kilometers, increasing to 225 kilometers at 37.5 degrees from periapsis. The 2-inch return beam vidicon has been selected for this application, primarily from the standpoint of high resolution and relatively large



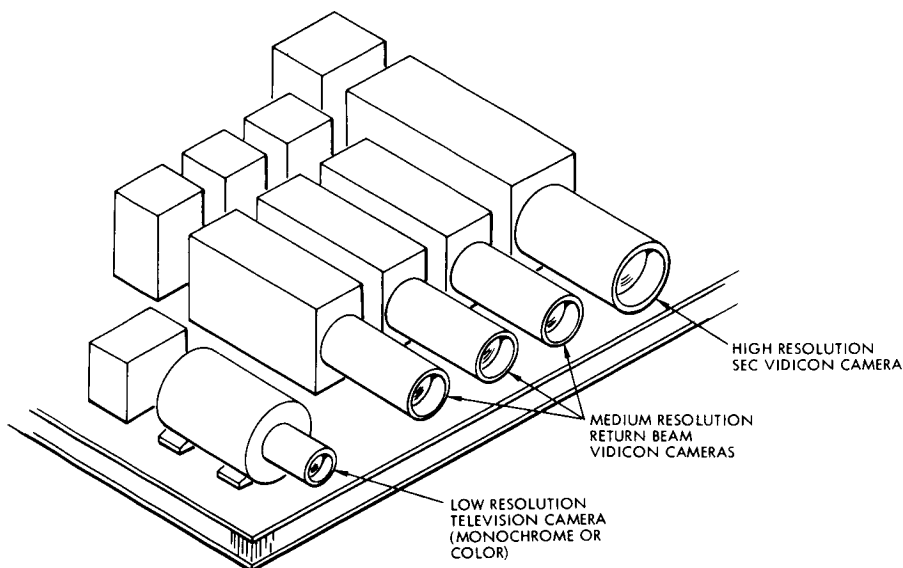
format. The resolution of this tube is presently 3100 TV lines at 10 percent response, permitting the use of a nominal resolution of 3000 x 3000 resolution elements on a 1-inch square raster. Again, the use of the slow-scan ASOS photoconductive target in conjunction with a quartz storage layer is proposed in order to accommodate the frame rate of 110 seconds, permitting the use of an acceptable low data rate to the digital tape recorders.

High Resolution Cameras. For high resolution imaging, one SEC vidicon camera is proposed. From an altitude of 1000 kilometers at periapsis, a resolution of 10 meters will be obtained with 10 x 10 kilometer ground coverage per frame. The primary basis for selection of the SEC vidicon for high resolution imaging has been the characteristic of high luminous sensitivity which permits an extremely short exposure time (1 millisecond), eliminating the requirement for image motion compensation. During one orbit, 18 frames of information may be obtained over a ground track length of 4480 kilometers with a minimum spacing between frames of 220 kilometers along the ground track (at periapsis). The minimum distance between adjacent frames is determined by the rate of acceptance of data by the digital tape recorder used with the SEC camera, which is configured to accept data at a rate of approximately 700 kb/sec. The SEC vidicon which is proposed for this camera is the Westinghouse type 5419B. The resolution at this time is 1000 TV lines. In order to obtain the required 2000 TV line resolution, additional development will be required in the design of the electron optical system in order to reduce the diameter of the electron beam which scans the photoconductive target.

Equipment Configuration on Planetary Scan Platform. The equipment configuration and preliminary specifications for the photo-imaging system are defined in Figure 2-5. One low resolution television camera, three medium resolution return beam vidicon cameras, and one high resolution SEC vidicon camera will be mounted on the planetary scan platform (PSP) and boresighted. If stereo photography is desired, this may be accomplished by fore and aft pointing of the PSP, or by obtaining overlapping coverage on sequential orbits. The total weight of the camera equipment on the PSP will be 230 pounds, and the average power for all

PRELIMINARY SPECIFICATION

Hypothetical Photo Imaging System



Purpose

To provide complete photographic coverage of the planet Mars at low resolution (1 kilometer) in color, mapping of large areas of Mars at medium resolution (100 meters), and imaging of selected areas at high resolution (10 meters)

Performance Characteristics

LOW RESOLUTION CAMERA

Sensor	RCA 1-1/2 inch vidicon with slow-scan target
Field of View	29 x 29 degrees
Ground Coverage per frame	500 x 500 kilometers
Nominal Ground Resolution	1 kilometer
Number of Frames per Orbit (maximum)	33 frames (11 color pictures)
Photography duration/orbit	48 minutes (maximum)

MEDIUM RESOLUTION CAMERAS

Sensor	RCA 2 inch return beam vidicons with slow-scan target
Field of View	8.6 x 8.6 degrees
Ground Coverage per frame	150 x 150 kilometers
Nominal Ground Resolution	100 meters
Number of Frames per Orbit (maximum)	27 (9 per camera)
Photography duration/orbit	26 minutes (maximum)

HIGH RESOLUTION CAMERA

Sensor	Westinghouse 3 inch SEC vidicon
Field of View	35 x 35 arc-minutes
Nominal Ground Resolution	10 meters
Number of Frames per orbit (maximum)	18
Ground Coverage per frame	10 x 10 kilometers
Photography duration/orbit	26 minutes (maximum)

Physical Characteristics

CAMERA EQUIPMENT MOUNTED ON PLANETARY SCAN PLATFORM

Weight:	230 pounds
Size:	3.64 cubic feet
Power:	195 watts (maximum) 11.3 watts (average)

TOTAL SYSTEM - INCLUDING 4 TAPE RECORDERS

Weight:	302 pounds
Size:	5.2 cubic feet
Power:	275 watts (maximum) 29.3 watts (average)

Figure 2-5

five cameras is estimated to be 11.3 watts. An associated 72 pounds and 80 watts peak (while recording) will be required for the four tape recorders required in the data storage system required for this photo-imaging system. The tape recorders will not be mounted on the PSP.

2.2.3 Mapping Coverage

The mapping coverage obtained by the photo-imaging system in one orbit is defined in Figure 2-6. Note that the lateral coverage of the three subsystems increases with increasing displacement from periapsis due to the corresponding increase in altitude. The maximum coverage obtained by the low resolution television camera in one orbit will be determined by the range of solar illumination angles permissible for color photography. However, the maximum one orbit coverage by the medium resolution cameras is limited by the storage capacity of the tape recording equipment. For high resolution imagery, one tape recorder is allocated, permitting as many as 18 selected areas, each 10 x 10 kilometers to be imaged within one orbit.

During a 6 month lifetime in orbit with a total data transmission capacity of 2.64×10^{11} bits, the coverage of Mars which may be obtained is defined in Table 2-1. Note that complete coverage may be obtained with the low resolution camera in color. Thirty-six percent of Mars may be mapped with medium resolution, and 0.11 percent with high resolution, assuming that 1600 selected areas are viewed.

The values presented in Table 2-1 are based upon the nominal values of ground coverage of each of the three television subsystems at periapsis. Note that the coverage obtained will increase as the photographic interval is increased beyond periapsis due to the increase in altitude inherent in the elliptical orbit (see Figure 2-4).

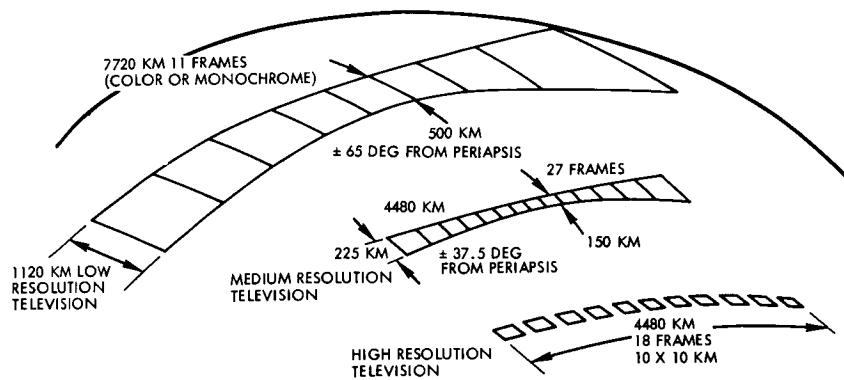


Figure 2-6

MAPPING COVERAGE PER ORBIT (MAXIMUM)-HYPOTHETICAL PHOTO IMAGING SYSTEM. The low and medium resolution television cameras will obtain continuous coverage over a ground track distance of 7720 and 4480 Kilometers, respectively. The high resolution television camera can photograph as many as 18 frames over a ground track distance of 4480 Kilometers. The ground coverage traces of the three cameras are displaced laterally on the diagram for purposes of illustration.



Table 2-1. Coverage of Mars Obtained With Low, Medium, and High Resolution Television Camera

	<u>Low Resolution TV (Color)</u>	<u>Medium Resolution TV</u>	<u>High Resolution TV</u>
Nominal ground resolution	1 km	100 m	10 m
Nominal coverage frame	500 x 500 km	150 x 150 km	10 x 10 km
Bits/frame (6-bit encoding)	8.58×10^6	77.04×10^6	34.3×10^6
Frames/orbit (maximum)	11 color pictures (33 frames)	27 (9 per camera)	18
Data/orbit (maximum)	2.83×10^8 bits	2.08×10^9 bits	6.17×10^8 bits
Ground coverage/frame (nominal)	$2.5 \times 10^5 \text{ km}^2$	$2.25 \times 10^4 \text{ km}^2$	100 km^2
Area of Mars	$1.46 \times 10^8 \text{ km}^2$	$1.46 \times 10^8 \text{ km}^2$	$1.46 \times 10^8 \text{ km}^2$
Data allotment (Total data link capacity = 2.64×10^{11} bits/6 mo)	1.52×10^{10} bits	1.94×10^{11} bits	0.55×10^{11} bits
Number of frames	585 (color pictures) 1755 frames	2620	1600
Percentage of Mars area covered	100	36	0.11



3. HYPOTHETICAL SCIENCE PAYLOAD INTEGRATION

3.1 INTRODUCTION

The hypothetical payload defined in Section 2 above consists of a core group of science experiments used basically for determination of Mars atmospheric constituents and surface temperatures, and a group of five television cameras. This section describes the detailed integration of this payload with the spacecraft.

All of the experiments in the hypothetical payload must either view the planet or the limb of the planet and are therefore mounted in the gimballed planetary scan platform. For future growth the basic size of the planetary scan platform has been chosen to accommodate three instruments from the list of additional instruments given in Section 1.3.2 (the gamma ray spectrometer, meteoroid flash detector, and atmospheric polarimeter), which also require gimballed pointing.

The data handling system and the system for switching and sequencing for the hypothetical payload have also been configured to accommodate changes in the science experiments since the experiments have not yet been rigidly defined. The interfaces with the science experiments are set up so that the system is completely flexible and new experiments may be wired into the equipments without changing previous wiring arrangements. For example, the remote units in the planetary scan platform are designed to accept the three additional instruments listed above, as is the thermal control system.

3.2 MECHANICAL INTEGRATION

The mechanical integration for the hypothetical payload consists of determining the experiment requirements for view angles, pointing, mounting, and general environmental protection. Each instrument will be aligned with the planetary scan platform gimbal systems to the tolerances shown in Table 3-1.

The general external configuration of the deployed planetary scan platform is illustrated in Figure 3-1. Note that the platform is supported on a

Table 3-1. Hypothetical Science Payload - Mechanical Interface Requirements

Experiment	Sensor					Colocated Electronics	
	Field of View (deg)	Alignment (deg)	Aperture (in.)	Volume (in. ³)	Weight (lb)	Volume (in. ³)	Weight (lb)
Photo-Imaging Equipment	Low-resolution TV camera	29	0.05	2.5 diameter	212	10	10
	Medium-resolution (RBV) vidicon cameras (3 each)	8.6	0.05	7 diameter	532	45	10
	High-resolution (SEC) vidicon camera	0.6	0.05	8 diameter	1,288	35	10
Core Experiments	UV spectrometer	2 x 6	0.1	4 x 4	1,536	20	10
	Broadband IR spectrometer	5	0.5	10 diameter	1,848	20	-
	High-resolution IR spectrometer	1 x 4	0.5	4 diameter	2,400	30	-
	IR radiometer	120	0.1	5 diameter	220	10	10
PSP Control	Mars sensor (2)	45 each side	0.05	8 x 3	70 each	5.5 each	7.0 each

Note: Sensors are boresight mounted
 Sensors require protective covers for micrometeorite erosion and thermal control
 Sensors must not face sun

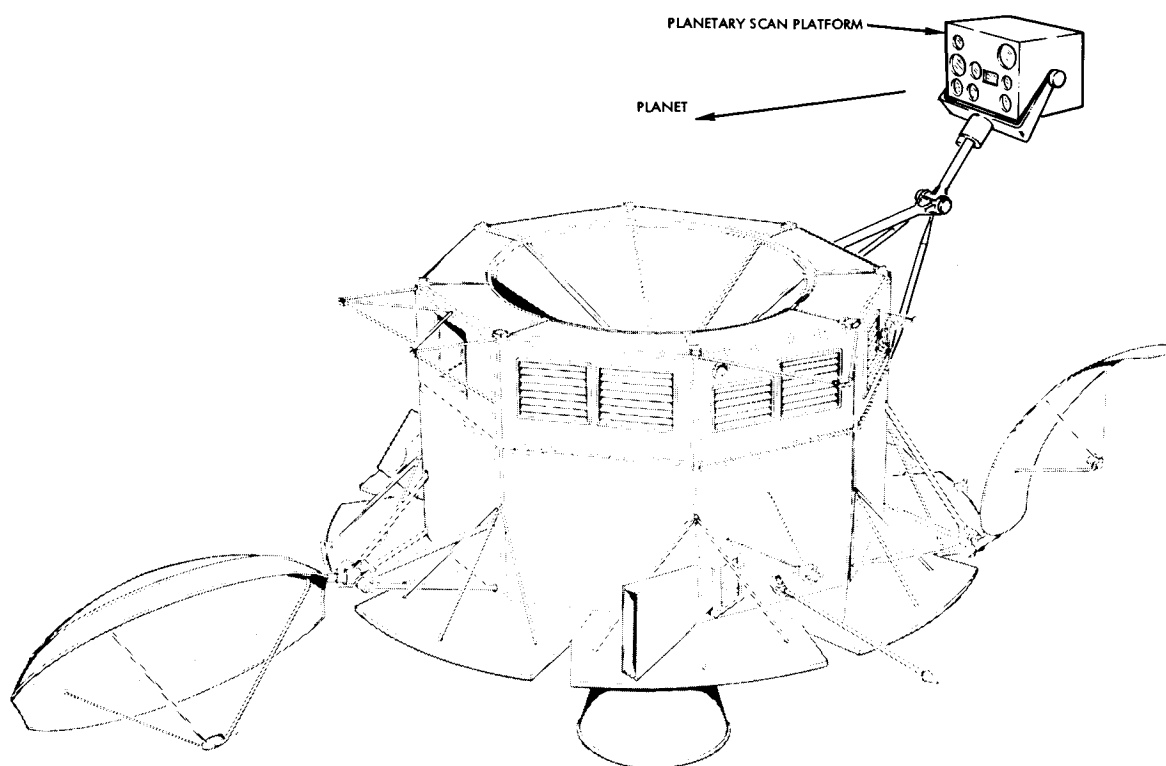


Figure 3-1
HYPOTHETICAL SCIENCE PAYLOAD is contained exclusively in the Planetary Scan Platform. It consists of the all-TV photo imaging-system and a "core" group of scientific instruments.

shaft and yoke gimbal drive positioning mechanism. This shaft in turn is mounted to a hinge-drive at the apex of a three-strut structure affixed to the equipment module adjacent to side panel No. VI. The manner in which this mechanism and the spacecraft control system couple to position the platform boresight face is described in paragraph 3.5 below. Viewing positions are shown on Figure 3-2.

The planetary scan platform interior volume bounded on all six sides by meteoroid protection panels is thermally controlled. Its interior dimensions are 45x37 in. on the face x35 in. in depth. This provides approximately 11.5 ft² of mounting area on the viewing surface and a usable volume of about 34 ft³. There is a full width, full depth shelf dividing the volume into two parts for mounting the majority of the experiments. In this way "floor plan" flexibility is provided to insure that the center of gravity can be located on the yoke drive axis and at the same time the inertia about this axis can be minimized. One arrangement of the hypothetical science experiments is illustrated in Figure 3-3 and 3-4 and their mechanical interface requirements are tabulated on Table 3-1.

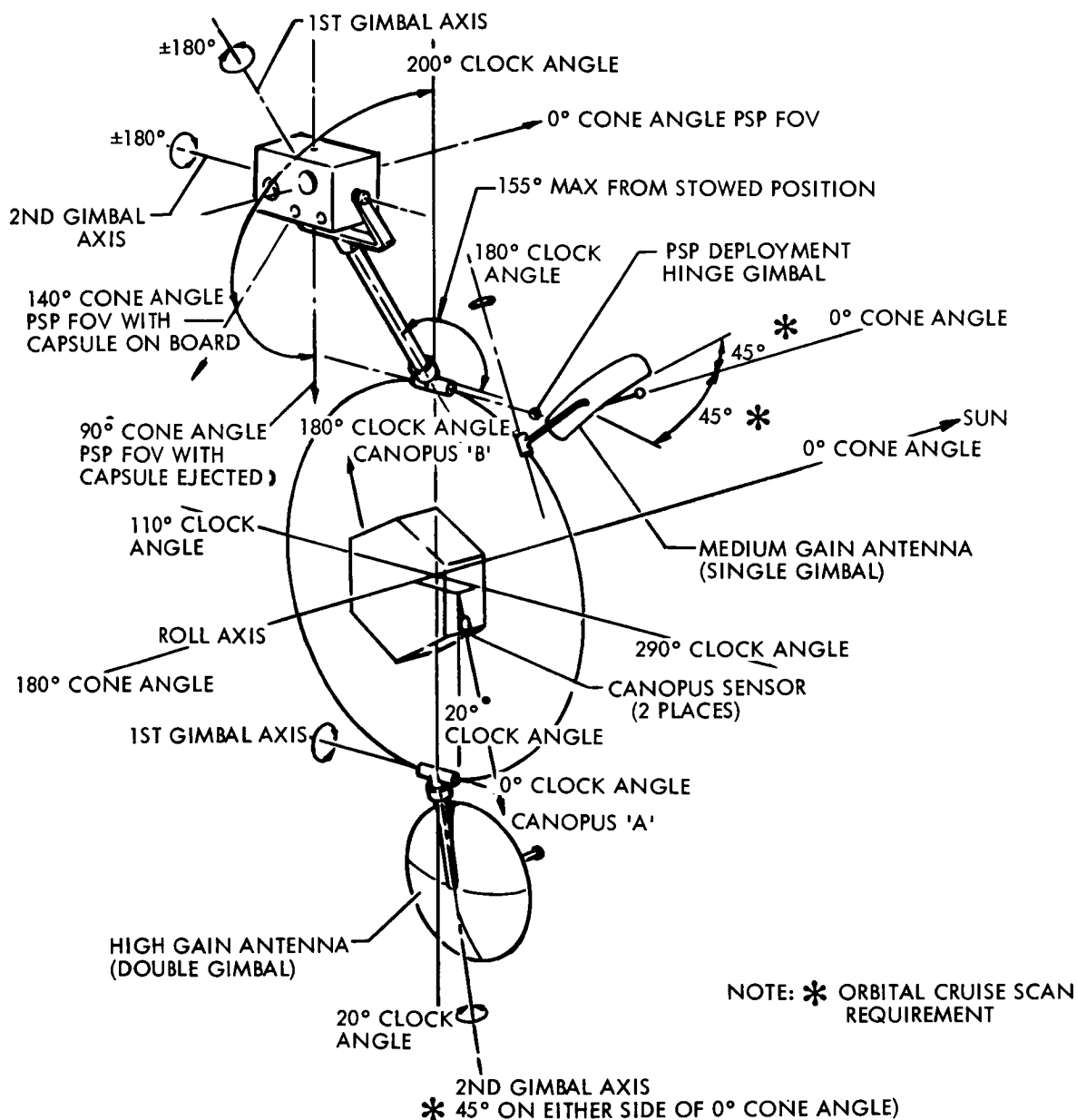
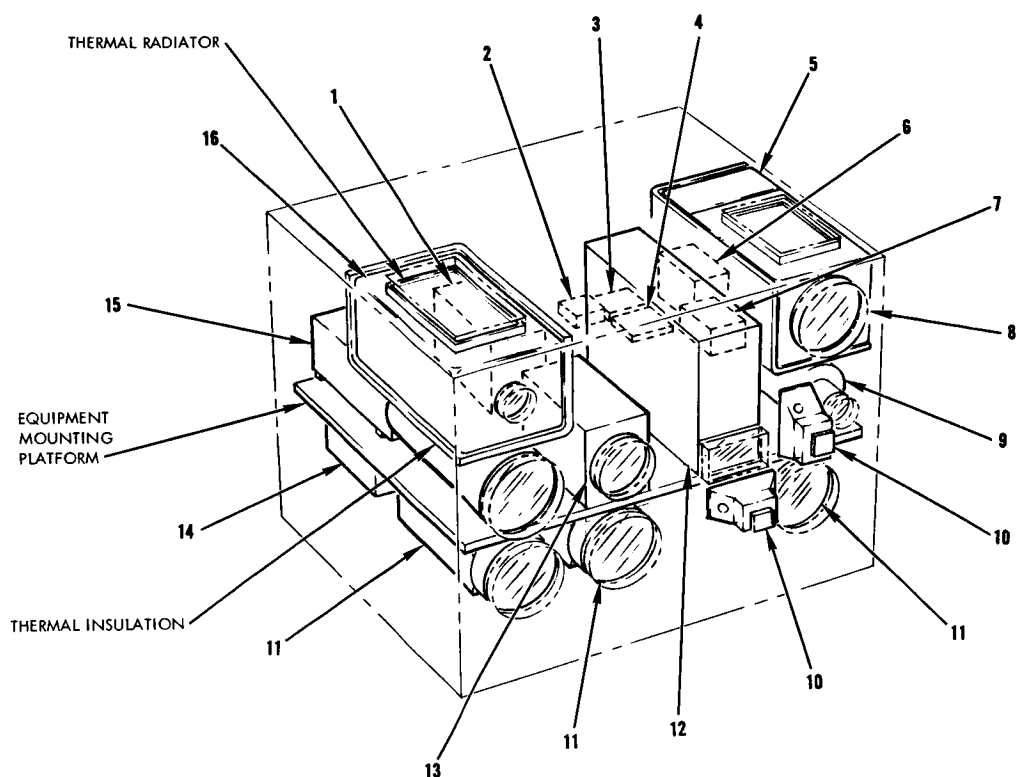


Figure 3-2

PSP GIMBAL AXES and allowable rotations about each axis are shown with reference to locations in cone and clock angles.



- | | |
|---|--|
| 1. HIGH RESOLUTION SEC VIDICON CAMERA ELECTRONICS | 9. LOW RESOLUTION COLOR TV CAMERA |
| 2. PSP REMOTE PHOTO-IMAGING MULTIPLEXER | 10. MARS SENSOR (2 PLACES) |
| 3. PSP REMOTE MULTIPLEXER UNIT | 11. MEDIUM RESOLUTION RBV CAMERA (3 PLACES) |
| 4. PSP REMOTE DECODER UNIT | 12. ULTRAVIOLET SPECTROMETER |
| 5. CRYOGENIC REFRIGERATOR | 13. INFRARED RADIOMETER |
| 6. PSP REMOTE SCIENCE SWITCHING ASSY. | 14. MEDIUM RESOLUTION RBV CAMERA ELECTRONICS |
| 7. LOW RESOLUTION COLOR TV CAMERA ELECTRONICS | 15. HIGH RESOLUTION SEC VIDICON CAMERA |
| 8. BROADBAND INFRARED SPECTROMETER | 16. HIGH RESOLUTION INFRARED SPECTROMETER |

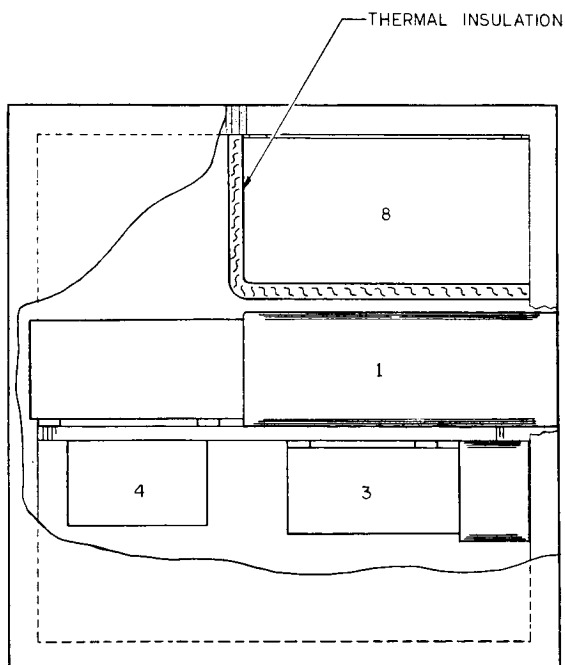
Figure 3-3

HYPOTHETICAL SCIENCE PAYLOAD, including five TV cameras, is arranged on PSP as shown in preliminary layout.

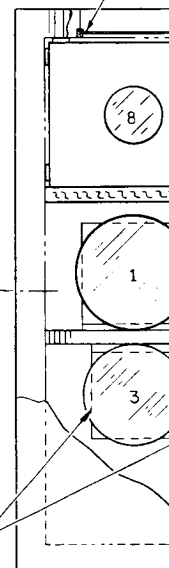
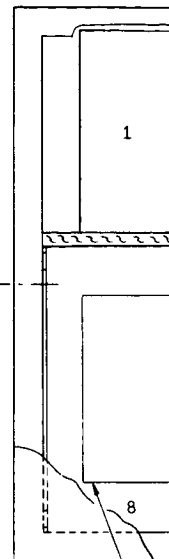
EQUIPMENT LIST

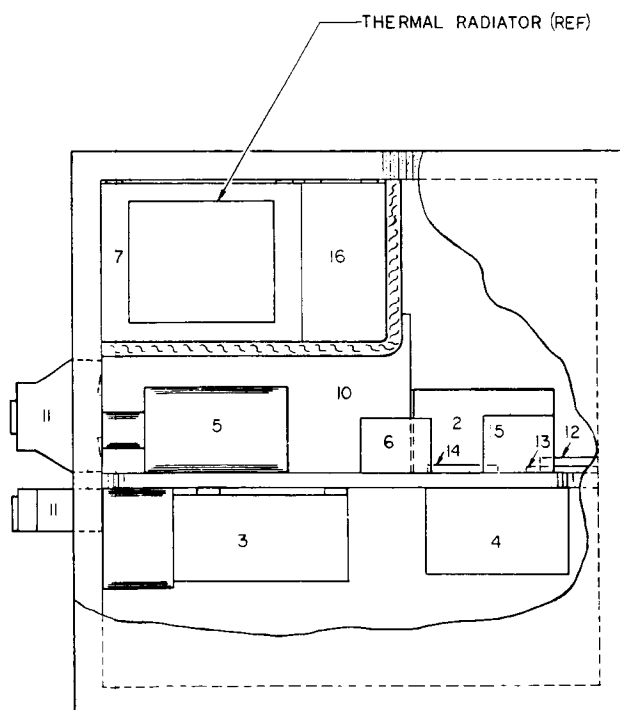
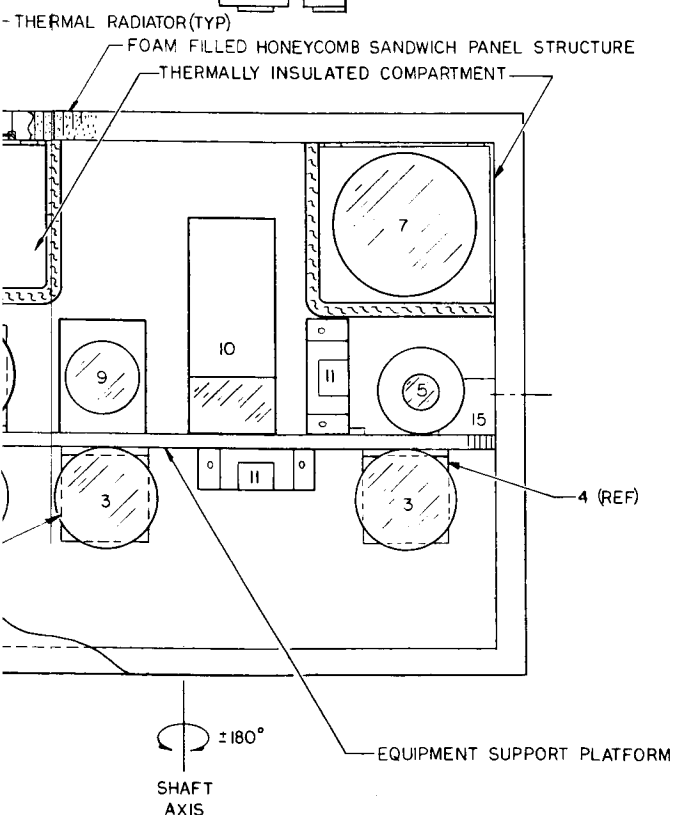
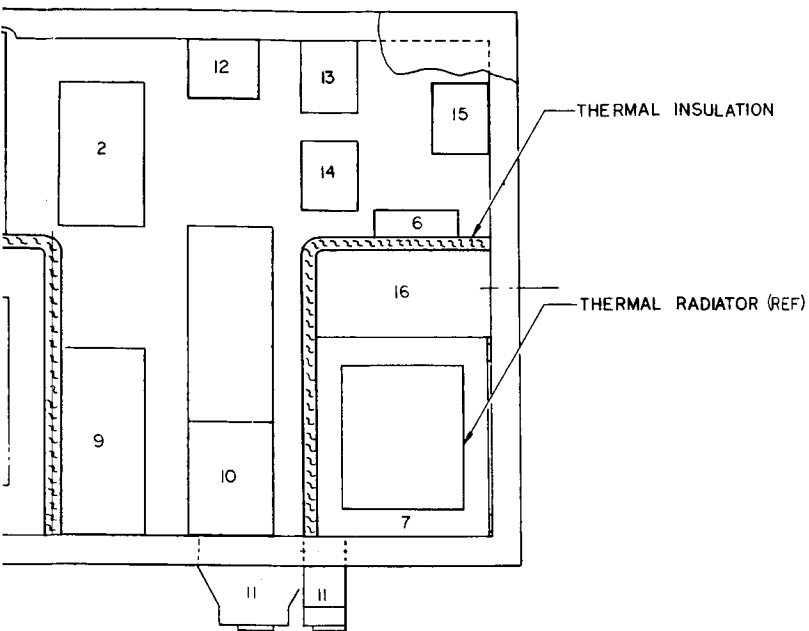
1. HIGH RESOLUTION SEC VIDICON CAMERA
2. HIGH RESOLUTION SEC VIDICON CAMERA ELECTRONICS
3. MEDIUM RESOLUTION RBV CAMERA (3 PLACES)
4. MEDIUM RESOLUTION RBV CAMERA ELECTRONICS (3 PLACES)
5. LOW RESOLUTION COLOR TV CAMERA
6. LOW RESOLUTION COLOR TV CAMERA ELECTRONICS
7. BROADBAND INFRARED SPECTROMETER
8. HIGH RESOLUTION INFRARED SPECTROMETER
9. INFRARED RADIOMETER
10. ULTRAVIOLET SPECTROMETER
11. MARS SENSOR (2 PLACES)
12. PSP REMOTE PHOTO-IMAGING MULTIPLEXER
13. PSP REMOTE MULTIPLEXER UNIT
14. PSP REMOTE DECODER UNIT
15. PSP REMOTE SCIENCE SWITCHING ASSY
16. CRYOGENIC REFRIGERATOR

YOKE AXIS $\pm 180^\circ$



4 (REF)





0 4 8 12 16 20
SCALE IN INCHES

TRW SYSTEMS TRW INC. ONE SPACE PARK • REDONDO BEACH • CALIFORNIA	
VOYAGER PRELIMINARY DESIGN- PLANETARY SCAN PLATFORM, HYPOTHETICAL SCIENCE PAYLOAD	20 SEPT 1967 AD7-178

Figure 3-4

ENGINEERING SKETCH OF PSP indicates equipment mounting, thermal control and micromechanical protection.



To meet the field of view and aperture requirements of each of the experiment sensors it was necessary to provide one aperture on the platform viewing surface for each. To protect the sensors when not in use, each hole is covered by an insulated, meteoroid protected, deployable flap. Only those flaps covering the sensors in use are opened. When the duty cycle of those experiments is completed the flaps are closed. A more detailed study of the viewing requirements and view times may allow combining flaps over more than one instrument to reduce the amount of actuation hardware.

Access to the experiments within the planetary scan platform is achieved by removing the appropriate detachable panels.

A weight statement showing the estimated weights for the planetary scan platform mounted experiments, photo-imaging equipment, and planetary scan platform for the hypothetical payload described above is given in Appendix G.

3.3 ELECTRICAL INTEGRATION

3.3.1 Science Data Handling

3.3.1.1 Summary

The conceptual design of the data handling system assumes that science data will be conditioned within experiments and subsequently provided to the data handling subsystem. This approach permits maximum flexibility in the choice of experiments and data format; experiments designed subsequent to the spacecraft can be accommodated with minor data handling subsystem design changes. The basic functions to be performed by the data handling subsystem in regard to science data are:

- Format and record science data for transmission at a later time as the normal operating mode
- Format and transmit real-time science data as an alternate operating mode

The photo-imaging data is serviced by the data handling subsystem independent of all other science data because of the high data rates and data bulk associated with the photo-imaging system. Data for all other science experiments is time multiplexed within the data handling subsystem

to provide maximum flexibility and efficient utilization of the telemetry link. Generally all individual experiment data requirements (resolution) can be satisfied by virtue of selectively assigning experiment input data channels to the data handling subsystem and the capability to change telemetry format by command with a flexible format generator.

The following subsections include detailed discussions of:

- Science data requirements
- Science data handling interface
- Science data formatting

3. 3. 1. 2 Science Data Requirements

A telemetry list for the hypothetical payload is contained in Table 1, Appendix B. The list is considered to be representative of those experiments listed. The total number of data channels including primary science and engineering required for the core experiments is 53 analog and eight digital.

The total for the hypothetical all-TV photo-imaging system is 62 analog, 15 digital and 33 discrete. The core experiment data channels will be serviced independent of the photo-imaging data by the data handling subsystem. (See Section 3. 3. 1. 3 and 3. 3. 1. 4).

A summary of orbital primary science data estimates for the hypothetical payload is contained in Table 3-2. The data acquisition assumptions are consistent with the science time lines for a noneclipse orbit discussed in Section 3. 3. 2. 2. The buffered data rates represent reasonable demands based upon expected scientific requirements alone.

3. 3. 1. 3 Science Data Handling Interface

To provide maximum flexibility in handling science data, it is desirable to have scientific experiments provide preconditioned data to the data handling subsystem in either analog or serial digital nonreturn to zero form, or possibly a combination of both. Experiments providing analog data are those in which some parameter (typically a voltage) varies as a function of time, and the value of this parameter at each instant of time is the information to be conveyed. Analogs of temperature, internal



Table 3-2. Orbital Science Data Estimate

<u>Instruments</u>	<u>Data Acquisition Assumptions</u>	<u>Buffered Data Rate (bits/sec)</u>	<u>Total Bits/Orbit</u>
<u>Core Experiments</u>			
High-resolution IR spectrometer	Data required continuously when spacecraft within ± 60 deg of periapsis. (Approx. 0.8 hr/orbit)	150	4.2×10^5
Broadband IR spectrometer	Data required continuously when spacecraft within ± 60 deg of periapsis.	800	2.4×10^6
IR radiometer	Data required continuously when spacecraft within ± 60 deg of periapsis.	2300	6.44×10^6
UV spectrometer	Data required continuously when spacecraft within ± 60 deg of periapsis.	2500	7.0×10^6
Total - Core Experiments			1.63×10^7
		<u>Maximum Frames/Orbit</u>	
<u>Photo-imaging System*</u>			
Low-Resolution TV Camera	Data required ± 60 deg around periapsis.	33	2.83×10^8

* A total of four tape recorders will be required for all-TV photo-imaging system; one recorder for each of the three medium-resolution cameras and one recorder that will be switched between the low-and high-resolution TV cameras. (See Section 3.3.1.4)

Table 3-2. Orbital Science Data Estimate (Continued)

<u>Instruments</u>	<u>Data Acquisition Assumptions</u>	<u>Maximum Frames/Orbit</u>	<u>Total Bits/Orbit</u>
Medium-Resolution TV Camera	Data required ± 60 deg around periapsis	9/ea	20.8×10^8
High-Resolution TV Camera	Data required ± 60 deg around periapsis	18	6.17×10^8
			29.8×10^8

experiment operating voltages, and current are in this category. When handling an analog signal by a time multiplexed telemetry system, it is necessary to sample the function at regular intervals of time and then convert the magnitude of each sample to digital form. The analog output voltage signal can be expected to lie within the nominal range of 0 to +5.1 volts. The experiment output impedance should be sufficiently low (less than 10 k ohms) to minimize errors in quantization from leakage currents in the analog transmission gates.

Experiments providing digital data are those in which data occurs as a sequence of impulses occurring at either regular or random rates. The information that is desired may be the number of impulses per unit time or a count related to the pulse height of the output of a detector. Energetic particle and micrometeoroid counters are typical of this class. The sampling rate required for digital words depends upon the word size and the frequency of occurrence of impulses. When pulses occur very frequently, the sampling rate must also be high or the word may overflow, causing ambiguity. At the other extreme, little is to be gained by sampling more often than the time required for the word to change.

In order for experiments providing digital data to interface properly with the data handling subsystem, each experiment should provide at least one shift register as part of the experiment equipment. A shift



register within the experiment is a set of possibly eight storage elements which may accept the eight bit digital experiment value in either of two ways:

- The eight bit storage elements may be filled simultaneously (in parallel) upon receipt of a signal from the data handling subsystem
- The shift register may be used to accumulate pulses or counts in serial fashion between interrogations and then be inhibited during interrogation by the data handling subsystem

In either case, the shift register will be required to shift out its contents, one bit at a time, during one word time, upon command from and at the rate specified by the data handling subsystem.

Generally, a separate set of three lines, two inputs and one output, will be used for each shift register contained in an experiment. A shift register read enable signal will be provided by the data handling subsystem. This signal is a level defining the period of time during which the shifting out of data is occurring. The second input signal provided by the data handling subsystem is a clock pulse signal consisting of a continuous series of pulses at the bit rate in use by the data handling subsystem. The information output signal carries the data from the shift register to the data handling subsystem in nonreturn to zero form.

In addition to the enable and clock pulses, the data handling subsystem will provide synchronization pulses that can be utilized by experiments if required. The synchronization pulses will be short duration pulses in the order of five volts in amplitude and 10 msec in width. The pulses will occur on separate lines and will be coincident with the following:

- The start of each main frame and selected multiples of main frame
- The start of each subcommutator frame

The manner in which experiments may utilize synchronization signals would be to accomplish one or more of the following:

- Control internal operational sequences (e. g. , step a mirror, advance a sweep)

- Initiate automatic calibration sequences as a regular part of each data cycle
- Synchronize internal experiment data cycles with the telemetry rate

3.3.1.4 Data Formatting

Figure 3-5 illustrates the basic interface between the science experiments, including the photo-imaging system, the planetary scan platform and the data handling subsystem. Contained within the planetary scan platform are two multiplexers, one under control of the flexible format generator (FFG) for sampling science data, and the other multiplexer for sampling and forming photo-imaging subsystem data. (The rationale for providing remote multiplexers is developed in Section 3.3.3).

Science Multiplexer. The planetary scan platform remote science multiplexer contains 64 analog inputs and 16 digital inputs versus the requirement of 53 and 8 respectively as specified in 3.3.1.2. The remote multiplexer is addressed by the FFG to establish the sequence of sampling of the individual experiment data channels. The sequence stored in the FFG can be altered by command.

Any of the 64 analog inputs are selected by the analog multiplexer and digitized by the analog-to-digital converter (A/D). The output of the A/D is mixed with the sampled digital data and sent to the main carrier combiner. Enable signals are provided by the digital multiplexer to gate eight bit groups of science digital data to the multiplexer. The clock signal from the data handling subsystem shifts the digital data from the selected experiment output register at the data handling clock rate. In addition to the enable and clock signals, the data handling subsystem provides frame rate and subframe rate pulses that can be used by experiments as required for synchronizing internal experiment logic with the data handling rate.

Figure 3-6 illustrates a typical science experiment data format. The sources selected for any inputs in the main frame or subcommutator except the fixed words is reprogrammable in the flexible format generator by command.

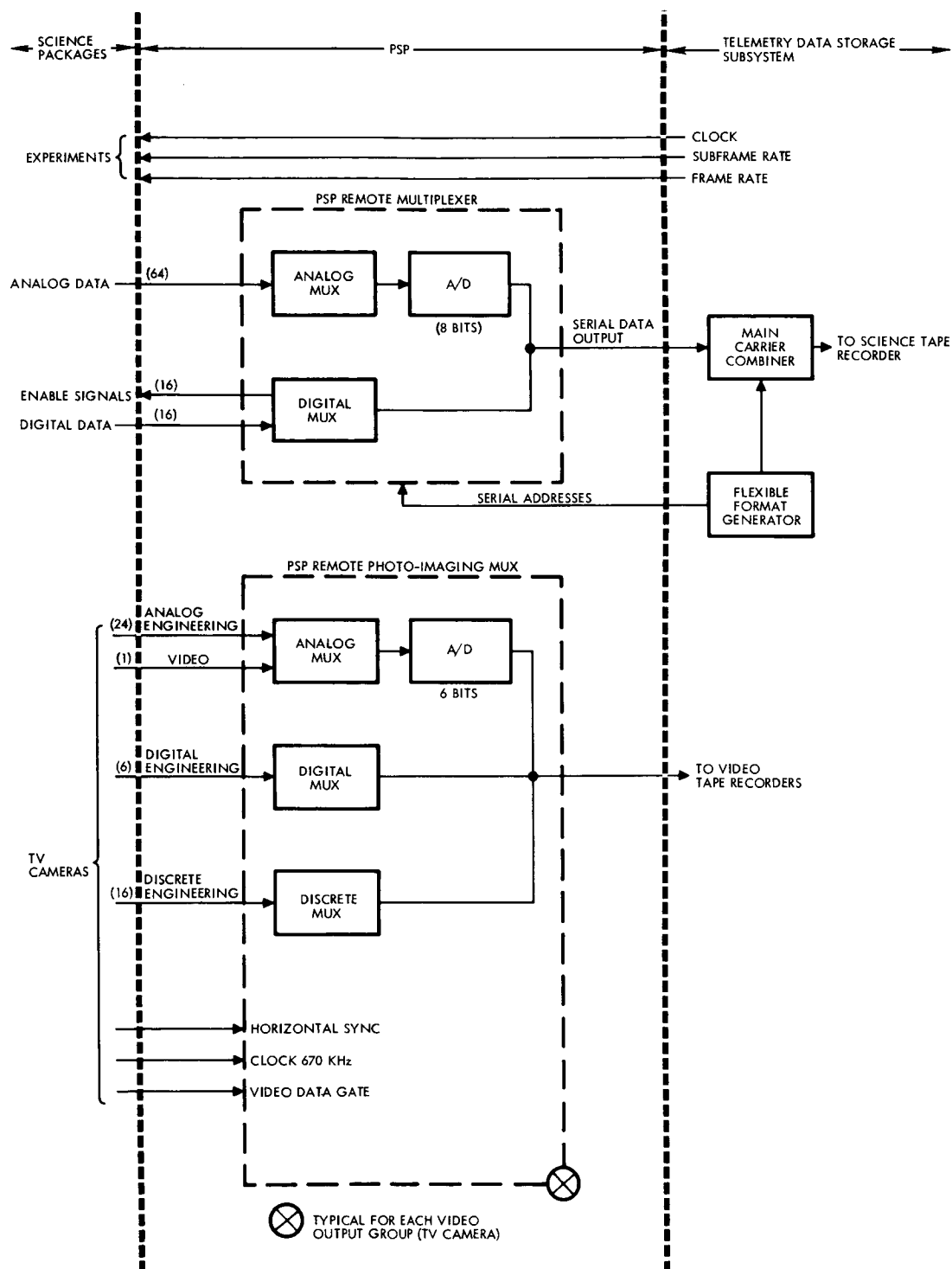


Figure 3-5

SCIENCE DATA for PSP-mounted equipment interfaces with the spacecraft telemetry and data storage subsystems through remote multiplexers on the PSP

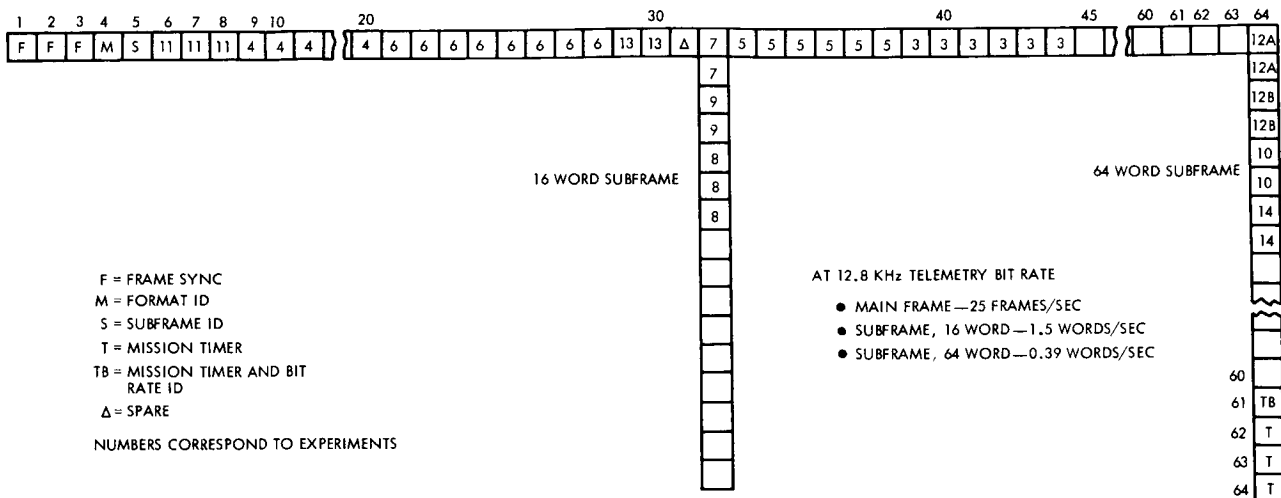


Figure 3-6

TYPICAL SCIENCE DATA FORMAT consists of a 64-word main frame, a 16-word subcomm, and a 64-word subcomm. Seven main frame words carry frame sync, format ID, subcomm ID, and the two subcomm values. The mission timer and bit rate ID are in the 64-word subcomm since they can be sampled at low rate. Thus 57 main frame words are used for multiplexing high rate science data.

The format shows a sampling sequence for the experiments in the hypothetical science payload that will satisfy the individual experiment data rates that were assumed for primary science data. Science engineering data is not shown but could be assigned to the subcommutators because the sampling rates for these functions can be rather low. Many words within the main frame are not assigned for the given format, which allows the capability to either telemeter more science functions or increase the data rates (resolution) of the given experiments.

3.3.1.5 Photo-imaging Multiplexer

The video data, within the planetary scan platform, is sampled and formatted by the remote photo-imaging multiplexer. Only one channel is shown on Figure 3-5. However, one such channel is required for each TV camera.



The multiplexer samples camera engineering data and enters it at the end of each picture data line as well as between pictures along with the required telemetry frame sync and mode data.

Video analog data and camera engineering analog data is encoded to six bits and combined with sampled digital and discrete camera engineering data to form a serial digital bit stream which goes to the video tape recorders and later playback. The photo-imaging subsystem provides the remote photo-imaging multiplexer with clock, horizontal sync and a video enable signal to synchronize data sampling and formatting.

The configuration of the data recording equipment for the TV photo-imaging system requires four tape recorders to accept the digitally encoded video data from the five cameras. Through the use of a redundancy switch, the recorders may be interchanged to accommodate a particular orbital recording sequence. Each of the four recorders will have a total data capacity of 9×10^8 bits, using a tape speed of 50 ips. Data will be accepted at the nominal rate of 700 kb/sec. This data stream will then be converted from serial to parallel form and recorded on six parallel tracks at the rate of 116 kb/sec per track, with a recording density of 2300 bits/inch. Each recorder has a total tape length of 3000 feet. After filling one length of tape with six parallel tracks of video information, the tape may be reversed to permit utilization of an additional six parallel tracks.

The maximum data which may be obtained in one orbit by the low-resolution television camera (2.8×10^8 bits) will utilize one-third the capacity of one tape recorder. However, the data obtained by one of the three return beam vidicon cameras (9 frames over a ground track of 4480 km, ± 37.5 deg from periapsis) completely fills one recorder including the usage of additional tape to accommodate the increasing dead time between recording of adjacent pictures as the altitude of the orbit increases. Thus three recorders are required to permit the recording of 27 adjacent frames (a total of 2.1×10^9 bits of data). The high-resolution camera will not provide continuous coverage, as time must be permitted for data recording between consecutive frames. Utilizing one tape recorder for storage of high-resolution imaging data, as many as 18 frames of data may

be obtained during one orbit. If the locations of the selected areas of which high-resolution imaging is desired are widely spaced, this recorder may be stopped between consecutive recordings to conserve tape.

3.3.2 Sequencing

3.3.2.1 Summary

The Voyager spacecraft is designed for maximum flexibility in the choice of experiments; additional or alternative science command and sequencing can be accommodated as required. The basic functions to be performed in regard to science command and sequencing are:

- Provide signals to switch main power ON and OFF to experiments
- Provide signals to switch power and execution signals for science pyrotechnic devices, if required
- Provide signals to initiate mode sequences, calibration, mode changes, etc. to experiments
- Initiate experiment commands in the normal mode referenced to orbital parameters such as morning and evening terminator crossings and periods around periapsis, apoapsis, etc.
- Initiate experiment commands during the occurrence of special events (e. g., solar flares) referenced to the special event with priority over all normal experiment programmed sequences. A preprogrammed experiment sequence will be defined for all special science events.

All science commands will be time sequenced by the computing and sequencing subsystem. Backup command and sequencing capability will be provided by the backup computer and sequencer. Individual stored science program commands with associate time tags within the computer and sequencing memory can be updated or altered by ground command.

The following subsections include discussions of:

- Science command requirements
- Science power requirements
- Typical science time lines



3.3.2.2 Science Command Requirements

A list of commands exclusively for the operation and control of experiments is contained in Table 1, Appendix A for the hypothetical payload. The command list represents an estimate of reasonable demands typical of those experiments listed.

The total number of commands for the hypothetical payload is 10 power commands and 103 discrete commands. A power command is defined as the switching of DC primary power to a given experiment where the switching is accomplished external to the experiment. A minimum of one power command will be required for each experiment. Additional power commands may be required based upon detail individual experiment requirements. A discrete command is defined as a short duration pulse in the order of 10 volts in amplitude and 10 msec in width. Commands such as these will be generated within the computer and sequencer subsystem and distributed to experiments to initiate specific functions such as mode changes, calibration sequences, etc.

Figure 3-7 illustrates the basic interface with science experiments in regard to command and sequencing. Contained within the planetary scan platform are redundant decoders and science power switching assemblies. (The rationale for the remote equipment is developed in Section 3.3.3.)

A total of 123 discrete commands are required at the output of the remote decoder. Twenty discrete commands are required for control of the switching of power (ON/OFF) within the remote science switching assembly for individual experiments. A total of 103 discrete commands are required directly by the individual experiments. The total number of 123 discrete commands are implemented with approximately nine wires interfacing between the remote decoder and the computer and sequencer subsystem.

The distribution of primary science power from the power subsystem is accomplished within the remote power switching assembly and the interconnect cable to experiments. A power switch for each power line required by experiments will be contained within the remote science

switching assembly. The switch will be controlled (command ON or OFF) by discrete commands from the remote decoder.

3.3.2.3 Science Power Requirements

Power estimates for the hypothetical payload are contained in Table 1 and Table 2, Appendix C, for an eclipse and a non-eclipse orbit respectively. The determinations are based upon the following:

- A 13.8-hr nominal orbit
- Total desirable experiment operating periods as specified in Section 3.3.2.4

The tables contain the peak power, duty cycle, and average power for each of the experiments and the science data tape recorders.

The total average power required for the hypothetical payload operations is 55.43 watts for an eclipse orbit and 34.60 watts for a noneclipse orbit.

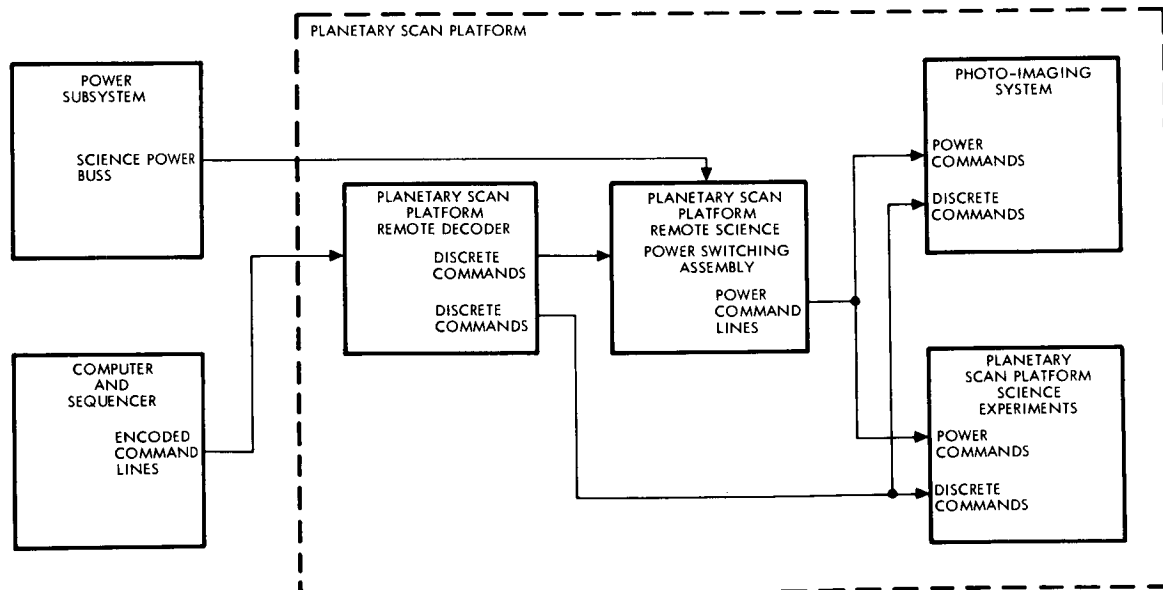


Figure 3-7
POWER AND COMMAND INTERFACE connections for mounted experiments and photo-imaging equipment are made through remote decoders and power switching assemblies.



3. 3. 2. 4 Science Time Lines

Figures 3-8 and 3-9 represent typical experiment operating periods (data acquisition periods) for an eclipse and a noneclipse orbit respectively for the hypothetical payload. The operations represent expected reasonable demands based upon anticipated scientific requirements alone. As experiments become more completely defined and as specific orbital conditions are identified, the experiment operating periods can be more exactly specified.

In addition to illustrating anticipated science operations as a function of altitude and true anomaly, also shown are the associated science data tape recorder operations and the composite science power profile including the tape recorders for each orbit. The power profiles are consistent with the individual experiment peak power requirements specified in Tables 1 and 2, Appendix C.

The orbital conditions assumed in regard to the time lines are:

- A 13.8-hr nominal orbit with periapsis of 1000 km altitude and apoapsis of 20,000 km
- An orbit inclination of 45 degrees with evening and morning terminator points occurring at true anomalies of approximately 30 and 210 degrees, respectively. This is assumed to correspond to conditions shortly after injection for an orbit with periapsis over northern latitudes.
- Time scales are measured from periapsis since this is the most favorable position for the surface mapping and atmospheric analysis experiments.

Following are general assumptions associated with science requirements in regard to the time line for the noneclipse orbit shown in Figure 3-9.

Photo-imaging. Mapping will be accomplished around the periapsis region with altitudes varying from 3000 km on each side of periapsis to 1000 km at periapsis for the nominal orbit. If photo-imaging is performed within approximately ± 76 degrees of periapsis then operation will be approximately 1.1 hours total per orbit.

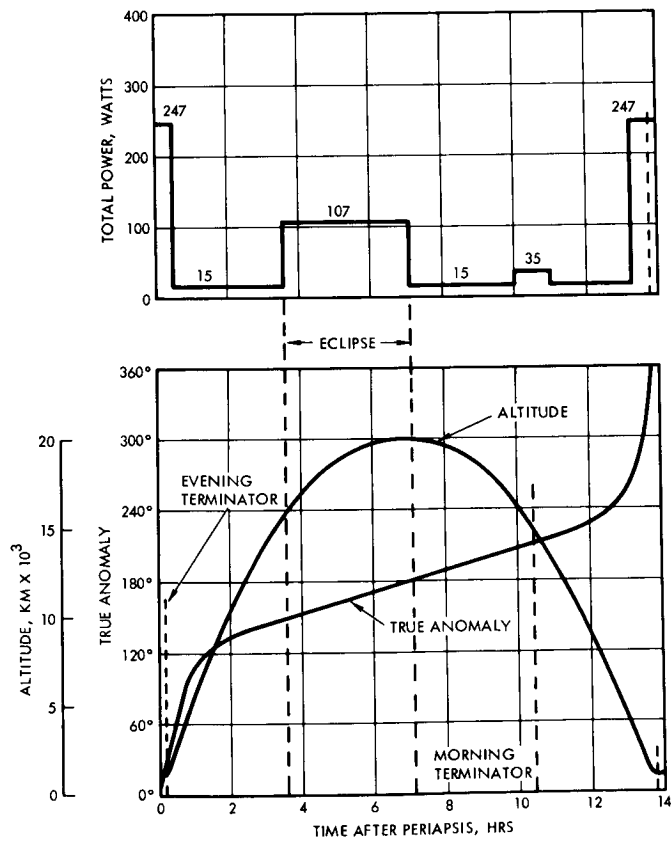
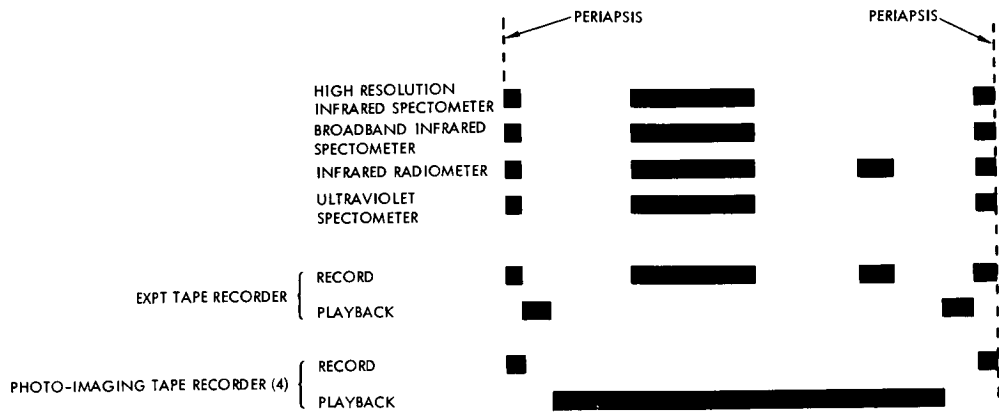


Figure 3-8
ECLIPSE ORBIT POWER PROFILES for the hypothetical science payload show additional operation of core experiments throughout eclipse period as well as around periaapsis.

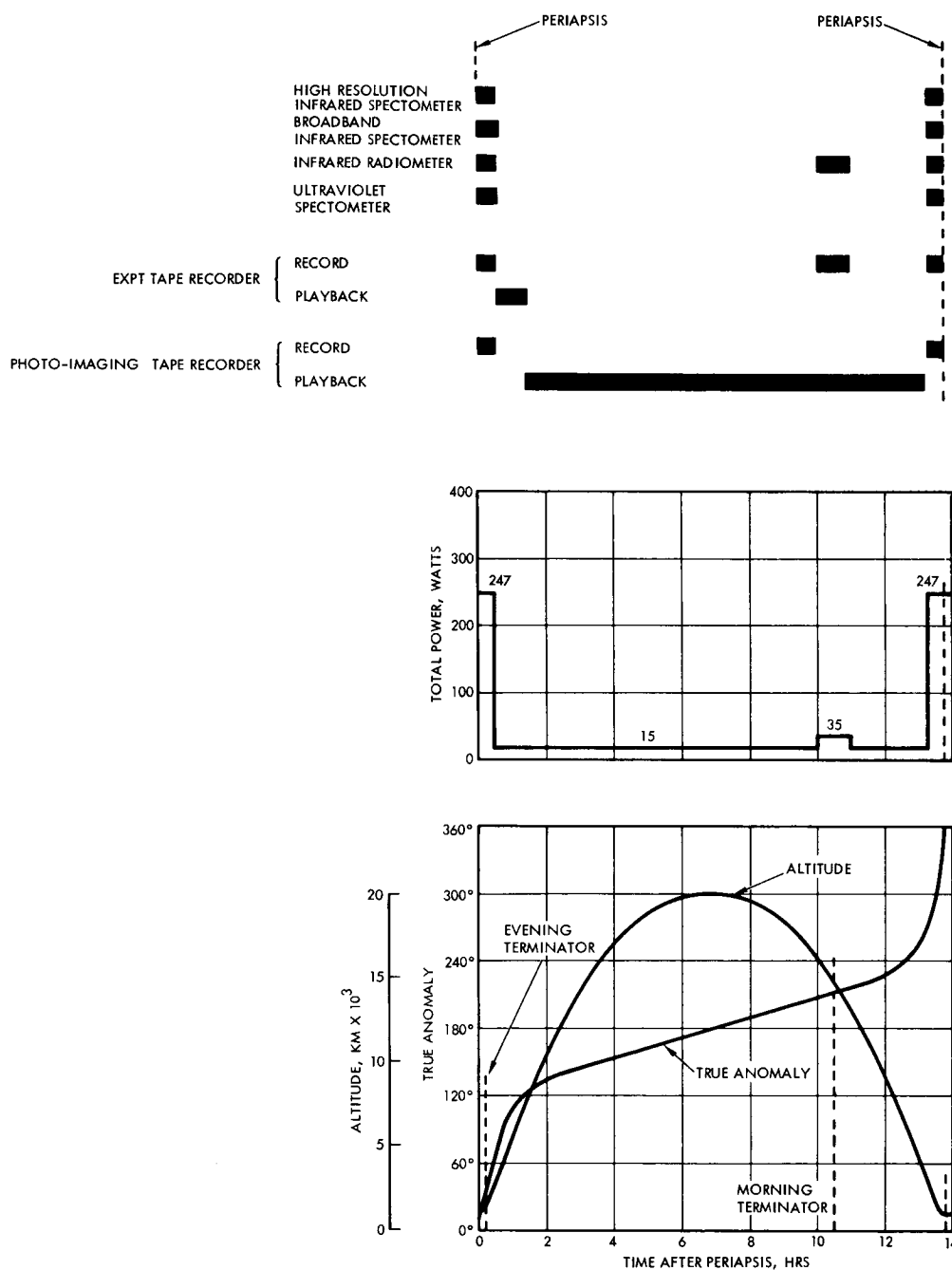


Figure 3-9

NONECLIPSE ORBIT POWER PROFILES for the hypothetical science payload show operation of most science experiments around the periapsis region. Playback of data occurs throughout the remainder of the orbit.

Infrared and Ultraviolet Spectrometers. The spectrometer experiments are primarily for atmospheric analysis and secondarily for cloud and temperature determination and mapping. Operation has therefore been restricted to orbital altitudes less than 3000 km (± 76 degrees about periapsis), corresponding to an operating time of 1.1 hours per orbit. The remainder of the orbit should, if possible, be available for optional operation, either intermittently or continuously, at the discretion of the principal investigators and on a time-sharing basis between the several experiments when necessary. The ultraviolet spectrometer should also observe each terminator crossing and another one hour operating period is scheduled around the morning terminator. (This may also be desirable for the infrared spectrometers, but is deemed less likely and should be adequately covered by the optional operation possibility previously described.) If the orbit is such that both terminators are outside the ± 76 degree periapsoidal region, the ultraviolet spectrometer may have one hour operation periods around each terminator crossing as well as the 1.1 hour periapsoidal period.

Infrared Radiometer. The infrared radiometer is primarily a mapping experiment. Hence, only periapsoidal operation is included. It is thought that temperature determination across the terminator is adequately provided for by the times, as shown, when the terminator is in the periapsoidal region; similarly, the dark-side temperature mapping can be done at those times when periapsis falls on the dark side. Hence, only periapsoidal operation is shown for this instrument.

Following are general assumptions associated with science requirements in regard to the time line for the eclipse orbit shown in Figure 3-8.

Infrared and Ultraviolet Spectrometers, Infrared Radiometer. The experiments are shown as operating slightly before, throughout, and slightly after the eclipse, since the time-histories of temperature and atmospheric properties are of great importance. It has been assumed that shutters are provided for these optical instruments so that they can safely look directly at the sun just before and after the eclipse.



3.3.3 Electrical Integration Design

An electrical interface chart for the hypothetical payload is contained in Table 1, Appendix D.

The numbers of electrical signals required by individual experiments is specified and is considered to be representative of expected demands for those experiments identified. The detailed command and telemetry functions for each experiment are contained in Table 1, Appendix A, and Table 1, Appendix B respectively.

The total number of wires required to service the hypothetical payload contained within the planetary scan platform is derived in Table 1, Appendix E. Approximately 374 wires will be required, excluding wires for planetary scan platform support equipment such as the Mars sensor, gimbal drive, gimbal angle encoders, and thermal control heaters.

Figure 1-2 is the general block diagram illustrating the basic electrical interfaces between the hypothetical payload and the spacecraft subsystems. A remote command decoder, science switching assembly, and data multiplexers are included within the planetary scan platform. By virtue of the planetary scan platform remote equipment approximately 30 wires without redundancy will be required in the planetary scan platform cable wrap-up through which connections are made to the spacecraft support subsystems for science power, command, and telemetry. Approximately 374 wires excluding redundancy would be required if experiments were connected directly to spacecraft subsystems through the planetary scan platform wrap-up.

With remote decoding, switching, and multiplexing equipment in the planetary scan platform, maximum flexibility exists in the choice of experiments designed subsequent to the spacecraft. This configuration allows for standardization of interconnect cables between spacecraft subsystems with little regard for individual experiment interfaces. The individual experiment electrical interface requirements can subsequently be satisfied by the design of the interconnect cable between the experiments and the remote equipment contained within the planetary scan platform.

3.4 PLANETARY SCAN PLATFORM THERMAL CONTROL

Temperature control for the planetary scan platform and its science equipment is accomplished by the following state-of-the-art, flight proven items: insulation, radiators, heaters and thermostats, thermal coatings, and thermal coupling. The broadband infrared spectrometer detector requires a combination of radiation cooling and cryogenic refrigeration. The temperature control system proposed for the planetary scan platform and all its science payload will maintain the instruments within their temperature limits as presented in Table 3-4 during all phases of the Voyager mission.

3.4.1 Equipment Mounting

The science equipment is arranged in the planetary scan platform to distribute the heat load. Exceptions to this are the IR spectrometers. The broadband IR spectrometer (Figure 3-10) requires two radiating surfaces and a viewing surface, and therefore is placed in a corner. The high-resolution IR spectrometer (Figure 3-10) requires one radiating surface and one viewing surface. Both IR spectrometers are mounted on fiberglass standoffs and attached to the planetary scan platform structure.

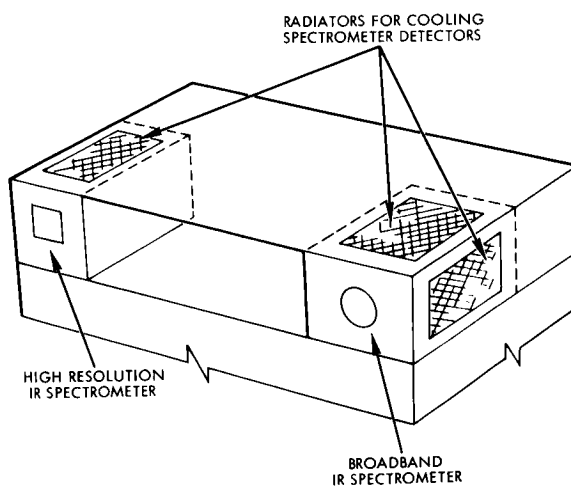


Figure 3-10

THREE RADIATORS on the PSP cool the broadband and high resolution IR spectrometers. A cryogen refrigerator provides additional cooling for a broadband detector.



Table 3-4. Planetary Scan Platform Equipment Temperature Limits

Equipment	Allowable Temperature, °F	
	Minimum	Maximum
Photo-imaging (all TV system)	35	85
High-resolution infrared spectrometer		
Spectrometer	- 76	50
Detector	- 76	- 76
Electronics	- 22	122
Broadband infrared spectrometer		
Telescope	- 44	- 26
Manochromator	- 44	- 26
Channel 1 chopper	-235	-217
Channel 1 detector	-379	-379
Channel 2 chopper	- 46	- 28
Channel 2 detector	-226	-226
Electronics	- 40	104
Infrared radiometer	4	58
Ultraviolet spectrometer	32	104
Mars sensor	0	113

The multilayer insulation on the broadband infrared spectrometer and the high-resolution infrared spectrometer is attached with the mylar side out, having an emissivity of 0.78. The remaining instruments and electronics units are to be coated with Cat-a-lac black paint having an emissivity of 0.86.

3.4.2 Photo-Imaging System

The power requirements for the operation of the cameras in the photo-imaging systems are such that each camera will be mounted directly on the planetary scan platform structure to enable the heat to be dissipated during operation. The hypothetical all-TV photo-imaging system requires an average of 11 watts of power while in Mars orbit. Each camera dissipates nearly equal amounts of heat (Table 3-5), and therefore, no special mounting is required.

3.4.3 Insulation

One inch of 70 layers-per-inch aluminized Mylar is to be attached to the internal facing surfaces of the IR spectrometers, providing maximum isolation of the components of these two instruments from all other planetary scan platform mounted equipment. The maximum conductance between the outer layer and inner layer when attached is 0.001 Btu/hr-ft²-°F, based on a hot side temperature of 85°F and a cold side temperature of -315°F. Velcro tape attaches the insulation to satisfy removal and replacement requirements. A more comprehensive discussion of the insulation design development and parameters affecting the design is included in Section 3.5 of Volume 10.

Table 3-5. Average Power and Duty Cycle For
Photo-Imaging System (all TV)

System	Duty Cycle (%)	Average Power (w)
1 low-resolution color TV	5.8	1.45
3 medium-resolution TV	5.8	8.70
1 high-resolution TV	5.8	1.16
Total		11.31



3.4.4 Radiators

The IR spectrometers contain detector components which must be radiatively cooled. Two eight and one half by ten inch radiators are used to control the temperatures of the broadband IR spectrometer telescope, monochromator, choppers, and detectors. The high-resolution IR spectrometer detector is cooled by a 7 x 13 inch radiator.

The radiators are three-sixteenths inch aluminum plates framed with fiberglass (Figure 3-11), which also provide adequate micrometeoroid protection. The external surface of the radiator is coated with IITRI Z-93 white paint with a hemispherical emittance of 0.9 and a solar absorptivity of 0.18; after 1000 hours of test exposure to one ultraviolet sun at a pressure of less than 10^{-6} torr the solar absorptivity was 0.25.

The detector components to be cooled are conductively tied to the radiators using high conductivity material. The inner surface of the radiator is insulated to provide isolation of those components which do not require cooling.

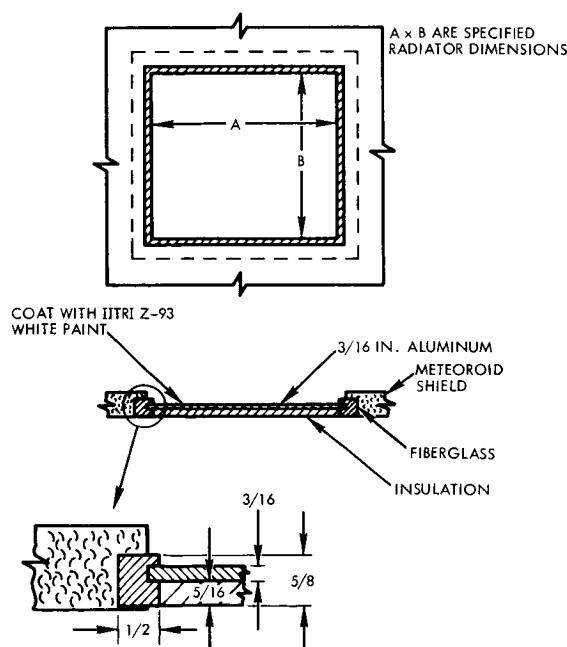


Figure 3-11

SPECTROMETER RADIATORS are fabricated from 3/16 inch aluminum sheets coated with white paint and insulated from other science experiments within the PSP box.

3. 4. 5 Cryogenic Refrigeration

The broadband IR spectrometer contains an infrared detector which must be maintained at 45°K . A radiator operating at this temperature having an emissivity of 0.9 can remove 0.02 watts per square foot of radiating area. To radiatively cool the detector then would require the radiator and the detector to be perfectly isolated from any heat sources, an impractical solution. However, a radiator used in conjunction with a refrigeration method can be effective.

There are four basic methods for spacecraft refrigeration: stored cryogenic fluids, thermoelectricity, sublimation, and mechanical refrigeration. Storing cryogenic fluids for extensive periods is not practical since the weight of the initial fluid required considering the loss due to boil-off would be prohibitive. Thermoelectric cooling at its present state of the art is only applicable to temperatures above 150°K , and thermomagnetic cooling is still under development. Present developments in mechanical and solid cryogen refrigeration appear to be promising, and should eventually lead to the development of several miniature cryogenic refrigerators appropriate for space applications.

The Voyager mission design time is two years including six months in Mars orbit. This length of time, of course, rules out any type of open cycle cooling. A closed cycle refrigeration system is, therefore, proposed to meet the 45°K requirement for the broadband spectrometer. A miniature cryogenic refrigerator suitable for immediate spacecraft use has been developed by NORELCO called the "Cryogem" which operates on the Stirling cycle. A "Cryogem" unit has been developed to operate with 38 volt dc instead of the more usual 3 phase 400 cycle and 60 cycle power. The current 28 volt model will pump 10 watts at 70°K with 300 watts input power and 1.5 watts at 20°K with 400 watts input. The system requires 400 watts input at turn-on and requires about 10 minutes to reach 70°K . For 45°K application 350 watts of input will pump 5 watts. The "Cryogem" unit is applicable for use in the broadband spectrometer, but it requires a large amount of power, and therefore poses an additional problem for the spacecraft power system.

The efficiency of all these refrigeration cycles is quite low, requiring from 0.1 to 2 kilowatts of power for 0.1 to 5 watts of



refrigeration, producing a system that is quite heavy when the power supply weight is included. At the present time, several approaches are being investigated for cryogenic refrigerators, including electrically powered refrigeration cycles and gas-powered systems. Three different refrigeration cycles (Stirling, Joule-Thomson, and Claude cycles) and their variations are being considered as possible candidates for miniature cryogenic refrigeration and much research is being carried out in the development of high-speed miniature compressors. As a result of this research and development activity, it is felt that miniature cryogenic refrigerators could be available in the next few years, which will require no more than 50 watts of input power and meet the 45°K cooling requirement.

In addition, solid cryogen refrigerator systems are currently under development. These consist of a container filled with a solid cryogen which is thermally coupled to an infrared detector, and thermally isolated from its surroundings. The cryogen to be used is determined by the temperature at which the infrared detector is to operate; temperature is maintained by controlling the vapor pressure over the solid. Development efforts have produced an Argon refrigerator which will maintain an infrared detector at 52°K and remove 17 milliwatts of heat from the detector; it has a life of one year. The complete cycle of cooling and solidification of the Argon required 24 hours. Further development of solid cryogen refrigerators for space application could result in an applicable system for use in the broadband infrared spectrometer. The cryogen would be solidified prior to launch and therefore require the solid cryogen to have a useful life in excess of one year.

3. 4. 6 Planetary Scan Platform Thermal Design

The planetary scan platform box structure also serves as micro-meteoroid shielding and thermal insulation. The outside panels are 2 inch thick aluminum honeycomb sandwich, foam filled. The micrometeoroid shield has a thermal conductivity of 0.02 Btu/hr-ft-°F. The micro-meteoroid shield provides sufficient insulation to maintain the planetary scan platform environment within instrument temperature limits.

The planetary scan platform support is insulated with multilayer insulation. To avoid freezing of the deployment mechanism and the drive

mechanism, a two watt thermostatically controlled heater is attached to each mechanism. The attachment of the support to the planetary scan platform and to the spacecraft is such as to inhibit heat flow.

The heaters to be used for temperature control are of the strip type. These are thin variable sized, flexible units which are adhesively-bonded to the surface. The resistive wire is bifilar-wound to reduce the magnetic field. With DC power applied the magnetic field does not exceed 2 gamma at 2 inches. The unit can be obtained from commercial sources per TRW Specification PT4-13004 in virtually any physical size and power rating required.

Associated with the heaters are thermostats having an OFF-ON range of 10°F. The heaters turn on when the temperature of the component to be heated is 10°F above its lower limit, and turns off when the component is 20°F above its lower limit.

Additional information on the thermal control area is included in Section 5.4.2 of Volume 3.

3.5 PLANETARY SCAN PLATFORM DESIGN

The planetary scan platform illustrated in Figure 3-12 is made up of the following principal groups of equipment: structural shafts and yoke; the equipment mounting platform and box; gimbal drive assemblies; deployment drives; drive electronics; and the Mars sensors. The planetary scan platform remote decoders, switching units, and data multiplexers, although contained in the planetary scan platform, are not considered part of the planetary scan platform equipments. For further definition of these remote units see Volume 4.

3.5.1 Structural and Mechanical Components

The planetary scan platform structure consists of the following components:

- Box structure
- Yoke structure
- Support struts
- Tiedown structure and release



The mechanical components include:

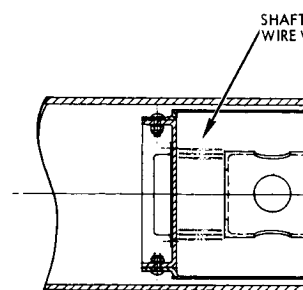
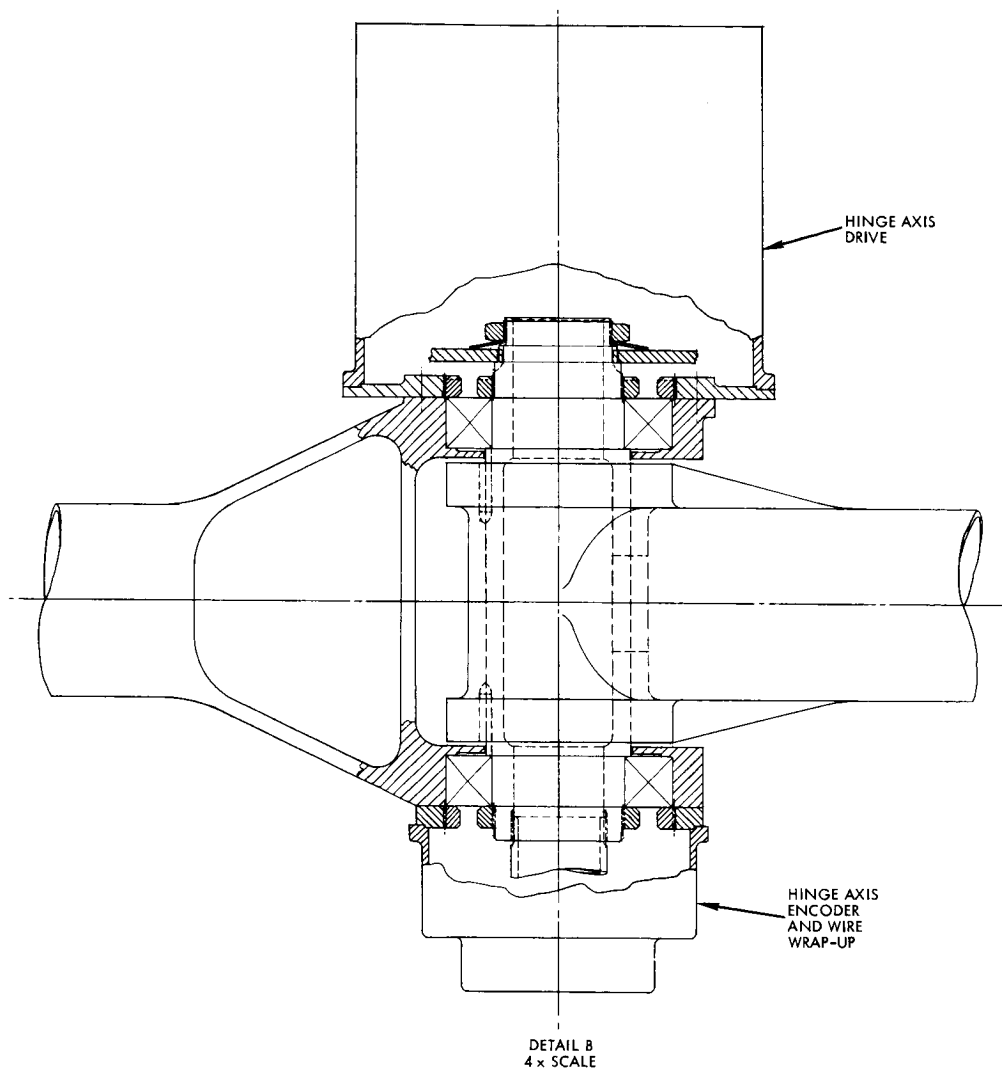
- Hinge axis drive
- Shaft axis drive
- Yoke axis drive
- Encoders
- Wire wrap mechanisms

3.5.1.1 Structural Components

Box Structure. The box structure houses all of the scientific and photo-imaging instruments. The outside walls serve as meteoroid protection panels as well as mounting panels where required. They are constructed of two inch thick aluminum honeycomb sandwich panels with the core filled with two pounds/ft³ foam. The face sheets are 7075-T6 aluminum alloy; the outer sheet is 0.010 inch thick and the inner is 0.030 inch thick. The inside panels are also honeycomb sandwich construction; sized for supporting the components of each configuration. All of the panels have "Z" section edge members which are bolted to 7075-T6 aluminum angles at the corners, to form the box, and along the inside of the outer panels to support the internal panels.

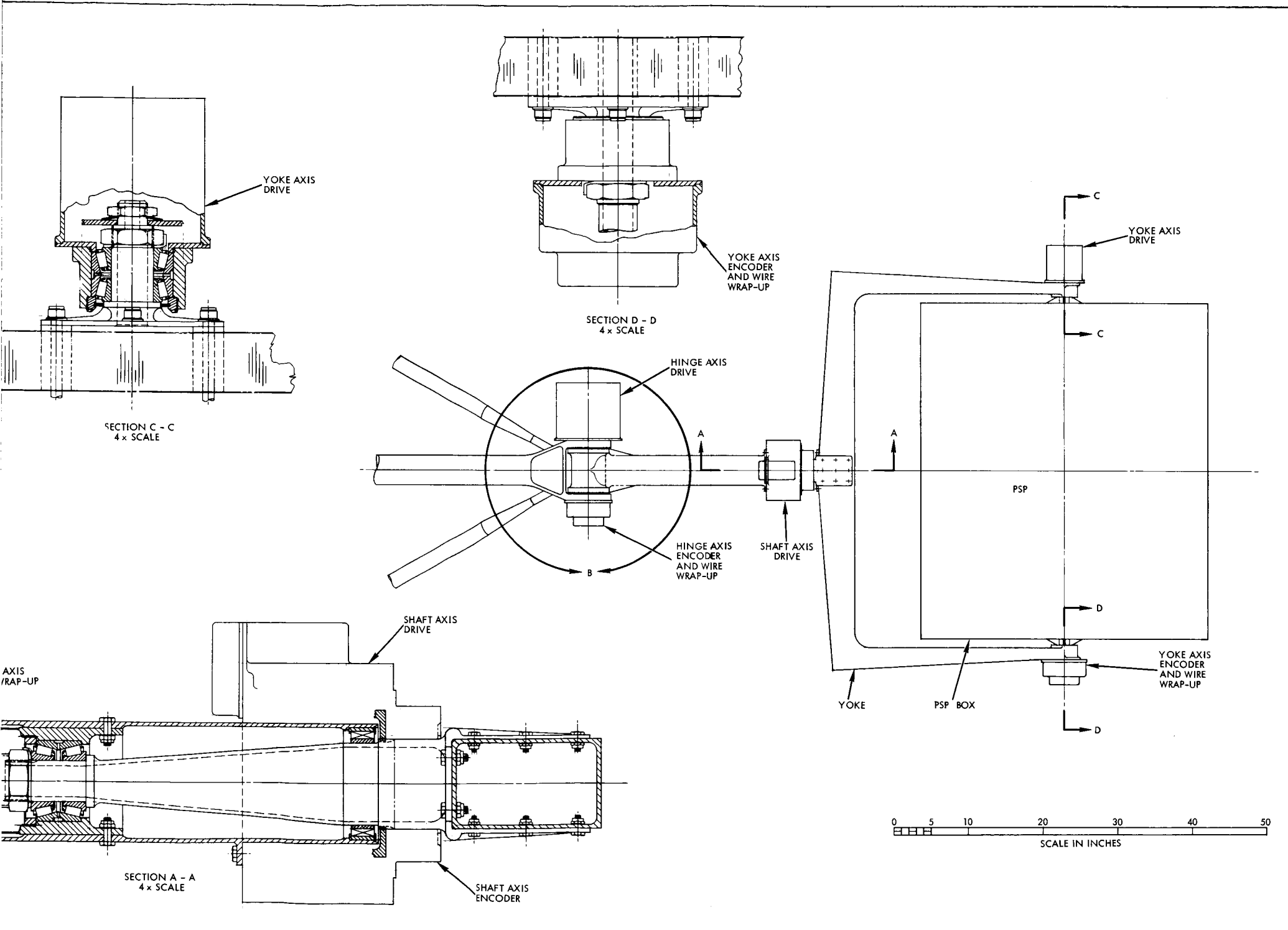
The box is supported on its centerline at each end with Titanium fittings which bolt to it and which serve as the shaft support as shown in Figure 3-12. The shaft rides in bearings in the end fittings of the yoke. The bearings are dry lubed to prevent cold welding or galling.

Support Struts. The planetary scan platform has three support struts which extend from the body of the spacecraft to the pivot axis of the yoke as shown in Figure 3-12. All struts are made of thin wall aluminum tubing welded to machined end fittings. The upper strut, which attaches to both the upper closeout panel and one of the upper side panels, is a 4 inch outside diameter x 0.035 wall 6061-T6 aluminum tube. It is sized to carry all bending and torsion loads induced when the planetary scan platform is extended during the orbit insertion maneuver, with the capsule off. This strut together with the other two struts form a tripod to carry the shear loads at the hinge axis. The three struts also carry the loads from



FOLDOUT FRAME

3-32



MECHANICAL ELEMENTS On the PSP consist of the hinge axis drive, shaft and shaft drive, yoke and yoke drive and main equipment mounting platform and box.

3-32A

3-32B / 3-33

FOLDOUT FRAME

the planetary scan platform box in the spacecraft longitudinal direction in the stowed position. The drive on the yoke hinge axis is bolted to a machined fitting on the end of the large strut. The other two struts attach to lugs on this same fitting. These struts are 3 inch outside diameter x 0.049 wall 6061-T6 aluminum tubes.

Yoke Structure. The yoke structure consists of the main support tube which extends from the hinge axis to the yoke axis drive and the yoke which supports the box. The main support tube is 4 x 0.095 inch wall aluminum tube with a machined fitting, which serves as the hinge hub, welded to it. The machined fitting at the outer end houses the bearings for the yoke spindle and provides a mounting flange to attach the drive as shown in Figure 3-12. The yoke is a welded assembly consisting of a turned spindle and a fork of two channels to form a closed box section. The fork tapers from 3 in.² at the box shaft to 5 x 3 inches at the intersection with the spindle.

Tie-down and Release System. The tie-down and release system is comprised of four bearing pads and two ordnance release devices. The release devices are the same as those used for all the other appendages as described in Volume 3. Each release is located at the centerline of the planetary scan platform box, midway between two pads as illustrated in Figure 3-13. Bathtub fittings on the side of the box provide for the attachment of the release yoke. The release device is mounted on a fitting which is supported by beam structure inside of the spacecraft equipment module.

The four pads have mating serrated plates on the planetary scan platform box and on the pad fittings. The fittings are mounted to backup structure inside the spacecraft. The release device preloads the box against the pads to react the shear and moment loads on the box. The longitudinal loads are reacted through the yoke hinge by the tripod struts.

3.5.1.2 Mechanical Components

The arrangement of the drives is shown on Figure 3-12. The drives shown are wobble gear drives similar to the antenna drives, as described in Volume 3. The drives are sized to carry the loads and torques imposed on them with the planetary scan platform extended during the orbit insertion

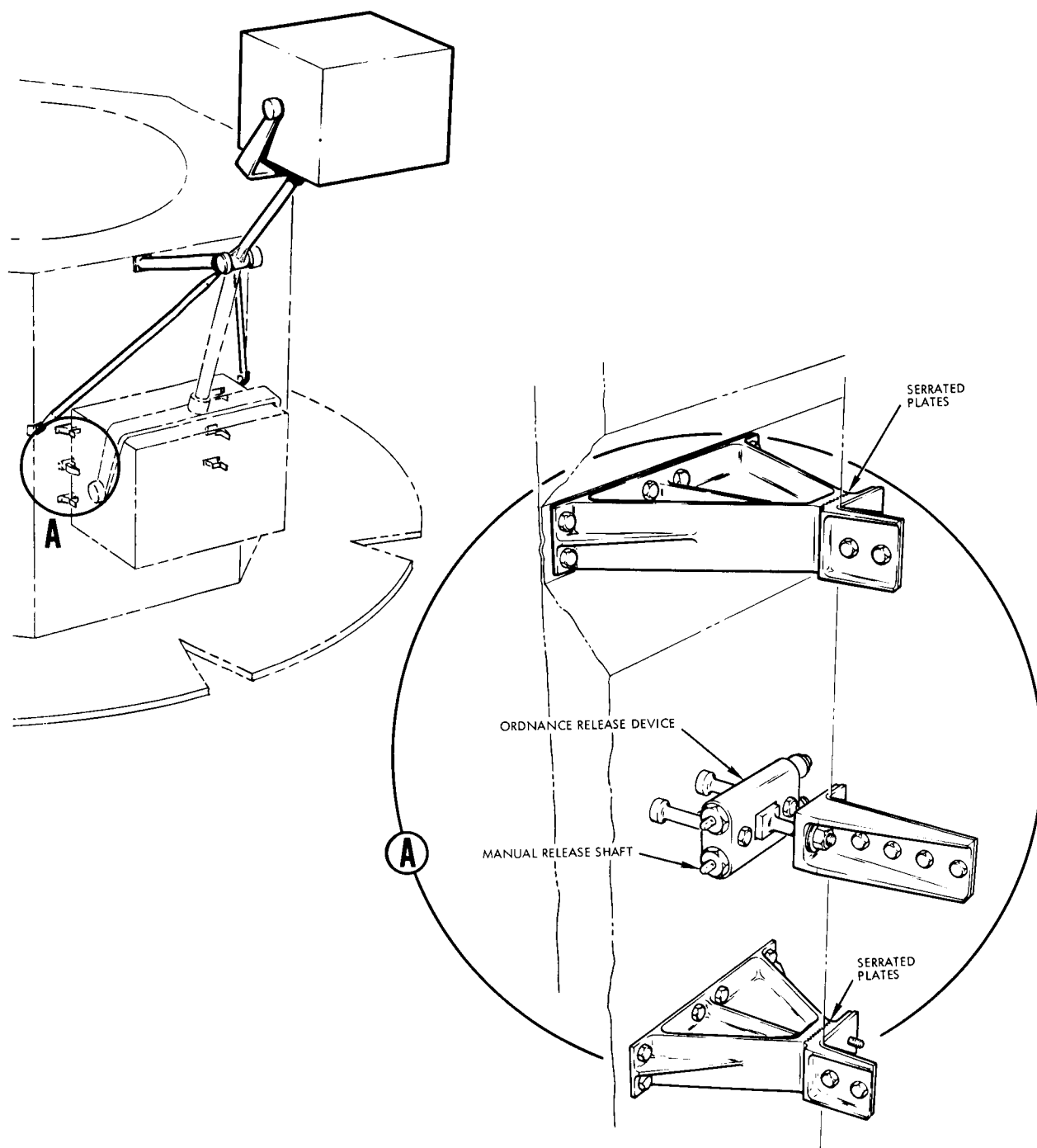


Figure 3-13

TIE DOWN AND RELEASE of the PSP is accomplished by an explosive ordnance device. The PSP is released for deployment after launch.



maneuver with the capsule off, which proved to be the critical loading condition. An estimated center-of-gravity offset of 10 inches from the geometrical center of the box was assumed to provide the torque requirements for the yoke and shaft drives.

Redundant encoders are provided on each drive axis for position feedback to the control electronics. Tachometers provide rate information.

Wire wrap-up mechanisms, like the ones developed for the Orbiting Geophysical Observatory (OGO), are used on each axis to rout the wires from the spacecraft to the planetary scan platform box. The angular travel is limited to ± 190 degrees at each axis to minimize the wire flexing in the wrap-up.

3.5.2 Planetary Scan Platform Pointing

The most severe pointing requirements for the planetary scan platform mounted equipments are those for the photo-imaging systems. All of the camera lenses will be boresighted together to allow correlation between low, medium, and high-resolution pictures. The other science experiments on the planetary scan platform have much less stringent requirements than the photo-imaging equipment and are aligned parallel to the camera axes.

Because there are different accuracy requirements for the two photo-imaging systems — all TV (pointing) and film/TV (image motion compensation), characteristics of both will be discussed.

The pointing and stability requirements for the Voyager photo-imaging systems are determined primarily by the optical fields of view and the desired values of resolution. During the photographic interval, the planetary scan platform must be driven with precise rate and position control. The planetary scan platform drive and its control system are required to provide a continuously variable speed to allow tracking of Mars and, at least for photographic film systems, to provide for image motion compensation. The drive should also allow slewing operations and pointing based upon commanded gimbal angles.

Two types of operation are envisioned, a) vertical photography, in which continuous low and/or medium-resolution photographs will be obtained, with high-resolution imaging at selected intervals, and b) off-vertical photography, in which low or medium-resolution stereo photographs may be desired, as well as selected high-resolution imaging, either monoscopic or stereo.

In vertical photography, two methods of pointing may be employed. The first consists of utilizing the Mars horizon tracker as a primary reference in a closed loop mode, with the planetary scan platform servo system utilizing the output signal of the horizon tracker as a control signal. This method offers the advantage of requiring no programmed rate information to maintain the planetary scan platform oriented to the vertical, as the required rate signal will be obtained from the horizon trackers. For offset pointing at relatively small angles from the vertical, a bias may be applied to the position signal from the horizon tracker.

The second method consists of programmed pointing the planetary scan platform in an open loop mode, utilizing the spacecraft as a primary reference. The open loop mode requires determination of the spacecraft position and velocity in orbit, which, in conjunction with the spacecraft attitude (determined by the celestial sensors) will enable computation of the series of directions in which the planetary scan platform must be pointed. During the photographic interval the inertial reference will be used as the reference coordinate frame from which pointing angles are determined. This second method has the disadvantage that the pointing accuracy will be decreased by the cumulative errors from the spacecraft to the planetary scan platform — primarily errors in alignment and in the planetary scan platform gimbal drive system.

For off-vertical photography at angles considerably displaced from the vertical, use of the programmed mode will be required where the desired angular displacement from the vertical exceeds the offset biasing capability of the Mars horizon sensor in the closed loop mode.



Pointing Accuracy. In Table 3-6 the optical fields of view are defined for the hypothetical all TV and recommended Film/TV photo-imaging systems. The fields of view for low- and medium-resolution imaging in either system dictate no stringent requirement in pointing accuracy. However, for high-resolution imaging, the hypothetical photo-imaging system utilizing the SEC vidicon camera poses the most stringent requirement. The SEC vidicon field of view is zero degrees 35 minutes square, and it would be desirable if the pointing accuracy of the planetary scan platform could be maintained to a fraction of this value (if specific areas on the Martian surface are to be observed with high-resolution). However, if only general high-resolution photography is desired, not of areas at a specific location on the planetary surface, the accuracy in pointing can be several times larger than the size of the optical field of view. The mission objectives for high-resolution imagery are therefore of considerable importance, as the impact on the design of the spacecraft (from the standpoint of alignment, boresighting requirements, and cost and complexity) is considerable.

Note that the recommended photo-imaging system, utilizing the Eastman Kodak type film camera, has a high-resolution lens field of view of three degrees 27 minutes, which is six times larger in linear dimension than the hypothetical TV high-resolution system utilizing the SEC vidicon. In this case the tolerances required in planetary scan platform pointing are considerably less stringent. Again, however, the requirement for pointing with an accuracy smaller than the size of the optical field of view will be dictated by the mission objectives of the high-resolution imagery — observation of areas at a specific location, or general high-resolution imagery without regard to the specific location of the area being observed.

Based on the more stringent requirements of the SEC vidicon camera, the planetary scan platform pointing accuracy design objective is 0.3 degrees (18 minutes of arc) which is approximately one-half of the field of view of the SEC lens.

Planetary Scan Platform Control Rate Accuracy Requirements.

Table 3-6 defines the ground resolution for the several subsystems of the all TV and combination Film/TV photo-imaging systems. The exposure interval has been calculated, considering the brightness of the scene (nominal value = 250 ft-l), optical system T-number, and sensitivity in the sensor (in meter-candle-sec). The allowable smear due to image motion has been established as one-third of the desired resolution, resulting in a root sum square value of system resolution approximately five percent larger than the nominal value.

The calculated equivalent planetary scan platform control, rate accuracy for the allowable smear is 0.17 milliradians/sec. This drive rate accuracy is applicable to all IMC rates regardless of the orbital position. Note that IMC is required only for the high-resolution Film Camera. Rates required vary from -4.05 to + 4.05 milliradians/sec for a 1000 km periapses and could reach 10 mrad/sec if periapses were 500 km.

3.5.3 Planetary Scan Platform Control System Design

3.5.3.1 Introduction

The planetary scan platform control system consists of the Mars edge tracker, the planetary scan platform drive electronics and the gimbal drives with tachometer and digital incremental encoder. The Mars sensor tracks the planet for closed-loop planetary scan platform pointing control. The planetary scan platform drive electronics processes the input to command the gimbal drives. The gimbal drives provide the planetary scan platform with three degrees of freedom; in the yoke, shaft and hinge axes using tachometer and encoder feedback for rate and position controls. The planetary scan platform control system provides the science payload with three distinct modes of operations:

- 1) Mars track mode - closed loop pointing control
- 2) Rate command mode - open loop rate control
- 3) Position command mode - open loop pointing control

3.5.3.2 Planetary Scan Platform Control System Design Approach

The performance requires a precision planetary scan platform drive system which must be capable of controlling its planet oriented

Table 3-6. Parameters Determining Pointing Accuracy and Stability Requirements for Hypothetical All TV and Recommended Film/TV Photo-Imaging Systems

	Hypothetical Photo-Imaging System			Recommended Film/TV Photo-Imaging System		
	Low - Resolution Color TV	Medium - Resolution R-B Vidicon	High - Resolution SEC Vidicon	Low - Resolution Color TV	Medium - Resolution Film	High - Resolution Film
Optical field of view	29° x 29°	8° 36' x 8° 36'	0° 35' x 0° 35'	29° x 29°	33° 27' x 33° 27'	3° 27' x 3° 27'
Ground resolution	1000 m	100 m	10 m	1000 m	100 m	10 m
Ground distance during exposure	125 m	31 m	3.1 m	125 m	31 m	62 m
Allowable smear	320 m	32 m	3.2 m	320 m	32 m	3.2 m
Exposure interval	40 m sec	10 m sec	1 m sec	40 m sec	10 m sec	20 m sec
Optical system	37.5 mm F/1.0	6.6 in. T/1.5	100 in. T/16	37.5 mm F/1.0	100 mm F/2.8	1000 mm F/4
Allowable IMC error	100%	100%	100%	100%	100%	4%



position (or pointing) by either a closed-loop servo with the Mars tracker as a feedback sensor or an open loop system to execute the position command from the onboard sequencer or ground station. In addition, it is also required to provide image motion compensation during photo-imaging operations by incorporating a programmed rate command closed-loop system using tachometer as a feedback means to achieve the planetary scan platform drive rate control. Furthermore, the operational requirements stipulate that the planetary scan platform drive system shall obtain full coverage of Mars from any point in the orbit when the spacecraft is in the nominal cruise attitude even during Mars eclipse.

Figure 3-14 shows the functional block diagram of a planetary scan platform control system meeting these requirements. The basic planetary scan platform drive system consists of the drive electronics, drive motors, gimbal drives, AC magnetic tachometer, magnetic incremental encoder and Mars edge tracker with its associated electronics. The drive electronics receives and amplifies the command input and in turn impresses a voltage on the AC motor thereby operating the gimbal drive which transmits torque to the planetary scan platform. The platform yoke axis or shaft drive position is sensed by the magnetic incremental encoder which sends binary coded pulses to the planetary scan platform drive electronics. The tachometer monitors the drive motor speed and feeds it back to the drive electronics for system stability and rate control during the IMC mode. Both encoder and tachometer signals are telemetered back to the earth for operational as well as engineering analysis.

The Mars edge tracker, biased or unbiased, scans the Mars horizon thereby establishing Mars local vertical. The error signal of the Mars tracker is processed through the tracker electronics and then fed back to the planetary scan platform drive electronics, thus closing the tracking (or pointing) loop.

An image motion compensation subsystem for the high-resolution photo-imaging operation is also shown in Figure 3-14. The subsystem accepts commands from the computer and sequencer subsystem or ground mission operation system to initiate the high resolution

photo-imaging programmer. The programmer is controlled by a pulse clock and in turn it controls the IMC rate schedule, which gives the IMC rate commands in an intermittent manner governed by a timer with a 0.150 second one shot circuit. The IMC rate schedule and the timer act in unison to effect the planetary scan platform rate control for IMC.

3.5.3.3 Modes of Operation

As noted previously, the planetary scan platform drive control system provides three modes of operation to cover the operational needs of the science payload.

- Mars Track Mode. The Mars tracker scans the planet continuously in this mode. Shown in Figure 3-14 the mode select switch closes the Mars tracker feedback loop switch S_1 and simultaneously opens the encoder feedback path switch S_2 . In this configuration, the Mars tracker scans the planet and provides the error signal to continuously adjust planetary scan platform position for compensation of the spacecraft orbital position. With this compensation feature, the planetary scan platform boresight points along the local vertical or at a desired direction provided the Mars sensor is biased.
- Rate Control Mode. The rate control mode may be required for specific scientific experiments or for image motion compensation (IMC) during photo-imaging operations. Both switches, S_1 and S_2 , stay in the Mars track mode terminals while the Mars tracker override commands to open the Mars tracker feedback loop by energizing switch No. 3. The override command, in this case, will be a rate command to slew the planetary scan platform with the tachometer feedback signal to close the loop of the planetary scan platform control system. When IMC operation is required, shortly before the opening of the camera shutter, the image motion compensation rate programmer with the schedule shown in Figure 3-14 switches (S_4) the planetary scan platform drive from the Mars track mode to IMC rate as a command input to the planetary scan platform drive electronics. The duration of the IMC rate mode is relatively short, in the order of 0.10 to 0.20 second. Since the photo-imaging operation is not a continuous operation, the IMC rate compensation is operated in an intermittent manner. The IMC rate command is

usually initiated sufficiently in advance so that at the moment the shutter of the camera opens, the compensation rate has already reached its steady-state rate to ascertain the best photographic result. In this mode the Mars tracker is kept out of the loop to avoid the IMC rate command fighting against the Mars tracker feedback signal. Upon completion of IMC mode, the programmer switches the Mars tracker mode to the planetary scan platform drive which slews the platform pointing towards the Mars center. The operation repeats in an intermittent manner as scheduled by the programmer throughout the photo-imaging operation.

The tachometer employed for rate feedback, to compare with the command for control of the desired rate, is capable of controlling to an accuracy of 0.17 milliradians per second over the range -10 to +10 mrad/sec.

- Position Control Mode. To orient the science instruments to a specific direction requires position control of the planetary scan platform drive system in an open loop manner. In this mode of operation, both switches S_1 and S_2 are activated to keep the Mars tracker out of loop while the magnetic incremental encoder is switched in the loop to continuously feed back the planetary scan platform drive position to the planetary scan platform drive electronics for comparison with the position command through the switch No. 1, as shown in Figure 3-14. The tachometer signal in this mode serves only as a rate feedback to stabilize the planetary scan platform drive control inner loops. The attainable pointing accuracy for the Mars track mode and position control mode are 0.292 degrees (3σ) each control axis and 0.486 degrees (3σ) each control axis respectively.
- Complete three-degree-of-freedom planet viewing. In addition to the basic control modes described above — which give three degrees of freedom limited in some view angles — there is another mode which gives complete three-degree-freedom of viewing. As shown in Figure 3-15, it is accomplished by combining a spacecraft roll maneuver with positioning of the PSP main shaft perpendicular to the orbit plane.

A detailed design analysis of the planetary scan platform control system for the three modes of operation and an error analysis of the planetary scan platform pointing accuracy are given in Appendix H. Appendix H also gives further detailed description of the planetary scan platform control system design implementation and supporting design analysis.

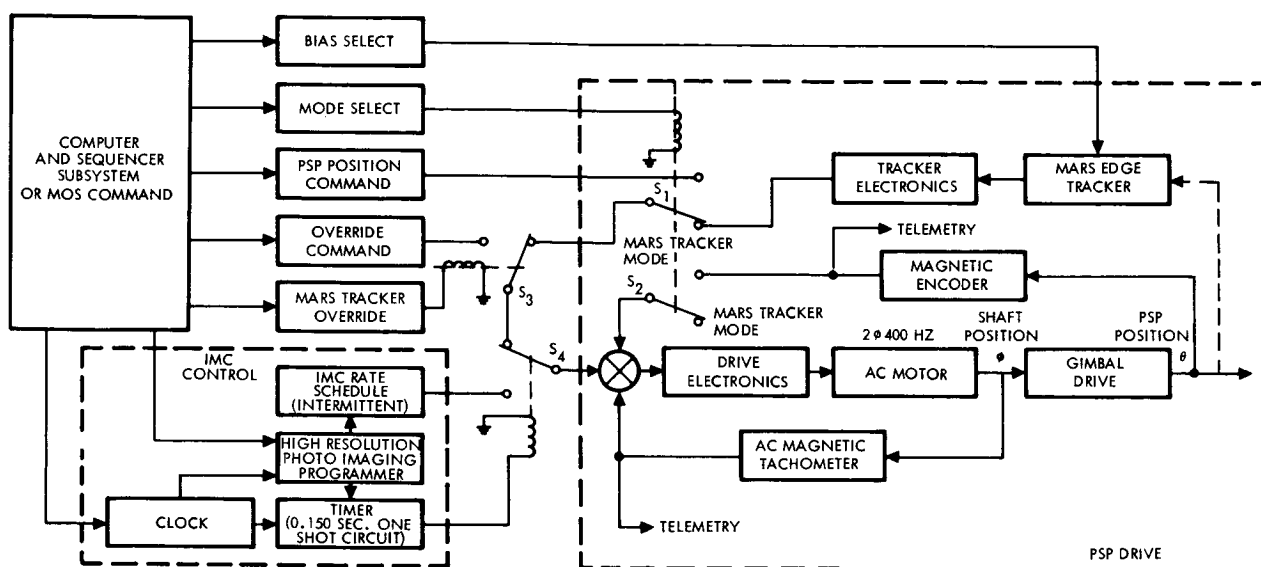
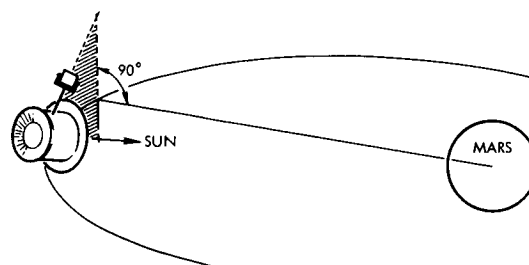
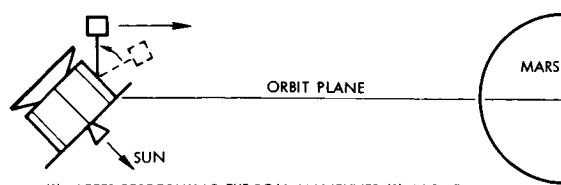


Figure 3-14
PSP DRIVE CONTROL SYSTEM, one for each axis, has three distinct modes of operation; Mars Track Mode, Rate or IMC Mode and Position Control Mode.



(1) THE SPACECRAFT IS TURNED ABOUT THE ROLL AXIS SO THAT THE PLANE MADE BY THE PLANETARY SCAN PLATFORM MAIN SHAFT AND THE ROLL AXIS IS PERPENDICULAR TO THE ORBIT PLANE.



(2) AFTER PERFORMING THE ROLL MANEUVER (1) ABOVE, THE PLANETARY SCAN PLATFORM MAIN SHAFT IS DRIVEN PERPENDICULAR TO THE ORBIT PLANE.

Figure 3-15
COMPLETE THREE-DEGREE-OF-FREEDOM planet viewing is accomplished by a combination of spacecraft roll maneuver and positioning of planetary scan platform shaft.



4. RECOMMENDED PHOTO-IMAGING SYSTEM

4.1 INTRODUCTION

The all TV photo-imaging system defined as part of the hypothetical payload has a number of deficiencies arising primarily from the rather small format sizes of the image tubes and the inherent lack of any storage capacity. For low-resolution imaging (1000 meters ground resolution) these characteristics are not particularly troublesome, but for medium- and high-resolution imaging (100 meters and 10 meters ground resolution, respectively) they have a decided impact on performance. The most objectionable characteristics of TV photo-imaging at medium- and high-resolutions are:

- 1) The system cannot fully exploit the 51.2 Kb/sec data link rate available in the early, critical part of the mission due to the limited storage capacity of the tape recorders.
- 2) Storage capacity is equivalent to only about one orbit worth of data transmission at 25.6 Kb/sec, or two at 12.8 Kb/sec. This greatly reduces operational flexibility for taking pictures at the most opportune times; moreover, if imaging is undesirable or impossible for any reason during a particular orbit, the opportunity for transmission during that orbit is irrevocably lost.
- 3) The field of view is quite small, especially for the high-resolution camera. This leads to difficult mosaicing problems, along with potential inability to locate high-resolution imagery with reference to the rest of Mars.
- 4) In order to get even marginally satisfactory coverage, multiple cameras and tape recorders must be used, adding to the weight and complexity of the system.

The deficiencies of the all-TV system enumerated above indicate the desirability of a different approach. Therefore, the photo-imaging system recommended by TRW replaces the three medium- and one high-resolution TV cameras with a dual-framing film camera. This system gives some very definite advantages over the TV cameras:

- 1) A larger, higher resolution sensor format gives much better fields-of-view, and with shorter focal length lenses.
- 2) Higher framing rates can be employed, since there is no relatively slow read-in to tape recorders required.

- 3) The permanent film record can be read out more than once if desired; for example, because of faulty transmission the first time or for verification of unusual or unexpected observations.
- 4) Frames can be read out quickly at low-resolution to determine if there is anything on them requiring the expenditure of the transmission time necessary for readout at the maximum potential resolution.

4.2 DEFINITION OF RECOMMENDED FILM/TV PHOTO-IMAGING SYSTEM

The photo-imaging system recommended for use in the Voyager spacecraft consists of a television camera which may be used in either a color or monochrome mode for low-resolution mapping coverage and a film system for medium- and high-resolution mapping coverage. This combination of dual-frame film and TV cameras can provide the most flexibility of any photo-imaging system. A preliminary specification for the photo-imaging system is defined in Figure 4-1.

4.2.1 Low Resolution Camera

The low-resolution television camera is identical to that proposed for the hypothetical (all-TV) system. It will provide either monochrome or color coverage with resolution of one kilometer from the altitude of 1000 kilometers (at periapsis). The use of color coverage will be of benefit in the identification of vegetation on the surface of Mars and in determining the seasonal variations in cloud formations and dust storms. When used in the color mode, with a field of 500 x 500 kilometers, 11 color pictures (33 monochrome frames – red, green, and blue in sequence), will provide coverage of approximately $6.25 \times 10^6 \text{ km}^2$, or 4.3 percent of the total area of Mars in one orbit. A more detailed description of the low-resolution television camera has been given in Section 2.2.2 of this volume.

4.2.2 Medium- and High-Resolution Film Camera

A dual framing Eastman Kodak film camera is recommended, similar to the camera developed for the lunar orbiter program. For medium- and high-resolution, two separate lens systems will be used. With lenses of 4- and 40-inch focal length the resolution will be 100 and 10 meters, respectively, from an altitude of 1000 kilometers (at periapsis).



PRELIMINARY SPECIFICATION

Photo-Imaging System

Purpose

To provide complete photographic coverage of the planet Mars at low resolution (1 kilometer) in color, mapping of large areas of Mars at medium resolution (100 meters) and imaging of selected areas at high resolution (10 meters).

LOW RESOLUTION CAMERA

Sensor - RCA 1-1/2 inch vidicon with slow scan target
Field of View - 29 x 29 degrees
Ground coverage - 500 x 500 kilometers (per frame)
Nominal ground resolution - 1 kilometer
Number of frames per orbit (maximum) - 33 frames (11 color pictures)
Photography time/orbit - 48 min (maximum)

MEDIUM AND HIGH RESOLUTION CAMERA

Sensor - Eastman Kodak 70 mm film camera

	Medium	High
Lenses-f.l.	100 mm (4 inches)	1000 mm (40 inches)
Format	60 x 60 mm	60 x 60 mm
Ground resolution	100 meters	10 meters
Frames per orbit	10	10
Ground track length per orbit (maximum)	7550 kilometers	
Photography time/orbit	48 min (maximum)	48 min (maximum)

CAMERA EQUIPMENT MOUNTED ON PSP

Weight 220 lb₃
Size 14.7 ft³
Power 50.6 watts (average)

TOTAL SYSTEM INCLUDING ONE TAPE RECORDER

Weight 238 lb
Power 58.6 watts (average)

EASTMAN KODAK
FILM CAMERA

LOW RESOLUTION
TELEVISION CAMERA
(MONOCHROME
OR COLOR)

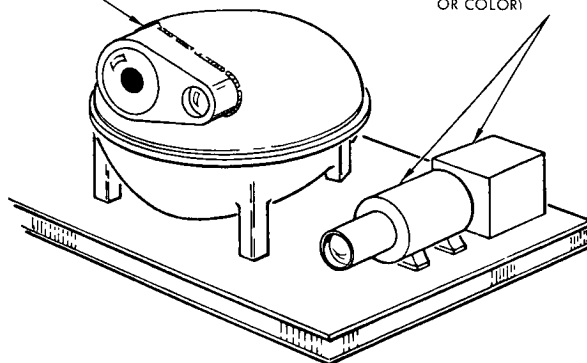


Figure 4-1

The camera system provides medium- and high-resolution frames on the same roll of 70 millimeter film. The two lenses (4- and 40-inch focal length) are boresighted, so that with simultaneous exposures each high-resolution frame covers the area at the center of the corresponding medium-resolution frame. The lateral ground coverage will be 600 to 60 kilometers for the medium- and high-resolution frames, respectively. During one orbit, 10 frames may be obtained in either or both of the modes on a 60 x 60 millimeter film format.

The Voyager film camera system will differ from the lunar orbiter in those respects dictated by differences in the missions. The lenses will be considerably different in focal lengths and design types, a new Bimat-type processing formulation appropriate to the Voyager mission duration will be used, additional radiation shielding will be added, and the readout system will be designed to permit continuous transmission of photographic information during the lifetime of the spacecraft in Mars orbit. A preliminary specification for the film camera is shown in Figure 4-2.

PRELIMINARY SPECIFICATION

Recommended Film Camera Systems

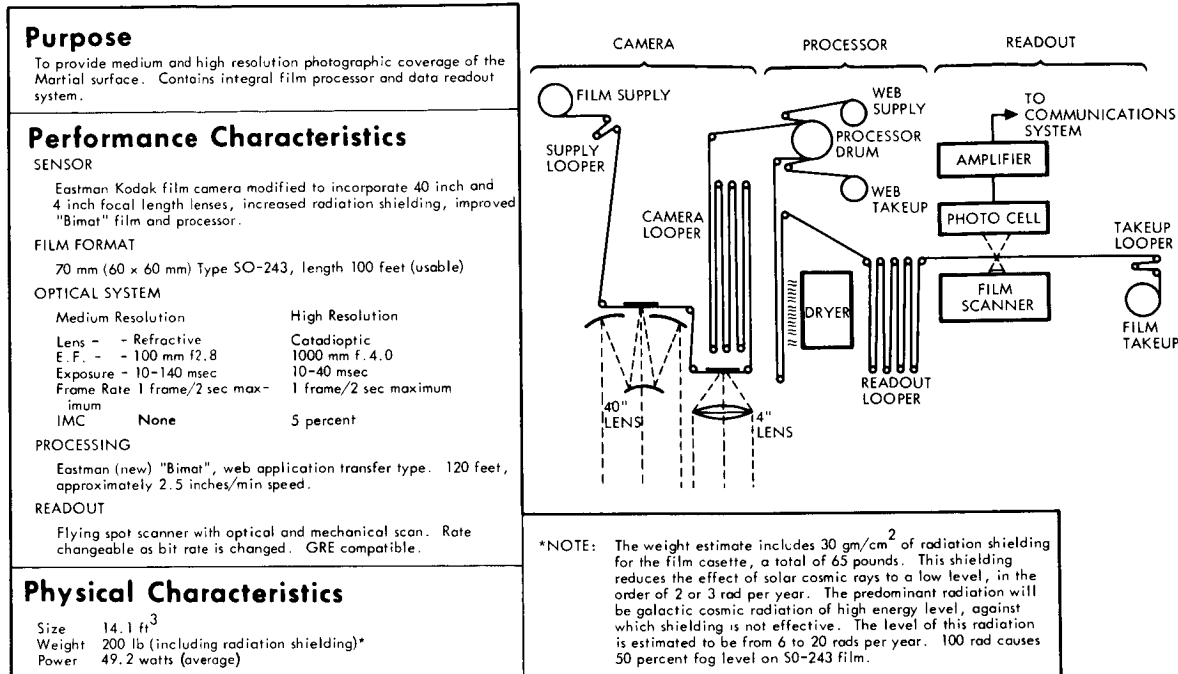


Figure 4-2

4.2.3 Equipment Configuration

The film camera and the low-resolution television camera will be mounted on the planetary scan platform and boresighted. The film camera will be contained in its own pressure vessel and will be provided with internal heating elements.

The total weight of the Film/TV equipment mounted on the planetary scan platform will be 220 pounds with an average power consumption of 50.65 watts. The total system weight, including the tape recorder for the low-resolution television camera, will be 240 pounds, with an average power consumption of 58.59 watts.

4.2.4 Photographic Coverage

The coverage obtained by the photo-imaging system in one orbit is shown in Figure 4-3. The lateral coverage of the two subsystems increases with increasing displacement from periapsis, because of the



corresponding increase in altitude. The maximum coverage obtained by the low-resolution television camera in one orbit will be determined by the range of solar illumination angles permissible for color photography, rather than by tape recording capacity. The coverage of the film camera during one orbit is also limited only by the range in solar illumination angles desired for monochrome photography.

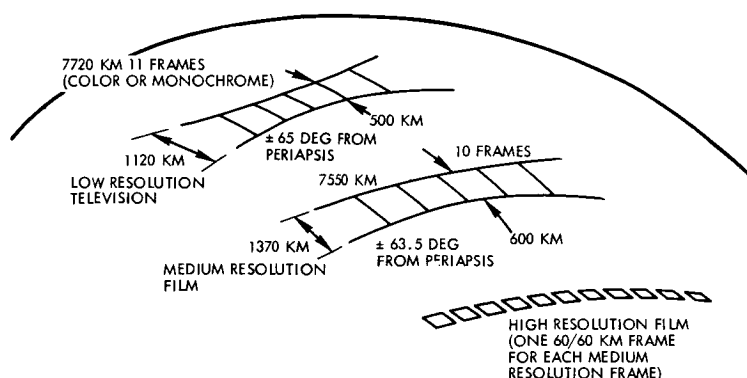


Figure 4-3

MAPPING COVERAGE-RECOMMENDED FILM AND TV PHOTO IMAGING SYSTEM. Both the color television and film cameras will obtain continuous ground coverage over the maximum range which is determined by the desired solar illumination angles. One high resolution film frame may be taken for each medium resolution film frame. The tracks of ground coverage which are illustrated are displaced for clarity.

During a 6-month lifetime in orbit, with a total data transmission capacity of 2.64×10^{11} bits, the coverage which may be obtained is given in Table 4-1.

The coverage indicated is limited by the data transmission rate, not by camera capabilities. For example, if a higher data rate were available, a camera loaded with 500 feet (less than five pounds) of film could gather about 2.6×10^{12} bits of data, or 260 percent Mars coverage at 100 meters resolution plus 2.6 percent Mars coverage at 10 meters resolution.

Table 4-1. Coverage of Mars Obtained with Film/TV Camera System*

	Low-Resolution TV (color)	Medium- Resolution Film	High- Resolution Film
Nominal ground resolution	1 km	100 m	10 m
Nominal coverage/frame	500 x 500 km	600 x 600 km	60 x 60 km
Bits/frame (6-bit encoding)	8.58×10^6	1.24×10^9	1.24×10^9
Frames/orbit (maximum)	11 color picture (33 frames)	10	10
Data/orbit (maximum)	2.83×10^8 bits	12.4×10^9 bits	12.4×10^9 bits
Ground coverage/frame (nominal)	$2.5 \times 10^5 \text{ km}^2$	$3.6 \times 10^5 \text{ km}^2$	3600 km^2
Area of Mars	$1.46 \times 10^8 \text{ km}^2$	$1.46 \times 10^8 \text{ km}^2$	$1.46 \times 10^8 \text{ km}^2$
Data allotment (total data link capacity = 2.64×10^{11} bits/6 months)	1.52×10^{10} bits	1.25×10^{11} bits	1.25×10^{11} bits
Number of frames	585 (color pictures) 1755 frames	100	100
Percentage of Mars area covered	100%	26%	0.26%

*This estimate of coverage of the Martian surface which will be obtained in low-, medium-, and high-resolution is based upon a data link capacity of 2.64×10^{11} bits over a period of 6 months. Equal portions of the total data capacity are assigned to medium and high resolution photography. This estimate is based upon the ground coverage obtained at the nominal altitude of 1000 kilometers.



4.3 INTEGRATION OF FILM/TV SYSTEM

4.3.1 Mechanical Integration

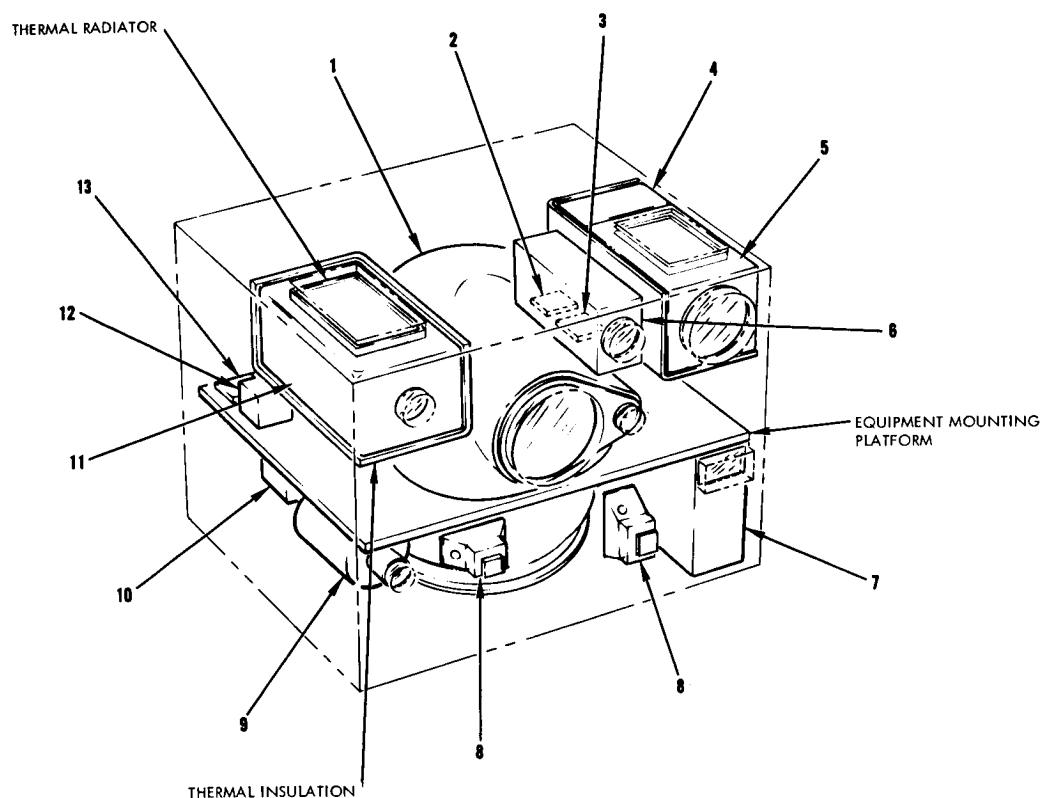
The general design features of the planetary scan platform for the recommended photo-imaging system are essentially the same as that given in paragraph 3.4.2.1. There are some significant deviations however, as illustrated in Figure 4-4. The combination film/TV system recommended for this configuration and the system's mechanical interface requirements are tabulated in Table 4-2.

Inclusion of the film/TV photo-imaging system significantly alters the interior arrangement of the planetary scan platform experiments, primarily because of its size and the way it is mounted. As may be seen in Figure 4-5, it occupies about one-third the width and almost the entire cross section of the interior. Also shown is the mounting geometry. Note that the photo-imaging package is also attached to the bottom panel for thermal control purposes. The shelf, which still serves to support other experiments, is sculptured around the package to allow for its installation. The box walls also support some equipment. The other features of the planetary scan platform — accessibility, thermal and meteoroid protection, etc. — are unchanged.

A weight statement for this Film/TV and core science payload is given in Sppendix G-II.

4.3.2 Electrical

The same technique of overall science electrical integration will be implemented for the recommended Film/TV photo-imaging system and core experiments contained within the planetary scan platform as described in Section 3.3.3 for the hypothetical payload. The individual experiment electrical interface requirements will be satisfied by the design of the interconnect cable between experiments and the remote decoding, switching, and multiplexing equipment contained within the planetary scan platform.



- | | |
|--|--|
| 1. FILM CAMERA SYSTEM-HIGH AND MEDIUM RESOLUTION | 8. MARS SENSOR (2 PLACES) |
| 2. PSP REMOTE PHOTO-IMAGING MULTIPLEXER | 9. LOW RESOLUTION COLOR TV CAMERA |
| 3. PSP REMOTE MULTIPLEXER | 10. LOW RESOLUTION COLOR TV CAMERA ELECTRONICS |
| 4. CRYOGENIC REFRIGERATOR | 11. HIGH RESOLUTION INFRARED SPECTROMETER |
| 5. BROADBAND INFARED SPECTROMETER | 12. PSP REMOTE SCIENCE SWITCHING ASSY. |
| 6. INFRARED RADIOMETER | 13. PSP REMOTE DECODER UNIT |
| 7. ULTRAVIOLET SPECTROMETER | |

Figure 4-4

CORE EXPERIMENTS WITH FILM-TV photo-imaging system (as recommended) are arranged on PSP as shown in preliminary layout.

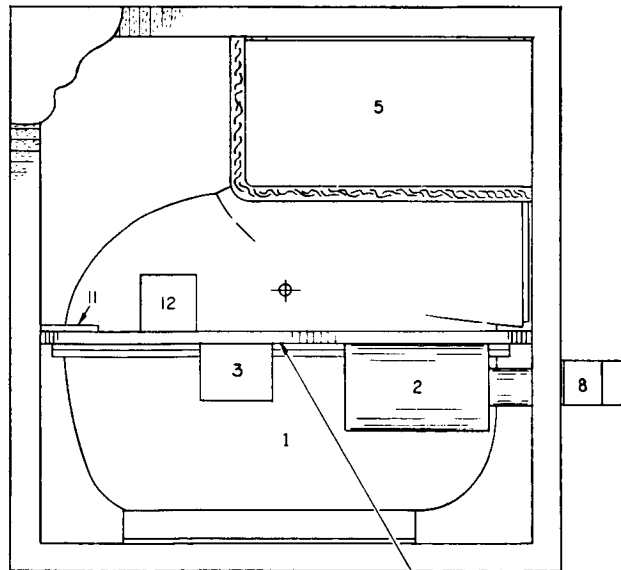
Table 4-2. Recommended Film/TV Photo-Imaging System

Experiment	Number Required	Sensor				Co-Located Electronics	
		Total Field of View (deg)	Alignment (deg)	Aperture (in. diam)	Volume (in. ³)	Weight (lb)	Volume (cu in. ³)
Film System							
a) High-resolution	1	3.45 x 3.45	0.05	11.0	20,000	200	-
b) Medium-resolution	1	33.45 x 33.45	0.05	2.0			-
Low-resolution TV camera	1	29 x 29	0.05	2.5	212	10	100

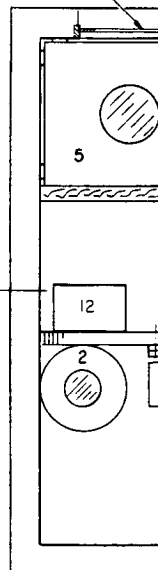
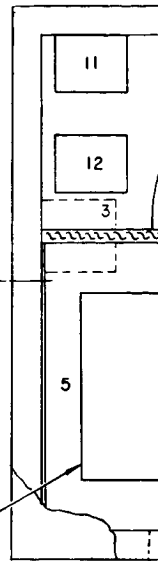


EQUIPMENT LIST

1. FILM CAMERA SYSTEM - HIGH & MEDIUM RESOLUTION
2. LOW RESOLUTION COLOR TV CAMERA
3. LOW RESOLUTION COLOR TV CAMERA ELECTRONICS
4. BROADBAND INFRARED SPECTROMETER
5. HIGH RESOLUTION INFRARED SPECTROMETER
6. INFRARED RADIOMETER
7. ULTRAVIOLET SPECTROMETER
8. MARS SENSORS
9. PSP REMOTE PHOTO-IMAGING MULTIPLEXER
10. PSP REMOTE MULTIPLEXER UNIT
11. PSP REMOTE DECODER UNIT
12. PSP REMOTE SCIENCE SWITCHING ASSY
13. CRYOGENIC REFRIGERATOR



THERMAL RADIATOR (TYP)



A07-179 1/4

FOLLOUT FRAME

4-9A

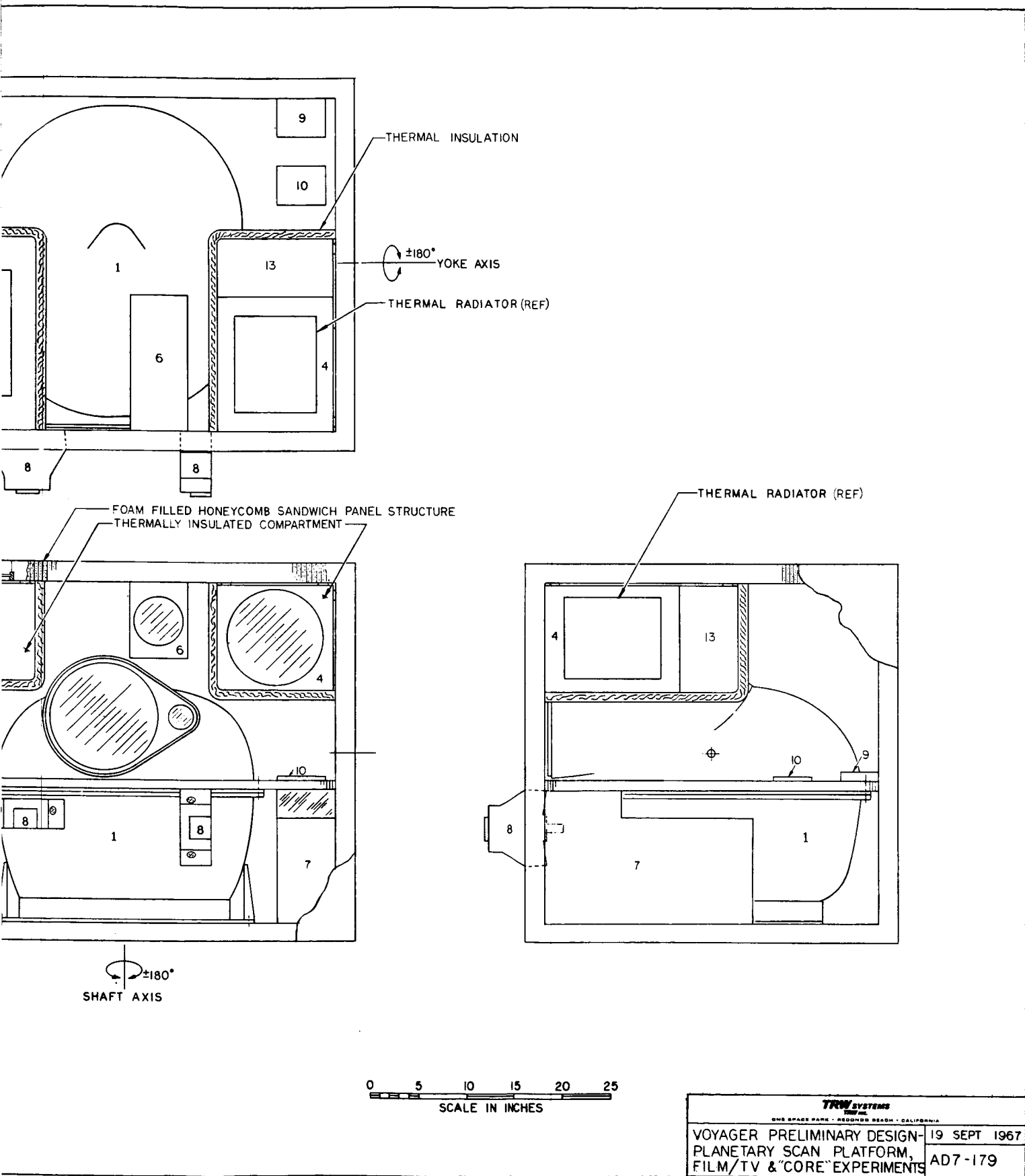


Figure 4-5
FILM/TV AND CORE EXPERIMENTS are located on PSP as shown in layout. Pressurized film system container is surrounded by other experiments and electronics packages to obtain balance of the PSP box about the yoke gimbal axis.

4-9B

FOLDOUT FRAME



4.3.2.1 Data Handling

The basic difference between the hypothetical photo-imaging system and the recommended Film/TV system in regard to data handling hardware is the number of data handling channels (multiplexers) and data tape recorders. For the hypothetical photo-imaging system, five multiplexers (one for each camera) and four data tape recorders are required. For the recommended system, two multiplexers, one for the Film and one for the TV camera, and one tape recorder for the low-resolution TV multiplexed data is required.

Appendix B, Table 2 contains a telemetry list for the recommended payload. The core experiment telemetry is the same as described in Section 3.3.11. The recommended Film/TV system requires 36 analog, 6 digital, and 22 discrete data channels as compared to 62 analog, 15 digital, and 33 discrete data channels for the hypothetical photo-imaging system.

4.3.2.2 Sequencing

A list of commands exclusively for the operation and control of the Film/TV system plus core experiments is contained in Appendix A, Table 2. The total required is seven power commands and 42 discrete commands as compared to 10 power commands and 103 discrete commands for the hypothetical payload.

A total of 56 discrete commands are required at the output of the planetary scan platform remote decoder for the Film/TV payload as compared to 123 discrete commands for the hypothetical all TV payload. Fourteen of the 56 discrete commands are required for control of power switching (ON/OFF) within the remote science switching assembly for individual experiments.

Power estimates for the Film/TV photo-imaging system and core experiments are contained in Appendix C, Table 3, and Table 4 for an eclipse and a noneclipse orbit respectively. The total average power required for the Film/TV payload operations is 84.99 watts for an eclipse orbit and 64.16 watts for a noneclipse orbit, as compared to 55.43 and 34.60 watts for an eclipse and noneclipse orbit respectively for the hypothetical payload. The additional power for the Film/TV payload is required for film readout.

Figure 4-6 and 4-7 illustrate typical experiment operations (data acquisition periods) for an eclipse and a noneclipse orbit respectively for the Film/TV photo-imaging system and core experiments. In addition to illustrating anticipated science operations as a function of altitude and true anomaly, the associate science data tape recorder operations are also shown. The composite science power profile including the tape recorders and photo-imaging film readout for each orbit are also shown. The orbital conditions and science operations for the Film/TV payload are basically the same as described in Section 3.3.2.4 for the hypothetical payload.

If a total data estimate of 24.8×10^9 bits for a Film picture taking sequence that includes 10 medium- and 10 high-resolution pictures is assumed approximately 134.5 hours will be required (at a rate of 51.2 kbs) to play back all data associated with that picture taking sequence. The playback of photo-imaging film data can be terminated (between frames for example) for short periods of time such that either the science or TV photo-imaging tape recorders for the low-resolution color TV can be played back whenever either is filled.

4.3.2.3 Electrical Integration Design

An electrical interface chart for the Film/TV photo-imaging system and core experiments is contained in Appendix D, Table 2. The total number of wires required to service the Film/TV payload contained within the planetary scan platform is derived in Appendix E, Table 2 based upon the interface chart. A total of approximately 227 wires is required for the Film/TV payload as compared to 374 wires for the hypothetical payload. By virtue of the planetary scan platform remote equipment approximately 30 wires (without redundancy) will be required in the planetary scan platform cable wrap-up.

4.3.3 Thermal

The recommended Film/TV photo-imaging system requires an average of 69 watts of power (Table 4-3). Of this the film-type photo subsystem requires an average of 67 watts, which produces a local "hot spot". The film-type photo subsystem requires mounting on cold plates in the planetary scan platform, attached so as to obtain maximum heat transfer

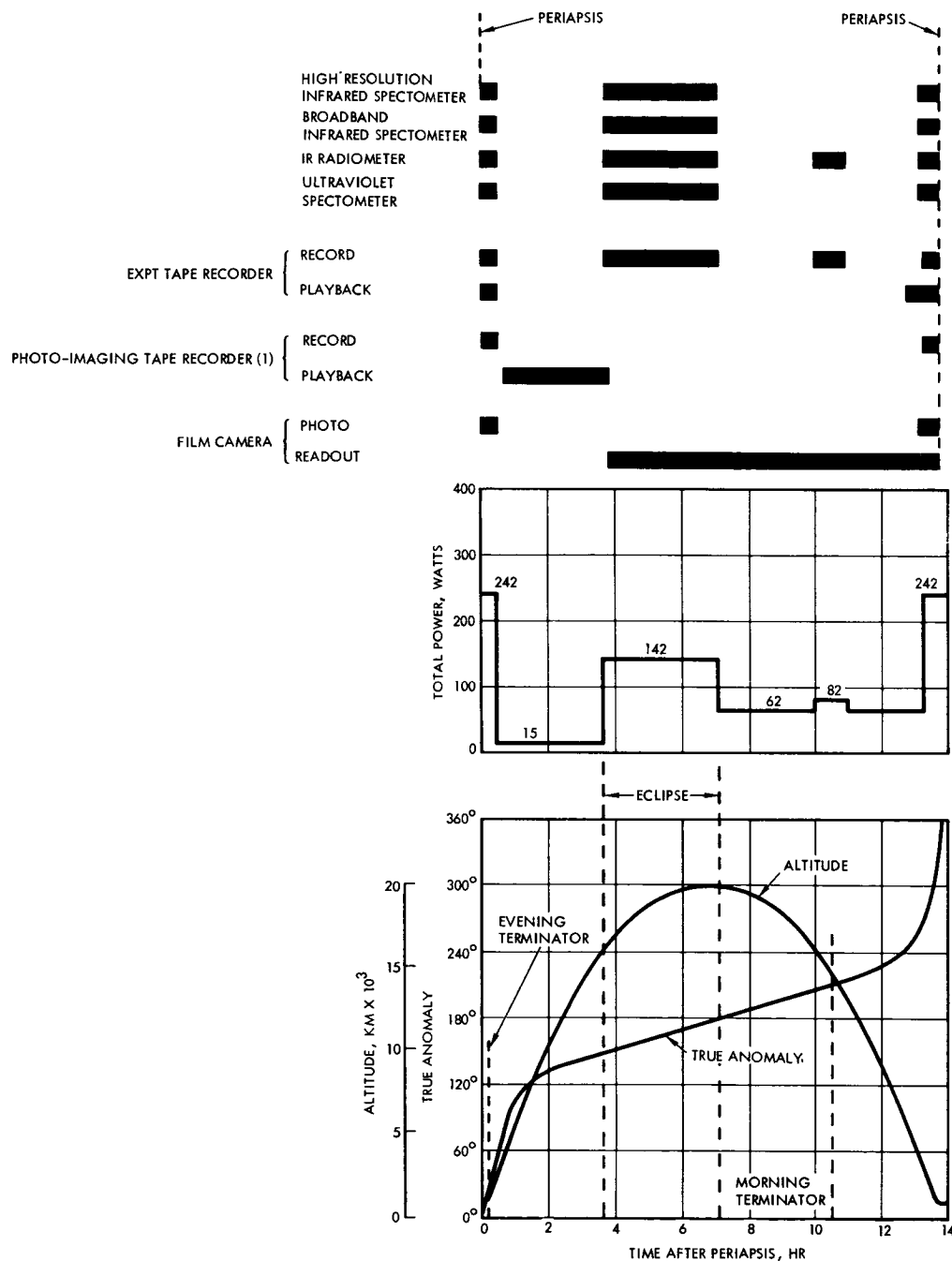


Figure 4-6
RECOMMENDED FILM/TV AND CORE EXPERIMENTS power profile is shown for a nominal eclipse orbit. The average power is 84.99 watts.

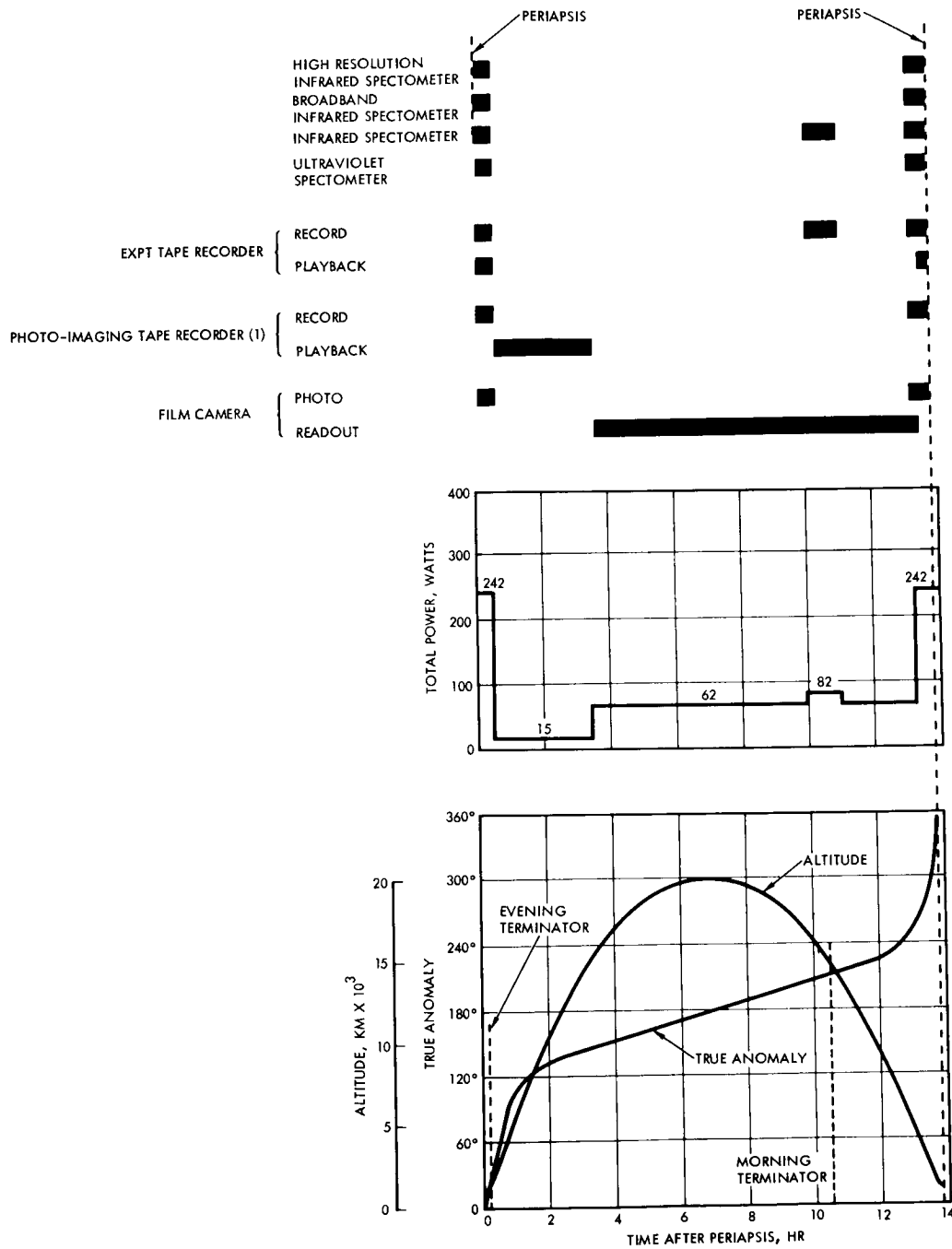


Figure 4-7
NONECLIPSE ORBIT POWER PROFILE for film/TV and core experiments is shown for a nominal 1000 KM X 20,000 KM orbit. The average power requirement is 64.16 watts.



between the two plates (see Figure 4-8). The mounting plate attached to the planetary scan platform structure is maintained at a temperature between 35 and 85°F.

Table 4-3. Average Power and Duty Cycle for Film/TV Photo-Imaging System

	<u>Duty Cycle Percent</u>	<u>Power (watts, ave)</u>
1 low-resolution color TV	5.8	1.45
1 film system		
Photo mode	5.8	6.10
Standby to photo	-	-
Processing	-	-
Readout	94.2	<u>61.50</u>
		69.05

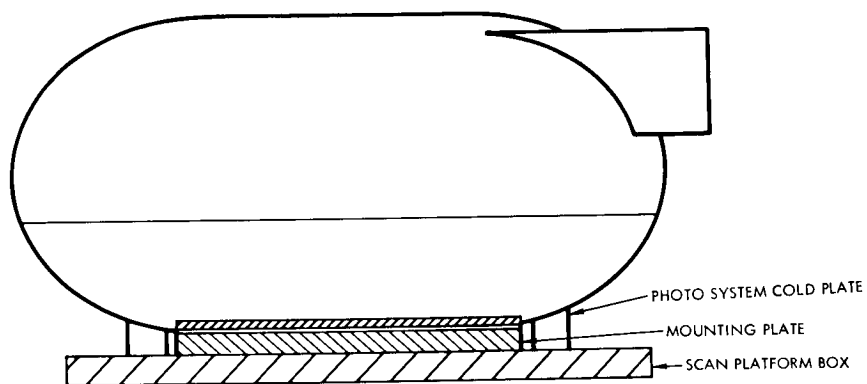


Figure 4-8
COLD PLATE MOUNTING is required for the film-type photo subsystem to dissipate a local hot spot.



5. DEFINITION OF ADDITIONAL EXPERIMENTS

5.1 INTRODUCTION

The twelve additional experiments described in this section are intended to be representative of those that may be included in the spacecraft if the weight allocated to the science payload is increased. The experiments included are based primarily upon the suggested list provided as a guideline by MSFC; two additional experiments are included: The Cosmic Ray Detector and the Micrometeor Impact Detector.

The Cosmic Ray Detector is a desirable adjunct to the Neutron Albedo Experiment, since the outgoing neutron flux from the planetary atmosphere is properly meaningful only if it correlates with the cosmic ray flux that is incident at that time. The Micrometeor Impact Detector is included, in addition to the Meteor Flash Detector, in order to measure the relatively larger flux of small particles whose mass is below the threshold of detectability of the flash detector, and to provide continuity with conventional micrometeoroid experiments on previous space missions.

With these two additions it is felt that the experiment list provides coverage of all of the important Voyager scientific objectives. Since the science payload is to be selected by MSFC, no assignment of relative priority has been made; rather the goal has been to demonstrate compatibility of an illustrative payload which will satisfy desirable scientific objectives. No attempt has been made to define each experiment in complete detail, for this is the province of the individual experimenters, and many changes may be expected as the payload definition progresses in the next phase of the Voyager program. Experiment parameters and requirements have, however, been specified in sufficient detail to include all principal interface requirements which affect the design of the overall spacecraft system. To this end, TRW has drawn upon its considerable experience in the area of science payload integration for such programs as the OGO and Pioneer series, and when appropriate, experiments from these spacecraft have been selected for discussion as the present illustrative examples. In most cases the

experiment so described is an optimum rather than minimal experiment, and this is felt to be wise for purposes of sizing a spacecraft system.

In this section each experiment is described by a brief statement of its purpose, followed by a discussion of the instrument and an outline of its performance characteristics and interface requirements. A block diagram has been given in as much detail as practicable, including a diagrammatic sketch of the functional parts of the apparatus and an overall view of the complete package or packages. Numerical values of the parameters are, in general, confined to the outline and the diagrams, and the discussion of the instrument includes the rationale of the design in relation to the experiment purpose.

5.2 ATMOSPHERE MARS SPECTROMETER

5.2.1 Purpose

To detect, and measure the density distribution with altitude of, mass species in the outer atmosphere of Mars - principally ions but also including neutrals, if possible.

5.2.2 Instrument

The most desirable type of mass spectrometer for use on a spacecraft which may include an ultra high-sensitivity magnetometer is one which does not employ a magnetic field. Quadrapole, time-of-flight and Bennett RF spectrometers satisfy this condition. The quadrapole, or massenfilter, type is perhaps the most desirable of these from the viewpoint of ruggedness, reliability and lightness of weight. Flight models have been built for OGO and for various rocket soundings of the earth's atmosphere which are sensitive from 1 to 45 AMU, a range which includes all expected constituents of the Martian atmosphere from H to CO₂. Accordingly, the OGO instrument has been taken as a typical one, and its parameters used for the purposes of the present illustrative payload.

This instrument measures both ions and neutral constituents. It is questionable whether neutral particles can be detected and measured in the present nominal orbit with periapsis at 1000 km. However, lower orbits are still being actively considered, and a periapsis of 400 km or



less might make the ability to detect the neutral exosphere very valuable. Even then the probability of detecting CO_2 is low. Nonetheless it seems preferable, for purposes of determining spacecraft sizing and compatibility with possible experiment payloads, to consider instruments with typical rather than minimal requirements. Hence, the OGO ion and neutral particle mass spectrometer with its full mass range of 1 to 45 AMU is described here.

The instrument characteristics and the interface requirements for a Mars orbiter mission are shown in Figure 5-1.

Problem areas requiring special consideration for this experiment are the data rate, the power requirements, and the instrument pointing direction. The mass spectrometer requires a very high data rate whenever it is operating, and the electric power requirement is high (to ionize neutrals) when it is operating in the neutral particle mode. It is desirable to reduce the duty cycle to achieve these results; for example, the experiment may be assigned to operate during part of only one orbit in, say, ten, and then only when the instrument viewing direction is proper, and other experiments may be turned off as necessary. This is to say that the mass spectrometer is given first consideration during the limited times when it is operating.

The problem of the pointing, or viewing direction, is simply that the instrument aperture should view generally forward of the spacecraft velocity vector, or more specifically, it should not look at the spacecraft wake, which has been swept free of particles. Neither the planetary scan platform nor any fixed-orientation mounting on the body of the spacecraft fully satisfies this condition. At orbital altitudes, the principal particle density is that of the ion exosphere, and it might at first thought seem desirable for the instrument to view the planet from the planetary scan platform to detect the particle flux rising up from lower altitudes. However, exospheric particles are describing ballistic trajectories. Some are rising and falling in parabolic orbits, some are in escape trajectories, and others may be in "permanent"

PRELIMINARY SPECIFICATION

Atmospheric Mass Spectrometer

Performance Characteristics

SPECTRAL RANGE
0.0 — 40.0 AMU

SPECTRAL RESOLUTION
1.0 AMU for masses <10.0 AMU
5% for masses 10.0 AMU to 40.0 AMU

SPECTRAL SCAN
6 (5 second) sweeps for one complete operational cycle

DYNAMIC RANGE
 10^6 for collector 10N current, accomplished by three spectral scans at sensitivity settings of 10^{-10} , 10^{-12} , and 10^{-14} amp

ACCURACY
2% over each spectral scan at a given sensitivity setting.

POINTING
Center line of instrument aperture shall be aligned to point in the general direction of the orbital velocity vector (e.g., at periaapsis of initial orbit)

ALIGNMENT
Pointing direction specified above shall be known to within $\pm 3.0^\circ$.

FIELD OF VIEW
 2π STERADIANS

Interfaces

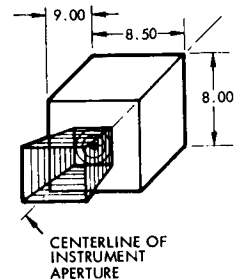
ELECTRICAL

Commands
Power — ON-OFF
Cone power
Filament selector

Telemetry
Data
8 — 8-Bit words
(The high data rate requirement for this instrument suggests the use of a flexible format sampled 1 orbit in 10)

Diagnostics
Electronics package temperature
Scan mode monitor
Sensitivity level monitor
Power supply (S) voltage monitor

Power
26.0 Watts (peak)
22.0 Watts (aver.)



MECHANICAL

Weight
8.5 lb

Dimensions
8.50 x 9.00 x 8.00

Volume
612 In.³

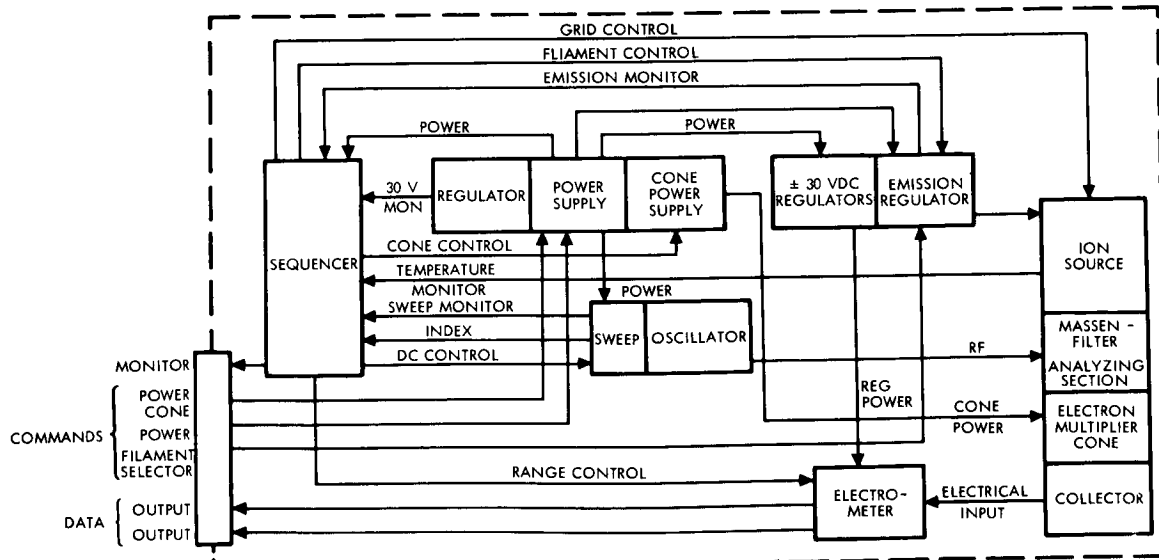
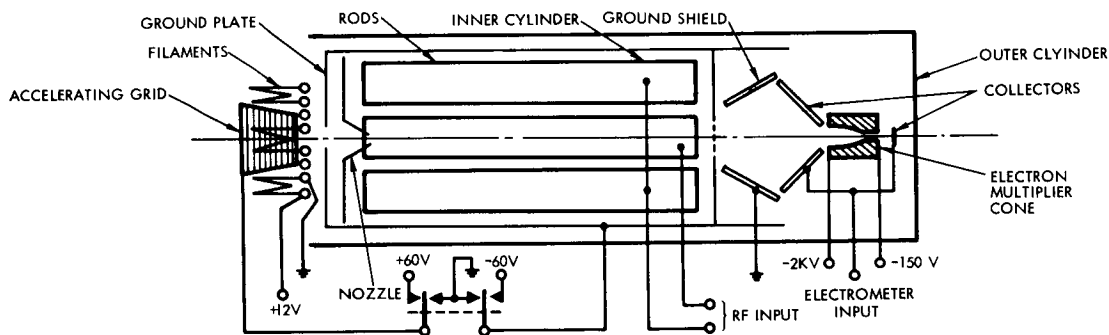


Figure 5-1



elliptic orbits. The flux is more nearly isotropic than directed; thus there is no significant advantage in viewing toward the planet. Since weight and space are at a premium on the planetary scan platform in any case, it is felt that the mass spectrometer should be body-mounted on the spacecraft and the operating periods chosen when the instrument is looking in the forward direction of motion. The mounting orientation can be chosen so that this is the case near periapsis for the initial orbits once the injection orbit has been definitely chosen. Exploratory operation at other parts of the orbit can serve to determine the significance of the wake effect. The projecting grid structure above the instrument aperture serves to draw obliquely moving particles into the aperture so that the field of view approaches 2π steradians, and it is anticipated that effective operation can be secured if the limits of the 2π field of view are inclined no more than 120 degrees to the forward spacecraft velocity vector.

5.3 POLARIMETER

5.3.1 Purpose

Measure the state of polarization of light scattered by the Martian atmosphere, to determine thereby the particle sizes and to study the distribution of aerosols in the atmosphere.

Measure the state of polarization of light reflected from the planet's disk and to correlate the results with the principal surface features, such as the bright areas, dark areas, and the polar caps.

5.3.2 Instrument

The instrument described here is a compact automatic polarimeter developed under a NASA contract as a pilot model for lunar and planetary missions by spacecraft, and described in considerable detail in the recent literature.* The optical configuration is shown diagrammatically in Figure 5-2. The complete instrument consists of two such optical

*S. F. Pellicori and P.R. Gray, An Automatic Polarimeter for Space Applications, Appl. Optics 6, 1121, June 1967.

PRELIMINARY SPECIFICATION

Polarimeter

Performance Characteristics

Spectral Range — 1900 Å — 6000 Å
 Spectral Resolution — 5 bands — 250 Å bandwidth
 Spectral Scan — 5 minutes — complete scan
 Dynamic Range — 10^6 — incident light intensity
 Accuracy — 1 percent — for P/M tube output
 Field of View — 1.0° total field per telescope
 Pointing — Views mars along vertical line-of-sight correlated with visual viewing equipment.
 Accuracy — Angular position (as specified above) shall be known to within $\pm 0.5^\circ$

Interfaces

ELECTRICAL

Commands

Power — ON-OFF
 Voltage Level — High or low (600 V — 1400 V)
 Calibrate

Telemetry

Data
 2 — 8-Bit words

Diagnostics

#1 High Voltage Power Supply Voltage Level
 #2 High Voltage Power Supply Voltage Level
 Electronics package temperature monitor
 Multiplexer status monitor
 Gain selector status

Power
 6.0 Watts

MECHANICAL

Weight

6.0 lb

Dimension

16.00 x 7.00 x 7.00 (+6.00 — Telescope)

Volume
 784 In.³

THERMAL

Operating temperature — 70°C + 40°C

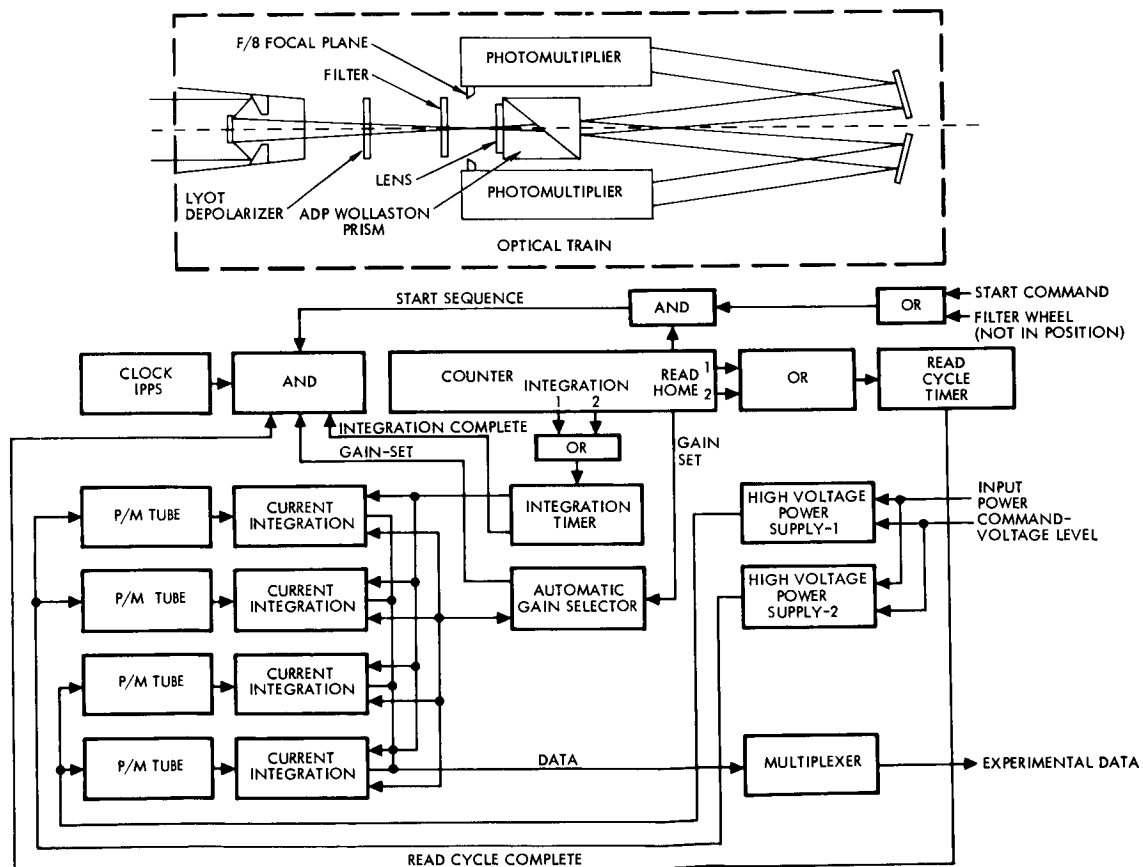
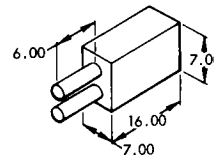


Figure 5-2

trains, physically separate but mounted in the same package with the two Wollaston prisms at 45 degrees to each other. This arrangement requires, therefore, four photomultiplier tube detectors in all, and permits linearly polarized light to be analyzed simultaneously into four intensities from which the Stokes parameters can be determined. This in turn yields the magnitude and azimuth of the polarization vector at a given wavelength with one observation.

Each optical train contains five filters, each passing a bandwidth of approximately $250 \text{ \AA}$; each of the five pairs of filters in the two telescopic systems consists of identical filters, so that the same spectral interval is observed simultaneously by both telescopes. The wavelength dependence of the polarization of scattered light is strongly affected by the size of the scattering particles. With the ability to select five narrow wavelength bands distributed over the complete ultraviolet and visible spectrum between $1900 \text{ \AA}$ and $6000 \text{ \AA}$, this instrument is well suited for a study of both atmospheric and surface particle size.

Polarimetric mapping of the planet's surface features from orbit is valuable not only as a backup to the direct surface exploration by the lander capsule but also because correlation of the telescopic mapping results with direct information via the lander can aid in the interpretation of similar polarimetric surveys of other planets, both by earth-based astronomy and from the early satellite fly-by and orbiter missions.

Hence, the polarimeter should be mounted on the planetary scan platform to view the planet along the vertical line-of-sight, correlated with the visual imaging equipment to a reasonably high accuracy $\approx \pm 0.5$ degree. Together with the infrared and ultraviolet spectrometers, the polarimeter should also at times view the planet's limb to a maximum excursion of 10 degrees from local vertical in order to receive light scattered by the atmosphere.

5.4 SOLAR OCCULTATION EXPERIMENT

5.4.1 Purpose

To determine the composition and density of the atmosphere of Mars at the surface and also as a function of altitude.

5.4.2 Instrument

The instrument consists essentially of a telescope and photometer combination with which the change in light intensity can be accurately measured over the short time interval just before and just after eclipse of the sun by the planet as the spacecraft enters or leaves the geometrical shadow. At such times the light path from the sun to the spacecraft traverses, in successive time intervals, atmospheric layers of progressively greater or less density.

Such a solar occultation experiment seems seldom, if ever, to have been performed from spacecraft up to the present time, and definition of both an instrument for this specific purpose and of the best experimental procedure is lacking. There are at least three experimental difficulties to be surmounted in the design of the instrument. For example, the change in light intensity due to changing optical density over the optical path through the planetary atmosphere must be distinguished from the varying intensity as the spacecraft traverses the penumbra. Light covering a wide spectral range would certainly give results characteristic of the total planetary atmosphere, but it is not known whether sufficient absorption might be expected from a minor constituent such as water vapor to yield the altitude distribution of that constituent if the measurement were performed at only the wavelength of a strong water vapor absorption band. It is not certain whether the most important data is the time-change in the light intensity or the determination of the exact time of extinction; the data readout requirement is more severe for the latter information. These questions can be answered only by a detailed study of the experimental situation.

In view of these uncertainties, the instrument described in Figure 5-3 represents a preliminary conceptual design rather than a final definition. Assumptions have been made regarding the problems described in the previous paragraph, and the result is believed to be a set of instrument requirements illustrative of what may be expected from this experiment.

The penumbra effect is eliminated by focussing the telescope on a small area of the sun's disk; a small aperture stop represents diagrammatically a means of accomplishing this. It is felt that the possibility should be left open of measuring the occultation in light of more than one wavelength band; four are shown for illustrative purposes and are arbitrarily assigned to known strong absorption bands of H_2O , CO_2 , and O_2 , in addition to the visible spectral range. In order to detect small intensity changes a differential measurement might be desirable. This could be done with a second optical system which views several degrees behind the main telescope and, for each wavelength, provides an unmodified intensity as a reference. Because of the added complexity this has not been shown. A two-telescope, differential intensity instrument with only one or two wavelength bands would be comparable in size, weight, and data requirements to the instrument described here.

The mounting and operation requirements of this instrument are simple, since it has a small field-of-view, it looks directly at the sun, and operates only in eclipse orbits during the few minutes just before and just after the eclipse itself. Hence the instrument is body-mounted on the spacecraft and accurately aligned (known to ± 0.1 degree) to look directly at the sun.

PRELIMINARY SPECIFICATION

Solar Occultation

Performance Characteristics

SPECTRAL RANGE

Experimenter will select appropriate detectors and filters for a given mission. Four spectral bands have been shown for illustrative purposes

- Channel 1 Visible (4000Å - 6500Å)
- Channel 2 O_2 Band (0.76μ)
- Channel 3 CO_2 Band (2.7μ)
- Channel 4 H_2O Band (3.2μ)

SPECTRAL RESOLUTION

Dependent upon filter (or combinations chosen by experimenter)

SPECTRAL SCAN

None

DYNAMIC RANGE

10^4 (for incident light intensity).

ACCURACY

0.1% for differential measurement, or less depending upon a detailed analysis of noise level and penumbra effect.

POINTING

Views the sun directly.

ALIGNMENT

The pointing direction specified above shall be known to within $\pm 0.1^\circ$.

FIELD OF VIEW

Should be very small; e.g., 0.1° or less.

Interfaces

ELECTRICAL

Commands

Power - ON-OFF

Calibrate

Telemetry

Data*

4 - 8-Bit words

* This experiment requires an extremely high data, particularly during entry and exit from an eclipse. A flex format mode of data sampling is suggested for this experiment.

Diagnostics

High voltage monitor

Package temperature monitor

Sensor temperature (3)

PM tube dynode volt monitor

Power

20 watts

MECHANICAL

Weight

18.0 lb

Dimensions

6.00 x 7.00 x 12.00

Volume

504 in.³

THERMAL

Operating Temperature

$-70.0^\circ\text{C} \rightarrow +40.0^\circ\text{C}$

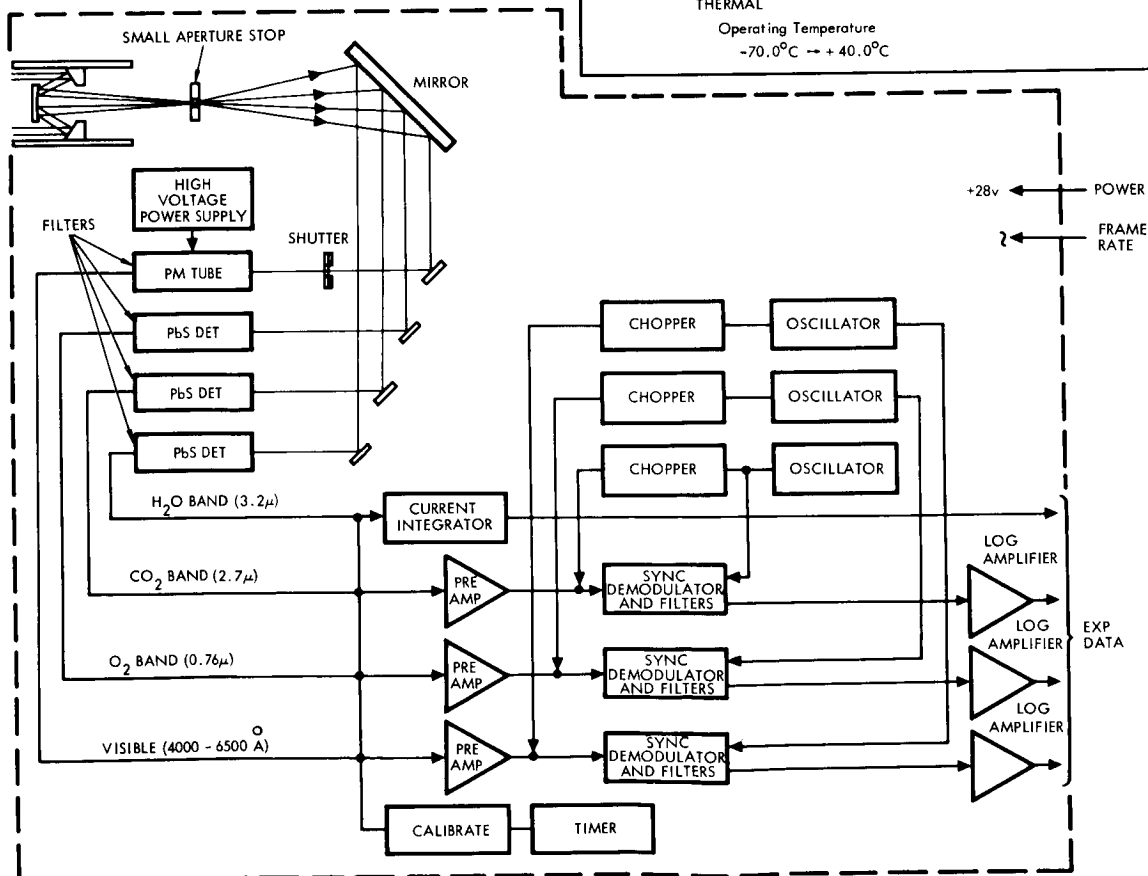
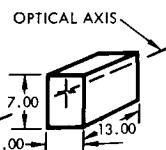


Figure 5-3



5.5 GAMMA RAY SPECTROMETER

5.5.1 Purpose

To measure the intensity and energy of gamma rays emitted from the surface of Mars due to both natural and induced radioactivity.

5.5.2 Instrument

A relatively simple prototype gamma ray scintillation spectrometer is described here which is similar to that used initially to measure the gamma radiation from the moon's surface. For the Mars orbiter it will provide a general survey of the high energy photon emission from the planet but only a crude mapping function. Mounting problems associated with background radiations from the spacecraft, and collimation and sensitivity problems are such as to indicate that a true gamma ray telescope for detailed surface mapping must be deferred until thorough studies and more detailed instrument definitions are available.

The detector package consists of a CsI(Tl) scintillation crystal surrounded by a thin shell of plastic scintillator to provide a phoswich system that discriminates against charged particles. Both scintillators are viewed by the photomultiplier tube, which feeds into standard pulse-height analyzer and counter electronics. Characteristics of the instrument are shown in Figure 5-4.

If the surface area viewed by the instrument is less than the full planetary disk, as is the case over essentially all parts of presently considered orbits, the gamma intensity at the spacecraft will not change much from near to distant parts of the orbit, and the problem of background radiation produced by cosmic rays impinging on the spacecraft is always present and always severe. To minimize this background it would be desirable to boom-mount the gamma ray spectrometer. However, the spectrometer must also view the planet. Ideally, it should occupy its own small planetary scan platform on the end of a boom. At present the choice has been made to locate it on the

PRELIMINARY SPECIFICATION

Gamma Ray Spectrometer

Performance Characteristics

Energy Range — 0.2 Mev — 10.0 Mev

Energy Resolution — 10 Percent

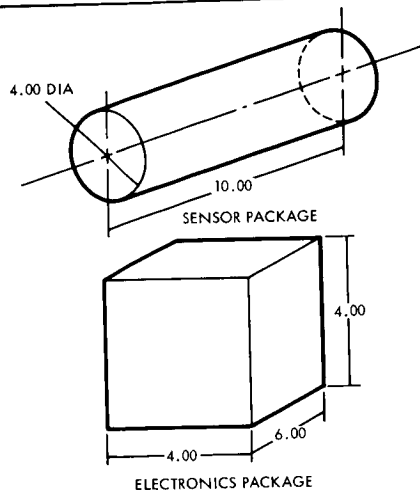
Dynamic Range (Count Rate) — Not Applicable

Accuracy — 2 Percent (Counter Readout)

Field of View — 10.0 Deg

Pointing — Views planet along vertical line-of-sight — within 2.0 Deg correlated with visual imaging equipment

Alignment — Angular position as described above shall be known within ± 0.5 Deg



Interfaces

ELECTRICAL

Commands

Power — ON-OFF

Calibrate

TELEMETRY

Data

2 8-Bit words

Diagnostics

Electronics package temperature

Sensor temperature

Height velocity power supply voltage level monitor

Dynode voltage level monitor

MECHANICAL

WEIGHT

Sensor — 7.0 lb

Electrical package — 6.0 lb

DIMENSION

Sensor — 4.00 Dia x 10.00

Electrical package — 4.00 x 4.00 x 6.00

VOLUME

Sensor — 125 In.³

Electrical package — 96.0 In.³

THERMAL

Operating Temperatures

Sensor — -20°C $\rightarrow$ $+30^{\circ}\text{C}$

Electrical package — -20°C $\rightarrow$ $+30^{\circ}\text{C}$

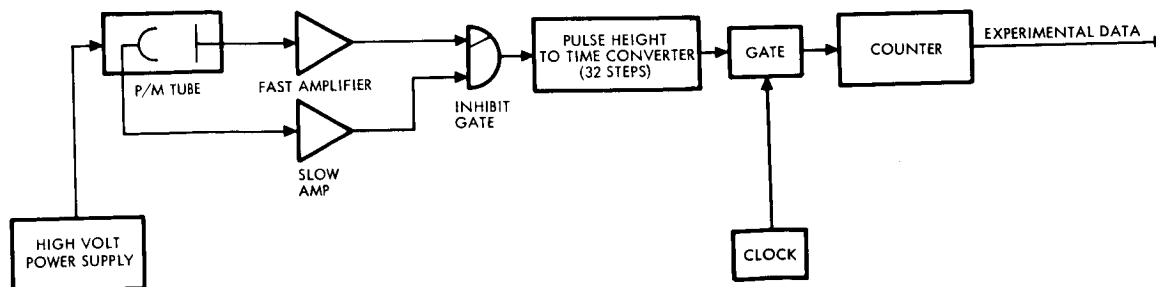
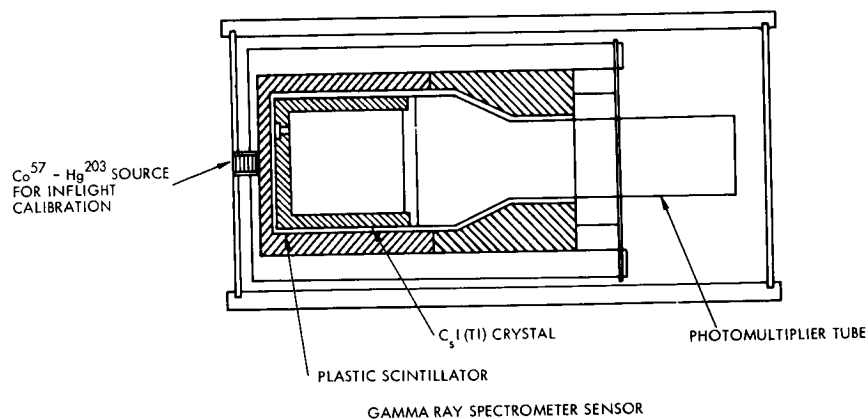


Figure 5-4



existing planetary scan platform rather than to mount it on a fixed-orientation boom and restrict its operating times to those periods when it happens to be directed toward the planet. This is a significant problem area, and a detailed study is required in order to better define the instrument and its requirements.

5.6 NEUTRON ALBEDO

5.6.1 Purpose

To measure the cosmic-ray-generated neutrons which are emitted outward from the atmosphere and surface of Mars.

5.6.2 Instrument

The experiment devised by J. A. Lockwood and co-workers to measure the earth's neutron albedo from OGO-F is satisfactory for a prototype for Mars as well, and has been selected for discussion here.

The neutron detector is a cylindrical He^3 proportional counter which responds essentially only to neutrons. It is surrounded first by a moderator and then by a cylindrical shell of charged particle counters which permit discrimination between those neutrons produced in the moderator and in the detector assembly itself and the true cosmic ray neutron-leakage flux.

As described by Lockwood: "The detector measures both the integrated neutron flux and the fast-neutron flux in the energy range from 2-10 Mev. --- The total number of counts in the neutron detector is monitored, as well as the gated neutron rate. Whenever a charged particle counter is triggered, the neutron detector is blanked off for at least 200 μ s. The output of this channel is referred to as the gated neutron rate. The difference between the total neutron-counting rate and the gated rate should be equal to the number of neutrons/second produced in the moderator-detector assembly. To measure the fast-neutron flux a photomultiplier views the scintillator (which is also the neutron moderator). Whenever a scintillation pulse is followed within

25 μ s by a neutron count, both of which are unaccompanied by a guard-counter firing, the size of the scintillator pulse is stored in the integral discriminator. From this information we hope to be able to evaluate the neutron energy spectrum in the energy range from 2-10 Mev. "

A simplified block diagram and characteristics of the experiment are shown in Figure 5-5.

The instrument is mounted on a 14 foot boom in order to minimize the count due to neutrons produced in the body of the spacecraft by cosmic rays, and since the detector is sensitive to particles arriving over essentially a full 4π steradians, no pointing or orientation is required.

In order to properly interpret the Martian neutron albedo data from this experiment, it is essential that the local cosmic ray activity be monitored simultaneously in order to detect variations in the particle fluxes and spectra associated with solar activity or other causes. A cosmic ray detector is therefore included in this set of experiments as a necessary adjunct to the Neutron Albedo Experiment.

5.7 COSMIC RAY DETECTOR

5.7.1 Purpose

To monitor the cosmic ray flux near Mars, of both galactic and solar origin, in as much detail as possible in order to aid the interpretation of the data obtained from the Neutron Albedo Experiment; specifically, to measure differential and integral energy spectra, composition, and time-fluctuations of the flux of high energy protons, alpha particles, and heavier ions in the range of 0.4 to 1000 Mev/nucleon.

5.7.2 Instrument

The instrument described here was developed by F. B. McDonald and co-workers for OGO-E. It consists of three detector systems for high, medium, and low energy ranges, respectively, plus highly developed electronics systems which provide unusually complete data readouts of the many significant parameters of composite high energy



PRELIMINARY SPECIFICATION

Neutron Albedo Experiment

Energy Range — 0.1 Mev → 2.0 Mev

Energy Resolution — 16 logarithmically spaced intervals over the above energy range

Energy Scan — Not Applicable

Accuracy — 2.0 Percent for data readout

Field of View — $\sim 4\pi$ steradians

Special Requirements — Must be boom mounted — boom length expected to be 10.0 feet minimum

Pointing — Not Applicable

Alignment — Not Applicable

ELECTRICAL

Commands

Power — ON-OFF

High voltage range set (2)

Calibrate

THERMAL

Operating Temperature

Sensor — $-20^{\circ}\text{C} \rightarrow +40^{\circ}\text{C}$

Electrical package — $-20^{\circ}\text{C} \rightarrow +60^{\circ}\text{C}$

TELEMETRY

Data

3 — 8-Bit words

Total BPS — 600

Diagnostic

Log count rate monitor threshold voltage level

Internal light pulser

Scintillator output

Power — 6 watts

MECHANICAL

Weight

Sensor — 5.5 lb

Electrical package — 6.5 lb

Dimensions

Sensor — 4.5 Dia x 12.50 In.

Electrical package — 8.00 x 7.60 x 4.00

Volume

Sensor — 19.8 In.³

Electrical package — 244 In.³

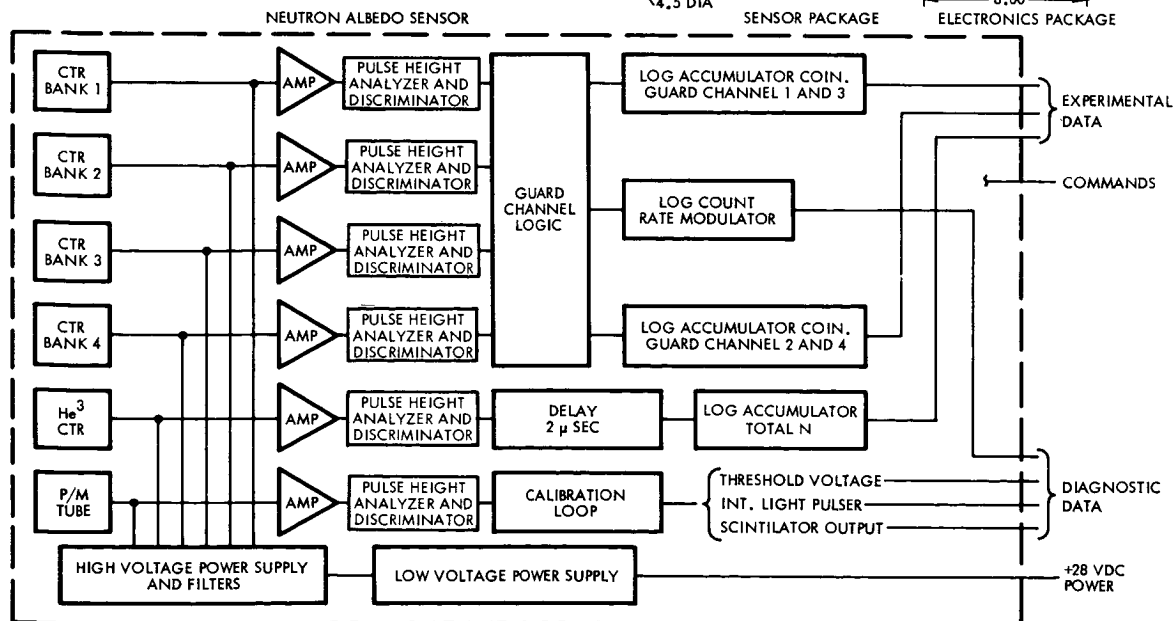
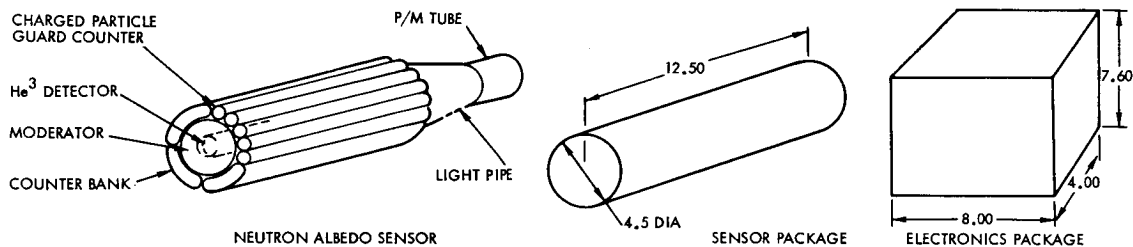


Figure 5-5

charged particle flux. It is a sophisticated instrument for a monitoring function, but the interface requirements which it brings to the spacecraft system are relatively modest in view of the information returned, and it would permit complete analysis of the interactions associated with the Martian neutron albedo flux, as well as providing more extensive information about the cosmic ray flux itself.

The high energy detector consists of two separated, relatively thin CsI(Tl) scintillators followed by a sapphire Cerenkov radiator which is in fact the window of the photomultiplier tube detector. Measurement of the pulse height outputs identifies the charge and energy of the incident particles. The medium energy detector consists of three CsI(Tl) scintillators, two being thin $\frac{dE}{dx}$ detectors and the third, a thick total energy detector which is enclosed by a plastic guard scintillator, followed by a light pipe and photomultiplier tube. In addition to regular charge and energy analysis, provision is made by an integral analyzer to provide flux and energy information during high flux solar proton events. The low energy detector consists of three totally depleted solid state diodes surrounded by a guard scintillator followed by a photomultiplier tube.

In Figure 5-6 the type of spectral information obtained by each of these detector assemblies is tabulated for each energy range in the summary of instrument characteristics. Triggering levels are provided for automatic gain switching and for determining the proper operating mode at any given time for optimum information readout. It should be noted that the high and the medium range detectors actually have only one photomultiplier each, whereas the diagrams in each case show the single tube as three in order to indicate the nature of the multi-channel outputs corresponding to the individual sensors.

Each detector assembly may be regarded as a wide-angle cosmic ray telescope; the fields of view for the high, medium and low energy telescopes are approximately 30° , 30° , and 55° , respectively, and the instrument should be mounted on the spacecraft body to view directly toward the sun in order to measure solar as well as galactic cosmic

Performance Characteristics

ENERGY RANGE

(For nuclei from $Z = 1$ to $Z = 20$)

HIGH ENERGY DETECTOR:

Differential charge and energy spectra, 50 to 1000 Mev/nucleon
Integral flux > 100 Mev/nucleon.

MEDIUM ENERGY DETECTOR:

Differential charge and energy spectra, 20-80 Mev/nucleon, flux and differential energy spectra - 80-300 Mev/nucleon.

LOW ENERGY DETECTOR

Differential charge and energy spectra, 3.5 to 18.5 Mev/nucleon.
Integral flux and energy spectra > 0.4 Mev/nucleon.

RESOLUTION TIME

Coincidence resolving time 2 nanosecond for each detector system.

ENERGY

High Energy

2.5 Mev/Channel

Medium Energy

0.24 Mev/Channel

Low Energy

55 Kev/channel

ACCURACY

3 percent (or less) for count rate.

FIELD OF VIEW

High Energy 30 deg - conical
Medium Energy 30 deg - conical
Low Energy 55 deg - conical

Pointing - Centerline of telescopes aligned parallel to solar normal

Interfaces

ELECTRICAL

Commands

Power - ON/OFF

Multiplexer status - 2

Telemetry

Data

2 - 8-Bit words

Diagnostics

Dynode voltage monitor

Electronics package temperature

Multiplexer status

Sensor temperature monitor

Power

5 watts

MECHANICAL

Weight

22 lb

Dimensions

8.00 x 8.00 x 14.00

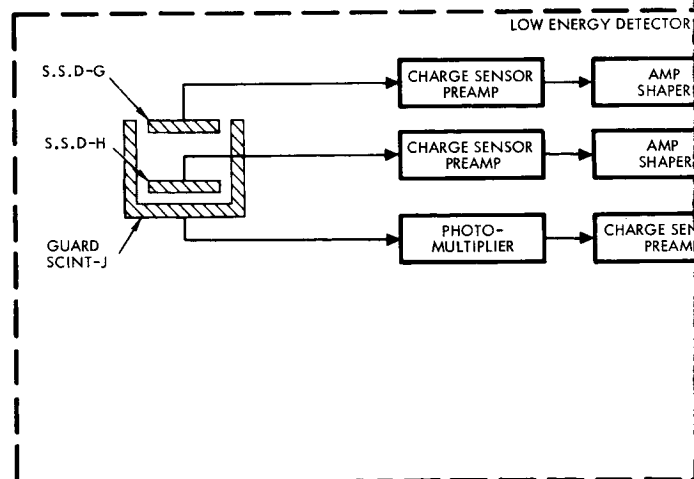
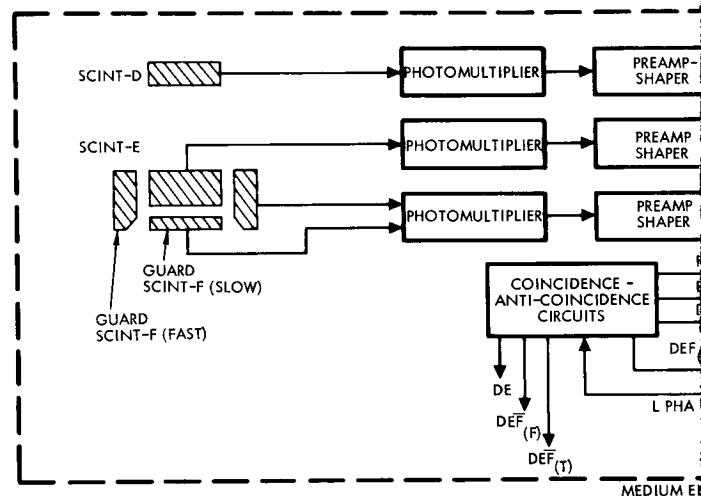
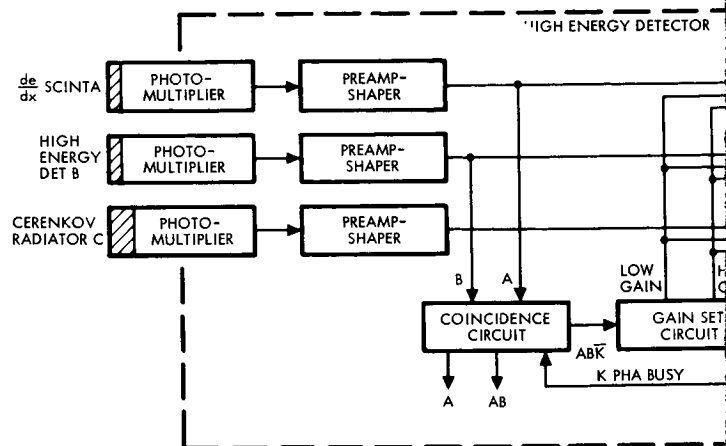
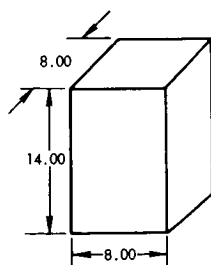
Volume

896 cu.in.

THERMAL

Operating Temperature

-30° → +60°C





RY SPECIFICATION

Ray Experiment

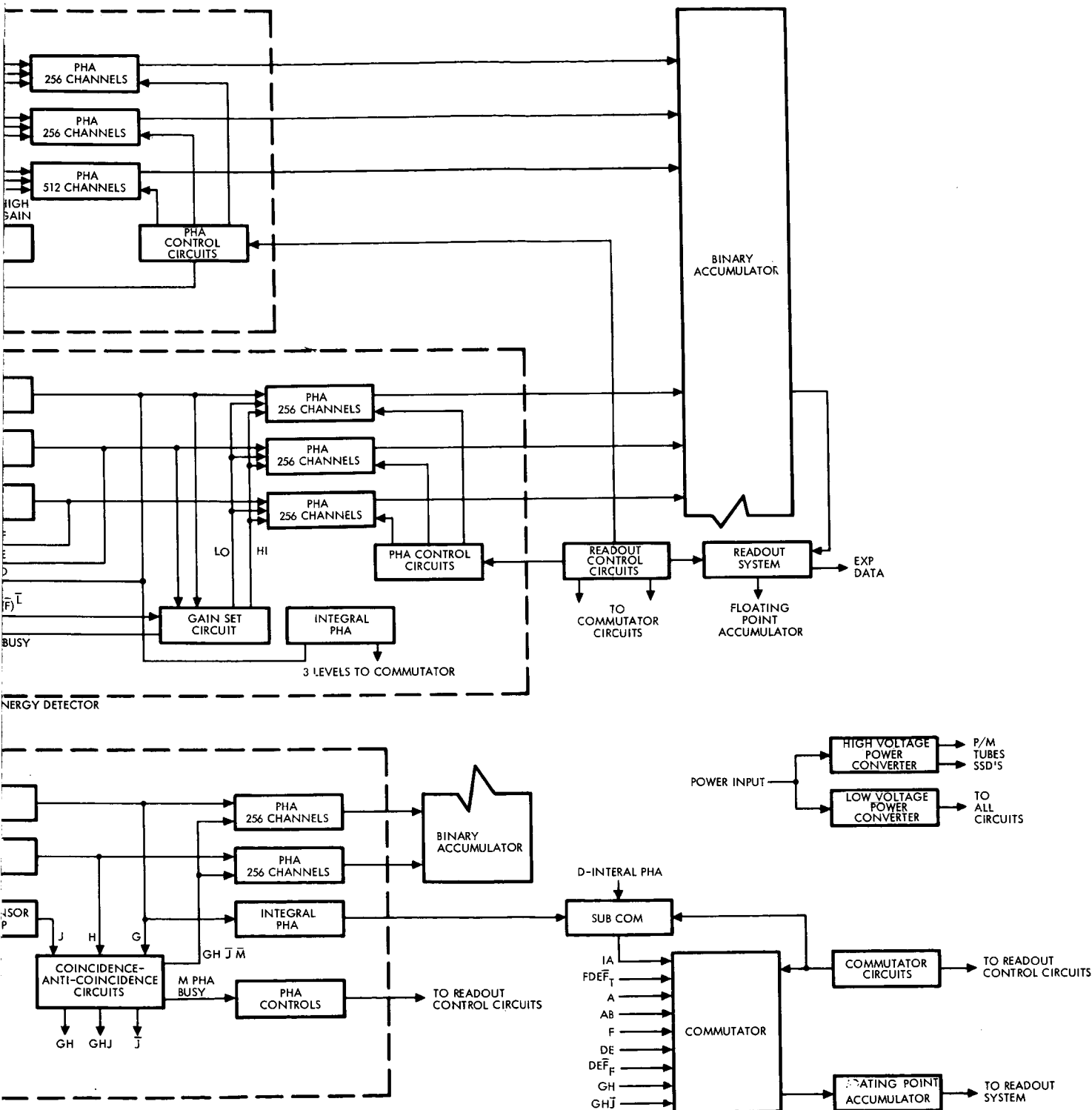


Figure 5-6

rays. All three telescopes must have light-tight shields - over the outer scintillator surface for the high and the medium energy detectors, and just inside the first (and thinnest) solid state detector in the low-energy telescope. This has been done satisfactorily in the case of similar instruments already flying - for example, on Pioneer. The outer surfaces must also have suitable thermal coatings. This is not expected to be a problem except possibly in the case of the thin solid state detector which is the first element in the low energy telescope; here careful analysis of the thermal problem and ways to solve it will probably be required.

5.8 MAGNETOMETER

5.8.1 Purpose

The instrument chosen for this experiment is a triaxial flux-gate magnetometer. Typically a fluxgate sensor measures the magnitude and sense of the ambient magnetic field along the axis of the sensor. Vector information regarding the field is obtained through measurement of the component fields in their mutually perpendicular axes.

Two range scales have been provided in the instrument. Information obtained from the Mariner IV mission indicated the level of the Martian field to be quite small; for this reason a low level range of ± 7.25 gamma was selected. A high level range of ± 60.0 gamma has also been provided to furnish data during periods of solar flare activity and magnetic storms. Normally the lower range is used, unless the field level exceeds ± 4.0 gamma at which time the scales are changed automatically. Instrument characteristics are shown in Figure 5-7.

Calibration of the instrument is accomplished by inserting a bias signal and monitoring the apparent change in field level. The sensitivity of a flux gate sensor is approximately 0.1 gamma. The readout accuracy of this instrument is ± 0.25 gamma.

The sensor package for this instrument will be mounted on a boom positioned to provide maximum separation from the remanent field of the spacecraft as well as to minimize effects of stray fields produced by operation of vehicle subsystems.

PRELIMINARY SPECIFICATION

Magnetometer

Performance Characteristics

Measure - 3 orthogonal components of Martian magnetic field

Range - Low ± 7.25

High $\pm 60.0 \gamma$

Response - 0.0 - 0.5 Hz

Commands

Power - ON/OFF

Range change

Calibrate

Heater power - ON/OFF

Telemetry

Data

1 - 8-Bit word/axis

Sampled ~ 25 times/sec

Diagnostic

P/S voltage monitor

Electrical temperature monitor

Sensor temperature monitor

Power

5 watts total

MECHANICAL

Electronics - $6 \times 6 \times 2$ inches

Sensor - $3 \times 3 \times 3$ inches

CONSTRAINTS

Spacecraft magnetic field at sensor position shall be $> 0.75 \gamma$.

Purpose

The magnetometer will operate on a 28 vdc (unregulated) power. Power line filters and pre-regulators will be used to minimize EMI problems and to provide internal regulation of power supplied to the instrument.

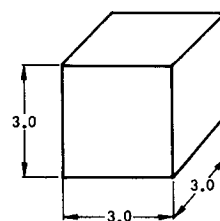
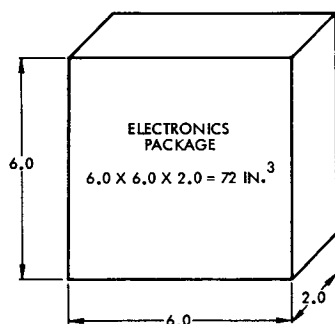
The magnetometer will have two ranges of measurement: 1) a low range of $\pm 7.25 \gamma$, and 2) a high range of $\pm 60.0 \gamma$. Data obtained during the Mariner IV fly by indicates the magnetic moment of Mars should be very low; however, in order to see variations in the field caused by solar flare activity or magnetic storms the above mentioned range scales are considered invalid.

Range switching and in-flight calibration of the instrument can be accomplished automatically within the experiment or by execution of a spacecraft command.

This instrument should be operated only after injection into a Martian orbit.

BIT	1	2	3	4	5	6	7	8	
L. R.	-4.0	-2.0	-1.0	-0.25	+0.25	+1.0	+2.0	+4.0	± 7.25
H. R.	-32.0	-16.0	-8.0	-4.0	+4.0	+8.0	+16.0	+32.0	$\pm 60 \gamma$

RANGE SCALES



SENSOR PACKAGE
(3)³ = 27 IN.³
TOTAL 72
99 IN.³ TOTAL VOLUME

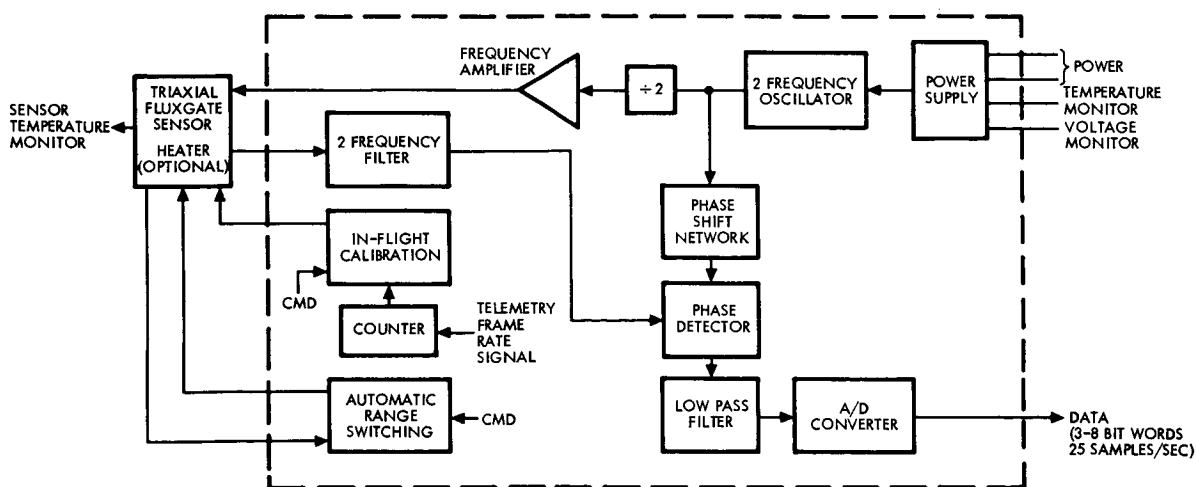


Figure 5-7

5.9 MICROMETEOR — IMPACT

5.9.1 Purpose

To measure the flux, momentum, energy, and spatial variation in the flux of micrometeoroids near Mars.

5.9.2 Instrument

The early micrometeoroid detectors usually provided, at best, only a count rate, or in the simplest form of a pressurized can, only a single event record. More recently, extensive efforts have been devoted to the development of more versatile sensors which will measure the significant parameters of mass and velocity or momentum and energy. One of these more sophisticated instruments, developed by W.M. Alexander for Pioneer C and D, is the instrument described here. It is a multi-sensor instrument involving relatively extensive data processing, and it is felt that it provides good probability of successful measurement for the important case of a Mars mission and a good example of optimum design for initial sizing of an illustrative payload.

The basic detector unit consists of three elements: 1) a front thin-film sensor, 2) a rear thin-film sensor, and 3) a piezoelectric crystal microphone immediately behind the second thin-film. Both the front and the rear thin-film sensors are fabricated in a matrix array in order to determine the micrometeoroid particle radiant. Penetration of the front and rear films provide timing signals for a time-of-flight measurement which determines the particle velocity, and because the rear film or impact plasma sensor completely absorbs the particle energy, its response signal measures a quantity related to the kinetic energy of the particle. In addition, the crystal microphone, or impact impulse sensor, is bonded directly to the rear film and its output signal is related to the momentum of the particle. Auxiliary sensors sensitive to mechanical vibration and electromagnetic noise (radiated and conducted) discriminate against false events.

The complete experiment consists of three of these detector units, one for each coordinate direction, together with a central electronics package. The three detector units can be mounted at different points on the spacecraft body, as long as they face the coordinate directions with unobstructed fields of view.

For particle velocities relative to the spacecraft in the most probable range of 4 km/sec to 72 km/sec, the detectors are sensitive to a mass range of approximately 10^{-11} gm to 10^{-15} gm. This range is typical of present impact detector designs; it corresponds to the cosmic dust thought to be responsible for the zodiacal light, and it is many orders of magnitude less than the mass range to which an atmospheric meteor flash detector responds. It is felt that both types of detectors play equally important roles in investigation of the micrometeor environment near Mars, and hence both types are included in this illustrative payload in order to cover adequately the meteoritic phenomena which are of such great importance to future manned space exploration.

Detailed interface data and schematic block diagrams for this experiment are given in Figure 5-8.

5.10 MICROMETEOR — FLASH DETECTOR

5.10.1 Purpose

Measure the flux of relatively large meteors (~ 0.1 gm) into the Martian atmosphere and to determine their mass and velocity spectra.

Measure the planet's night glow intensity in the two wavelength bands $\lambda 0.25\mu$ to $\lambda 0.30\mu$ and $\lambda 0.30\mu$ to $\lambda 0.45\mu$.

5.10.2 Instrument

During the past several years a continuing program of studies, both experimental and theoretical, in the field of micrometeoroid phenomena has been carried out within the Physical Research Center, TRW Systems Group. One result has been the development of a detailed design for a meteor flash analyzer; this instrument has been selected for discussion here.

Performance Characteristics

SPECTRAL RANGE

Mass - $10^{-15} \rightarrow 10^{-11}$ gm

Velocity (relative to spacecraft) - 4 $\rightarrow$ 72 km/sec

Momentum

$4 \times 10^{-10} \rightarrow 7.2 \times 10^{-5}$ gm cm/sec

Energy

$8 \times 10^{-5} \rightarrow 2.6 \times 10^2$ erg

SPECTRAL RESOLUTION - Not known

SPECTRAL SCAN - Not Applicable

DYNAMIC RANGE

Time of flight sensor 0.5 $\rightarrow$ 63.5 μ sec in 0.5 μ sec steps. Impulse impact sensor (crystal microphone) measures a quantity related to the momentum of the particle at impact over a range of 300:1, divided into 15 logarithmically spaced steps. The sensor output is a damped 100 kHz sine wave with a 100 μ sec envelope and voltage amplitude of 50 $\rightarrow$ 15 μ . Impact plasma sensor (rear thin-film matrix) measures a quantity related to the kinetic energy of the particle at impact over a dynamic range of 200:1 divided into 15 logarithmically spaced steps. The sensor output consists of two current pulses of opposite polarity, each having an amplitude of 5 $\rightarrow$ 1000 μ amp. The front thin-film matrix output consists of two current pulses of opposite polarity, with an amplitude of 0.3 $\rightarrow$ 5.0 μ amp.

POINTING

Orientation of the detector packages is not critical. Generally, one unit should view the ecliptic plane, the remaining units should be positioned to view at right angles to the first unit (see note* under interlaces)

ALIGNMENT

Orientation of the detectors, specified above, should be known to within ± 1 Deg.

FIELD OF VIEW

2 π steradians

Interfaces

ELECTRICAL

Commands

Power - ON/OFF

Calibrate

Multiplexer status

Telemetry

*Data

4 - 8-Bit words
(subcom-low rate)

*Diagnostics

Electronics package temperature

Vibration level sensor

Power supply voltage monitor

*Power

3 watts

MECHANICAL

*Weight

5.5 lb

*Dimensions

6.00 x 6.00 x 6.00

*Volume

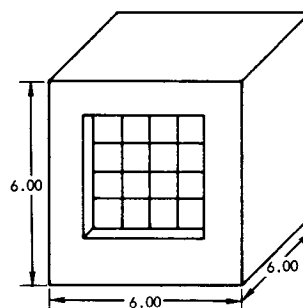
216 cu.in.

THERMAL

Operating Temperatures

-20 $^{\circ}$ $\rightarrow$ +65 $^{\circ}$ C (in package)

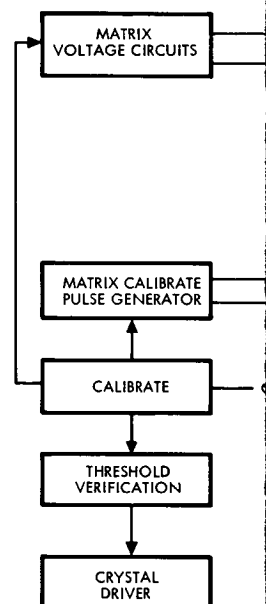
-80 $^{\circ}$ $\rightarrow$ +150 $^{\circ}$ C (for exposed surfaces)



*NOTE: THIS EXPERIMENT REQUIRES 3 IDENTICAL UNITS (SEE ABOVE) PLACED TO VIEW 3 MUTUALLY PERPENDICULAR AXES. WHEREVER INDICATED BY * MULTIPLY BY 3 TO OBTAIN VALUES FOR TOTAL EXPERIMENT.

SA = SATURATED AMPLIFIER

CA = CHARGE SENSITIVE AMPLIFIER



5-21A



PRIMARY SPECIFICATION

for Detector-Impact

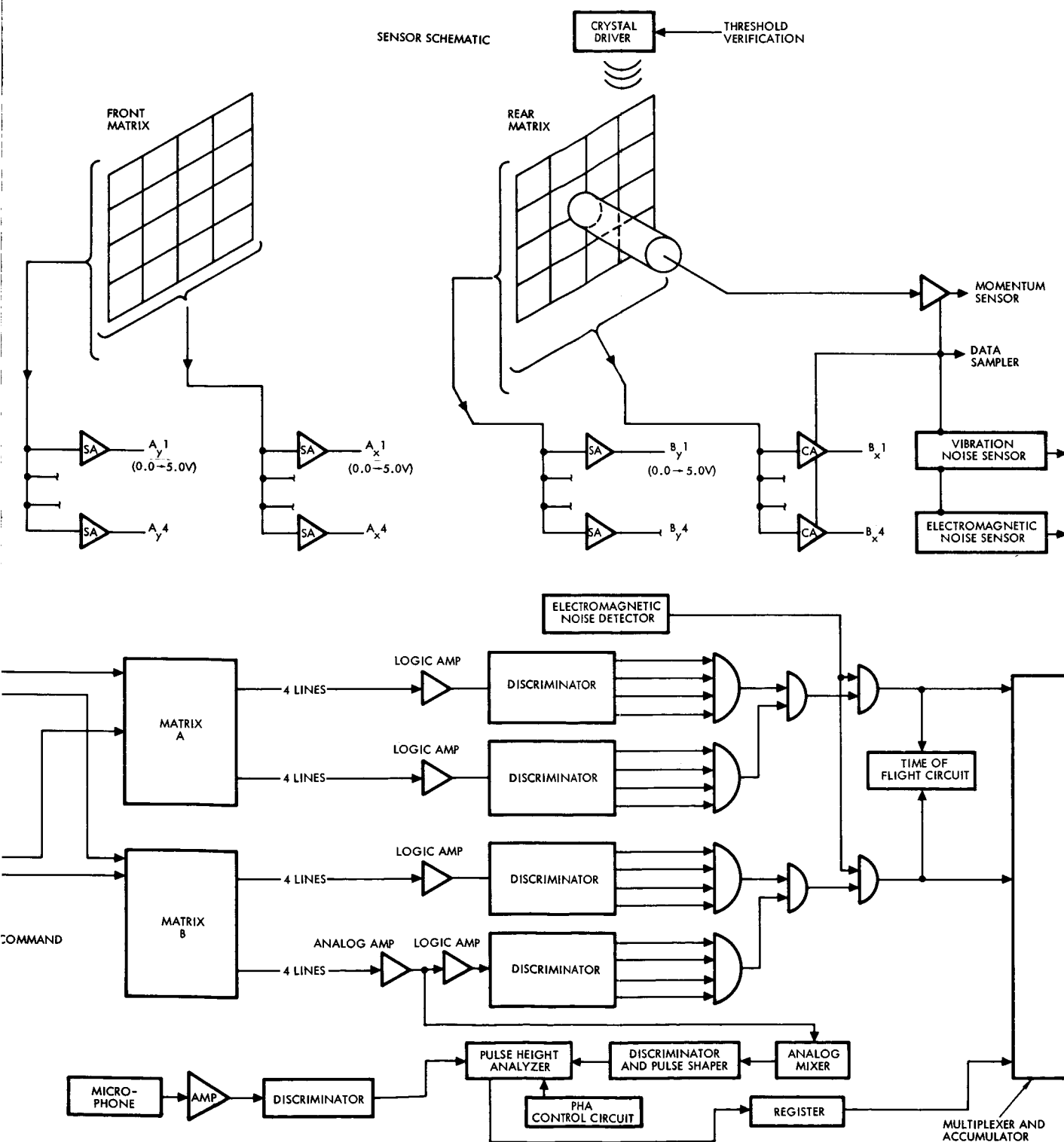


Figure 5-8

The instrument is a two-channel photometer which measures both integrated and maximum meteor flash intensity. Theory shows that the former is proportional to mV^3 , where m is the mass and V is the entry velocity of the meteor into the planetary atmosphere, and that the ratio of the maximum to the integrated intensity is proportional to $V \cos \theta / H$, where θ is the entry angle and H is the scale height of the atmosphere. Hence, a velocity spectrum can be deduced from the observed $(V \cos \theta)$ distribution if enough data points are available to allow statistical conclusions. Calibration is to be carried out in the earth's atmosphere, and the distribution of θ for Mars is expected to be identical to that observed on earth. Then, if the scale height is known independently, both mass and velocity spectra can be deduced.

Two separate assemblies each consisting of a quartz optical system, photomultiplier detector, and electronics are provided, one for the "photographic" range, 0.30μ to 0.45μ , for correlation with astronomical data on meteors, and the other for the important ultraviolet range from 0.25μ to 0.30μ where the detection sensitivity is expected to be a maximum because of optimum signal-to-noise ratio. An added feature of the instrument is that it permits continuous measurement of the planet's night glow in these two wavelength bands.

An important characteristic of this type of instrument is that it measures a regime of meteors larger than that for which present impact detectors give useful data, because of the infrequency of larger events. The minimum mass that can be detected with the atmospheric flash detector depends on the entry velocity, but for a "typical" velocity of 30 km/sec it is approximately 9×10^{-2} gm for the 0.30 - 0.45μ channel and 1.6×10^{-1} gm for the ultraviolet channel and thus is intermediate between the small particles, $\sim 10^{-12}$ - 10^{-14} gm, usually detected by impact, and the much larger range of terrestrial meteorites. Since the flash detector views the entire half-atmosphere of the planet, useful count rates are obtained for the large particles detected; design estimates of the order of 10 minutes/meteor have been made.

The instrument parameters and requirements are summarized in Figure 5-9. It may be mentioned that the most recent design proposal has added a third channel farther in the ultraviolet, from 0.20μ to 0.26μ , and if desired this can be incorporated with only slight increases in size, weight, and power. The 3-channel instrument has been proposed for the initial exploratory and calibration mission for the earth's atmosphere. It is felt that the simpler instrument is more appropriate for consideration for the initial Mars mission, at least until experimental evidence to the contrary is available.

5.11 SOLAR PLASMA

5.11.1 Purpose

To measure the charged particle flux in the solar wind near Mars; specifically to determine the energy spectra and the time-variations of the flux of low energy electrons, protons, and heavy ions from the sun.

5.11.2 Instrument

The instrument described for this experiment is a curved-plate electrostatic analyzer employing channel multipliers as collectors. This instrument is based on the principle that, for a given charged particle species, only those particles with energy lying within a narrow range determined by the applied voltage will be passed through the analyzer to the collector, and thus a differential energy spectrum is obtained. Separate channels are provided for electrons and for protons and heavier positive ions. The instrument count rate has an upper limit of the order of 10^6 counts $\text{cm}^{-2} \text{sec}^{-1}$.

The energy spectrum is divided into 16 discrete energy bands. The energy resolution, defined as the ratio of the full width at half-maximum to the midpoint energy of the channel, can vary from approximately 4 percent to a maximum of 10 percent, depending on the analyzer voltage for each energy band.

Instrument characteristics are given in Figure 5-10.

PRELIMINARY SPECIFICATION

Micrometeor Detector-Light Flash

Performance Characteristics

SPECTRAL RANGE

Channel 1 0.25 → 0.30 μ
Channel 2 0.30 → 0.45 μ

SPECTRAL RESOLUTION

Radiation for the channel No. 1 and No. 2 detectors will be determined by the filters (filter combination) and cannot be further resolved at this time.

SPECTRAL SCAN

None

DYNAMIC RANGE

Detector (PMT) output: 1.3×10^{-9}
Amp → 1.3×10^{-6} amp at an average noise level of 2.6×10^{-10} amp
(Dynamic range is 10^3 at S/N = 5) or 10^4 times the light intensity at the limit of detectability.

SENSITIVITY

Channel No. 1 1.6×10^{-1} gm
Channel No. 2 9.2×10^{-2} gm
(With respect to a meteor with a 30 km/sec entry velocity).

ACCURACY

Not known

POINTING

Views the center of the planet to within ± 3.0 deg

ALIGNMENT

Not critical, pointing direction above should be known to within ± 1.0 deg

FIELD OF VIEW

$2\theta = 60$ deg

Interfaces

ELECTRICAL

Commands

Power — ON-OFF

Telemetry

Data

2 — 8-Bit words

Diagnostics

Sensor temperature monitor (1 and 2)

Dynode voltage monitor (1 and 2)

Electronics package temperature

Low voltage power supply voltage monitor

Power

6.0 watts

MECHANICAL

Weight

8.5 lb

Dimensions

13.00 x 7.00 x 7.00

Volume

637 in.³

THERMAL

Operating Temperature

0.0°C → +45°C

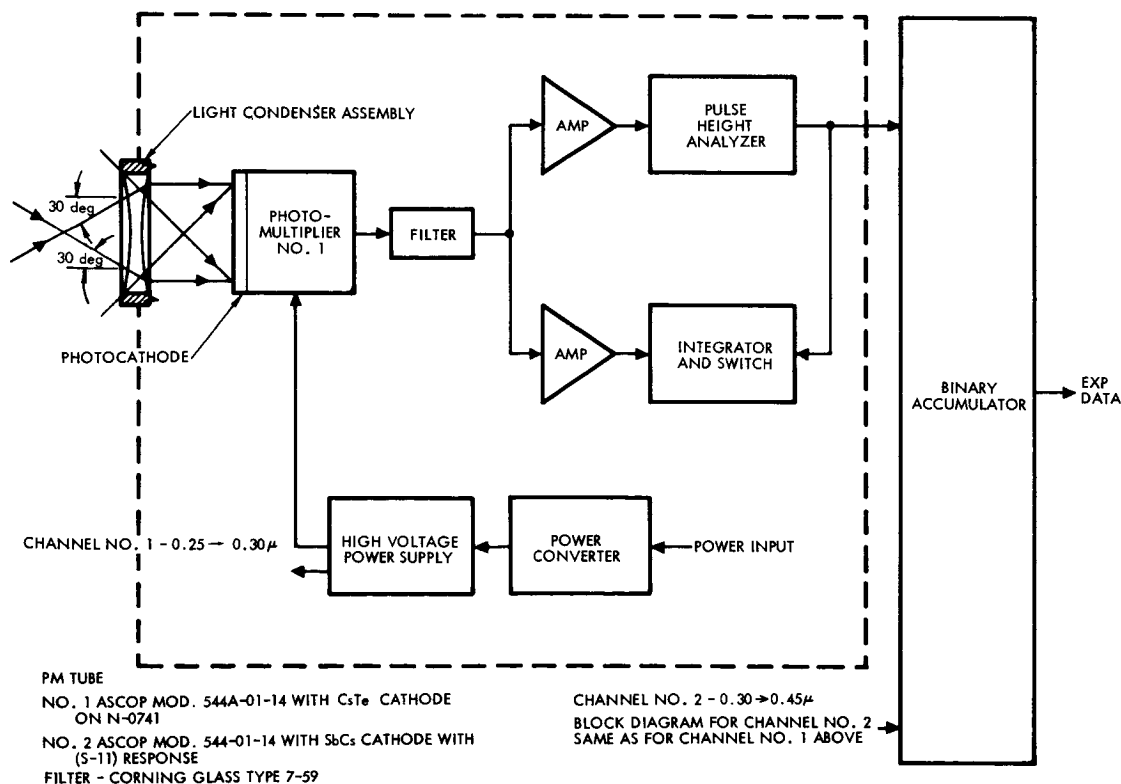


Figure 5-9

PRELIMINARY SPECIFICATION

Solar Plasma

Performance Characteristics

ENERGY RANGE

5.0 eV - 70.0 KeV

ENERGY RESOLUTION

The above energy spectrum shall be divided into 16 discrete energy bands

ENERGY SCAN

Each energy band shall be scanned for a period of 5.0 sec low energy bands shall be scanned first and then sequentially switched to the higher bands - upon completion of the scan of all bands the ranges will be stepped down sequentially.

DYNAMIC RANGE

3.0×10^5 cts/sec

ACCURACY

4% - 10% for energy resolution - dependent upon the geometry of the curved plate analyzer and the particle path length.

POINTING

Centerline of viewing aperture shall be parallel to sun line (within $\pm 1.0^\circ$).

FIELD OF VIEW

Instrument collimator look angles

9.0° Perpendicular to ecliptic

25.0° Parallel to ecliptic

ALIGNMENT

Shall be known to within $\pm 0.5^\circ$

Interfaces

ELECTRICAL

Commands

Power - ON-OFF

Multiplexer - Status change

TELEMETRY

Data

2 - 8-Bit words

Diagnostics

Multiplexer status

Step - Voltage monitor

Power supply voltage monitor

Electronics package temperature

Power

6.0 Watts

MECHANICAL

Weight

5.0 lb

Dimensions

6.00 x 7.00 x 8.00

Volume

336.0 In.³

THERMAL

Operating Temperature

-20°C + 60°C

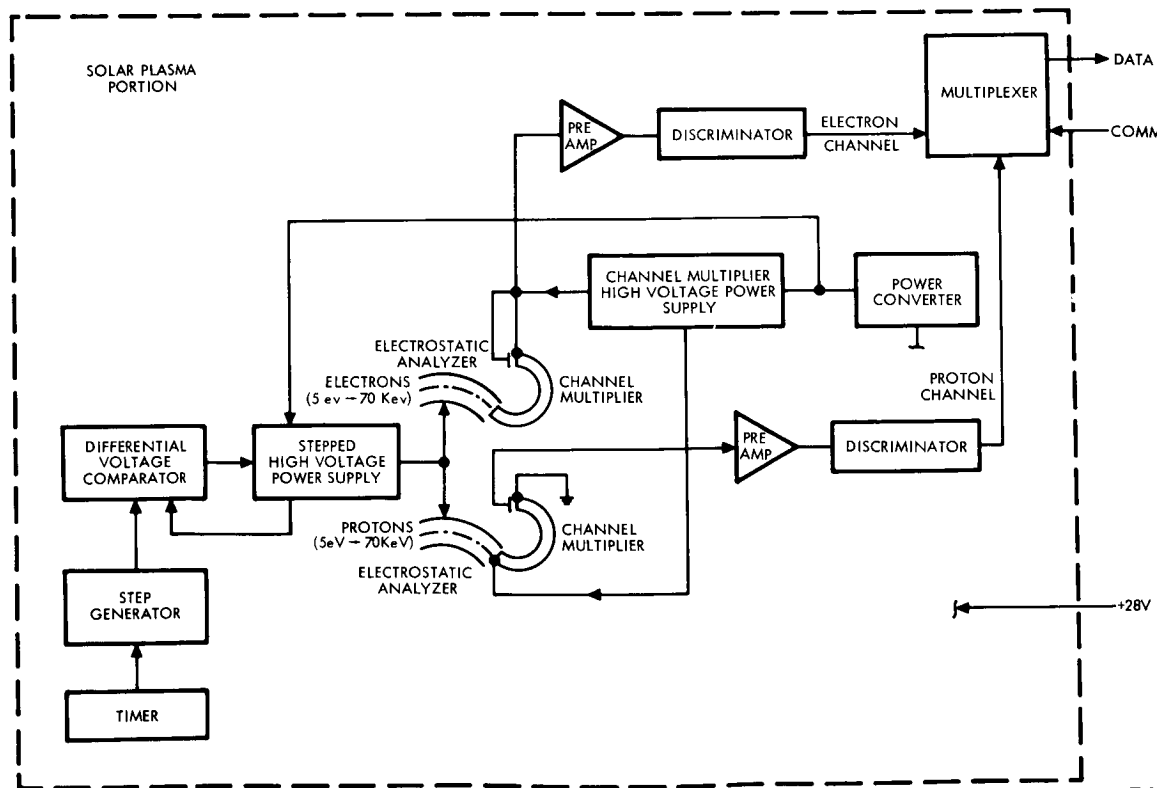
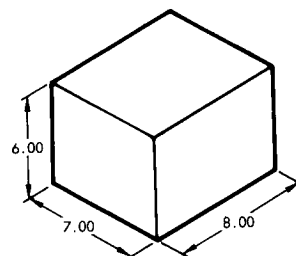


Figure 5-10



The instrument should be aligned to view the sun directly to within ± 1 degree. It may be noted that internal light baffles and absorbers are an integral part of the design of these curved plate analyzers. The aperture has a field of view of $9^\circ \times 25^\circ$ and the instrument should be oriented so that the 25 degree look angle is approximately parallel to the ecliptic plane.

5.12 SOLAR X-RAY

5.12.1 Purpose

The purpose of this experiment is to measure the integral flux and temporal variations of x-rays generated during periods of solar activity.

5.12.2 Instrument

This instrument employs four Ionization Chambers to obtain data over four discrete energy bands. The spectral ranges cover the following bands:

- 1) $0.5 \text{ \AA} - 3.0 \text{ \AA}$
- 2) $2.0 \text{ \AA} - 8.0 \text{ \AA}$
- 3) $3.0 \text{ \AA} - 16.0 \text{ \AA}$
- 4) $44.0 \text{ \AA} - 60.0 \text{ \AA}$

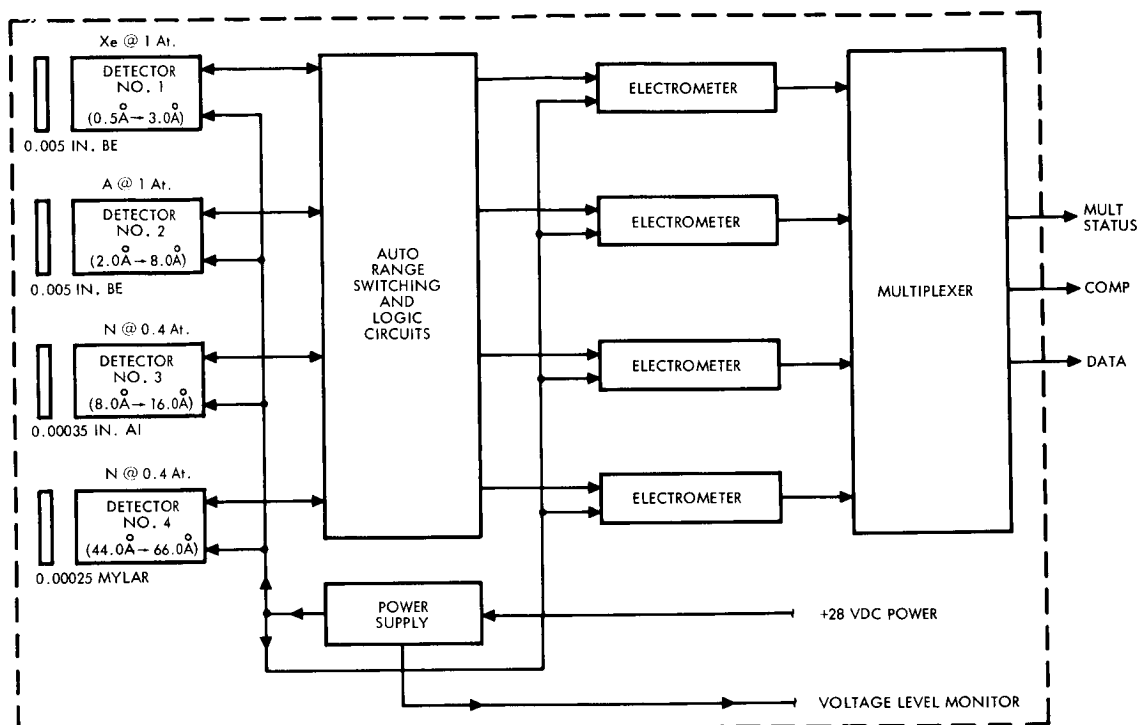
Selection of the "fill gas" and internal pressurization level as well as the window material and thickness result in an integral energy spectrum over a specific band.

Automatic range switching is included to select data from the highest energy band. The output from the electrometer(s) is A to D converted and subsequently stored in the multiplexer allowing a delayed readout.

Figure 5-11 gives experiment characteristics.

PRELIMINARY SPECIFICATION

Solar X-Rays



Performance Characteristics

SPECTRAL RANGE

Detector No. 1	0.5Å → 3.0Å
Detector No. 2	2.0Å → 8.0Å
Detector No. 3	8.0Å → 16.0Å
Detector No. 4	44.0Å → 66.0Å

DYNAMIC RANGE

>10⁵ (Radiation Intensity)

POINTING

Detectors shall be positioned to view the sun throughout the orbit. (Normal to the sun within ± 1.0°)

ALIGNMENT

0.5° (with respect to the above)

FIELD OF VIEW

0.1 steradian

Interfaces

ELECTRICAL

Commands

Power — ON-OFF

Telemetry

Data

2 — 8-Bit words

Diagnostics

Sensor (1 thru 4) Temperature monitor

Electronics package temperature

Power supply voltage level monitor

Power

10.0 Watts

MECHANICAL

Weight

7.5 lb

Dimensions

7.50 x 11.50 x 6.50 In.

Volume

~ 600 In.³

THERMAL

Operating Temperature

+ 5.0°C + 45.0°C

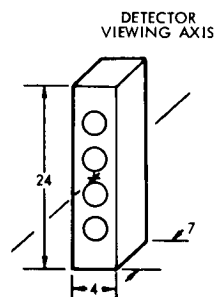


Figure 5-11



The individual detectors have a field of view of 0.1 steradian, and should be aligned to view the sun within ± 1 degree throughout the orbit of vehicle.

5.13 CELESTIAL X-RAY

5.13.1 Purpose

To detect and measure x-rays from galactic sources and also the weak nearly isotropic background flux which has been attributed to extragalactic origin.

5.13.2 Instrument

The discovery of discrete cosmic x-ray sources, notably the Crab Nebula, and also of the diffuse background flux, is very recent and their existence is of great interest to cosmologists and astrophysicists. Instrumentation for the study of this radiation is difficult because the flux is weak and the most significant spectral range is that of very soft x-rays. The detectors must be as large as possible, and have windows as thin as possible - a difficult combination - and spectral information, polarization data, and knowledge of the incident direction are all desirable. Large rectangular Geiger counters with thin Mylar windows of various thicknesses and large honeycomb baffles have been flown on Aerobee rockets by H. Friedman and colleagues at NRL, yielding crude spectral data, and a few proportional counter pulse height spectra have been obtained. Ambitious plans have been proposed for ultra-large x-ray telescopes (1000 sq. ft. aperture!) for the Apollo Extension Program, and projects of intermediate scope are also under active discussion.

Celestial x-rays are not characteristic of Mars or of the Martian environment, and their measurement may have a lower priority in the Voyager program than experiments which measure indigenous phenomena. For this reason and also because of the instrumental difficulties, the celestial x-ray experiment described here has been defined in somewhat less detail than have the others in the present group of experiments.

A proportional counter has been described, in order to provide detailed spectral data. Since high-gain amplification and an ultra-stable power supply are required for such a detector, the corresponding spacecraft requirements are thus typical of an advanced instrument and are suitable for an illustrative payload. A single counter has been used since the size can not really be defined at present and hence the mounting position is uncertain. The detector should look in or near the ecliptic plane to observe galactic sources and in the generally anti-solar direction to minimize detection of scattered solar x-rays. The lander capsule biological shield prevents direct anti-solar viewing. and it is felt to be satisfactory to have a single detector which views the ecliptic plane at right angles to the spacecraft-sun line, and hence slowly scans the galaxy (and outer space) throughout the Martian year. See Figure 5-12 for instrument characteristics.



PRELIMINARY SPECIFICATION

Celestial X-Ray Experiment

Performance Characteristics

SPECTRAL RANGE

0.1—15Å

SPECTRAL RESOLUTION

Unknown - dependent upon the experimenter's requirement

SPECTRAL SCAN

Not required

DYNAMIC RANGE

Not applicable

POINTING

Detector shall be positioned 90 deg to the solar x-ray sensor (coplanar)

ALIGNMENT

Position of detector specified above shall be known to within ± 2 deg

FIELD OF VIEW

$2\theta = 20$ deg

Interfaces

ELECTRICAL

Commands

Power - ON-OFF

Telemetry

Data

3 - 8-Bit words

Diagnostic

Dynode voltage monitor

Detector temperature monitor

P.S. voltage monitor

Electronics package temperature

Power

1 watt

MECHANICAL

Weight

Detector package - 2.0 lb

Electronics package - 2.5 lb

Dimensions

Detector package - 2.00 Dia x 12.50

Electronics package - 6.00 x 6.00 x 2.00

Volume

Detector package - 38.25 cu. in.

Electronics package - 72.0 cu. in.

THERMAL

Operating Temperatures

Detector package | -20° → +70°C

Electronics package

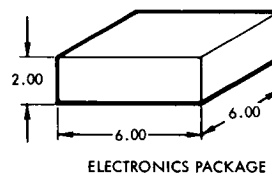
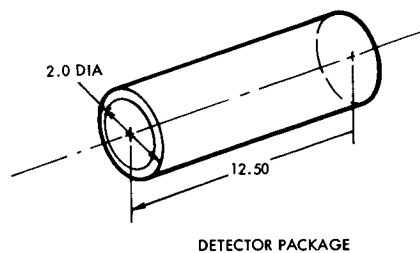
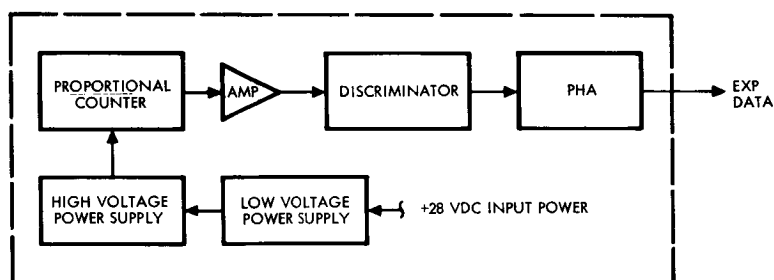


Figure 5-12

6. INTEGRATION OF ADDITIONAL EXPERIMENTS

6.1 UPGRADING CONCEPT

In Section 2, a hypothetical science payload was defined which consists of four core experiments and an all-television photo-imaging system. Section 3 described the integration of this hypothetical science payload on the recommended spacecraft. In Section 4 a recommended Film/TV alternate to the all-television system was proposed and the integration of this system with the recommended spacecraft was described. Section 5 describes an extensive list of additional experiments which could be carried on the recommended spacecraft as complements to the four core experiments. The planetary scan platform for the recommended spacecraft was configured with this type of growth in mind and as described below in Section 6.2, three of these additional experiments (which must view the planet) can be accommodated in the planetary scan platform without changing the overall dimensions of the box, without changing the planetary scan platform shaft structure and without changing the planetary scan platform control system and drives. Additional instruments not on the recommended spacecraft can also be accommodated in two ways: 1) booms will be provided for sensors which must be extended from the main body of the spacecraft, and 2) other mounting locations around the spacecraft periphery or the main equipment mounting panels provide adequate area for attaching body-fixed experiments.

The recommended spacecraft subsystems were configured with growth capability to accommodate additional individual instruments or the total list of additional experiments. Given the additional weight allocation and by adding additional solar cells or altering the experiment duty cycles these instruments can be supported as required by mission objectives. Section 6.2 describes the detailed integration of these experiments on the recommended spacecraft and identifies the changes in the basic interfaces required to accommodate the additional experiments.

6.2 INTEGRATION OF ADDITIONAL EXPERIMENTS

6.2.1 Mechanical Integration

6.2.1.1 Planetary Scan Platform Mounted Experiments

Figure 4-4 illustrates the basic box geometry for the planetary scan platform configuration supporting additional experiments. It is similar to the planetary scan platform configuration discussed in Section 4.0 primarily because it contains the same recommended film/TV photo-imaging system. The experiment list differs somewhat however and Table 6-1 tabulates the additional experiments proposed to be placed in the planetary scan platform and their mechanical interface requirements. Figure 6-1 shows the planetary scan platform configuration when the three additional experiments are added.

6.2.1.2 Boom Mounted Experiments

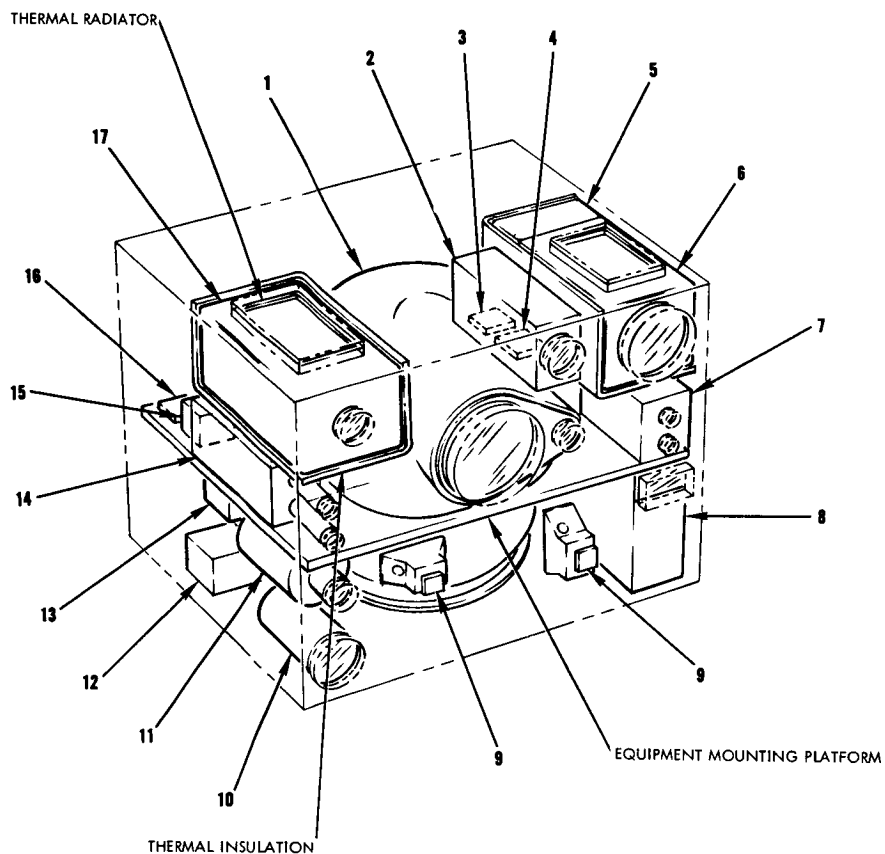
Two of the additional experiments require mounting on booms away from the main body of the spacecraft. These are the magnetometer and the neutron albedo experiments. It is estimated that the magnetometer sensor will be sufficiently far from the spacecraft magnetic field if mounted approximately 20 feet from the central body of the spacecraft. Booms for supporting these instruments are described below.

6.2.1.3 Magnetometer and Neutron Albedo Experiment Booms

The magnetometer and neutron albedo experiment boom structure and deployment mechanisms are identical. They are located on the spacecraft as shown in Figure 6-2. These booms are like those on the OGO spacecraft with two improvements made which will be described later.

The booms are comprised of the following components:

- Two tubes
- Two hinge mechanisms
- Two support cradles
- Ordnance release device



- | | |
|---|--|
| 1. FILM CAMERA SYSTEM | 10. GAMMA RAY SPECTROMETER |
| 2. INFRARED RADIOMETER | 11. LOW RESOLUTION COLOR TV CAMERA |
| 3. PSP REMOTE PHOTO-IMAGING MULTIPLEXER | 12. GAMMA RAY SPECTROMETER ELECTRONICS |
| 4. PSP REMOTE MULTIPLEXER UNIT | 13. LOW RESOLUTION COLOR TV CAMERA ELECTRONICS |
| 5. CRYOGENIC REFRIGERATOR | 14. POLARIMETER |
| 6. BROADBAND INFRARED SPECTROMETER | 15. PSP REMOTE SCIENCE SWITCHING ASSY. |
| 7. MICROMETEOR DETECTOR — LIGHT FLASH | 16. PSP REMOTE DECODER UNIT |
| 8. ULTRAVIOLET SPECTROMETER | 17. HIGH RESOLUTION INFRARED SPECTROMETER |
| 9. MARS SENSOR (2 PLACES) | |

FIGURE 6-1

ADDITIONAL EXPERIMENTS ON PSP are shown with the basic core group of experiments and the Film/TV photo-imaging system. Note that the basic box configuration has not changed from the recommended PSP.

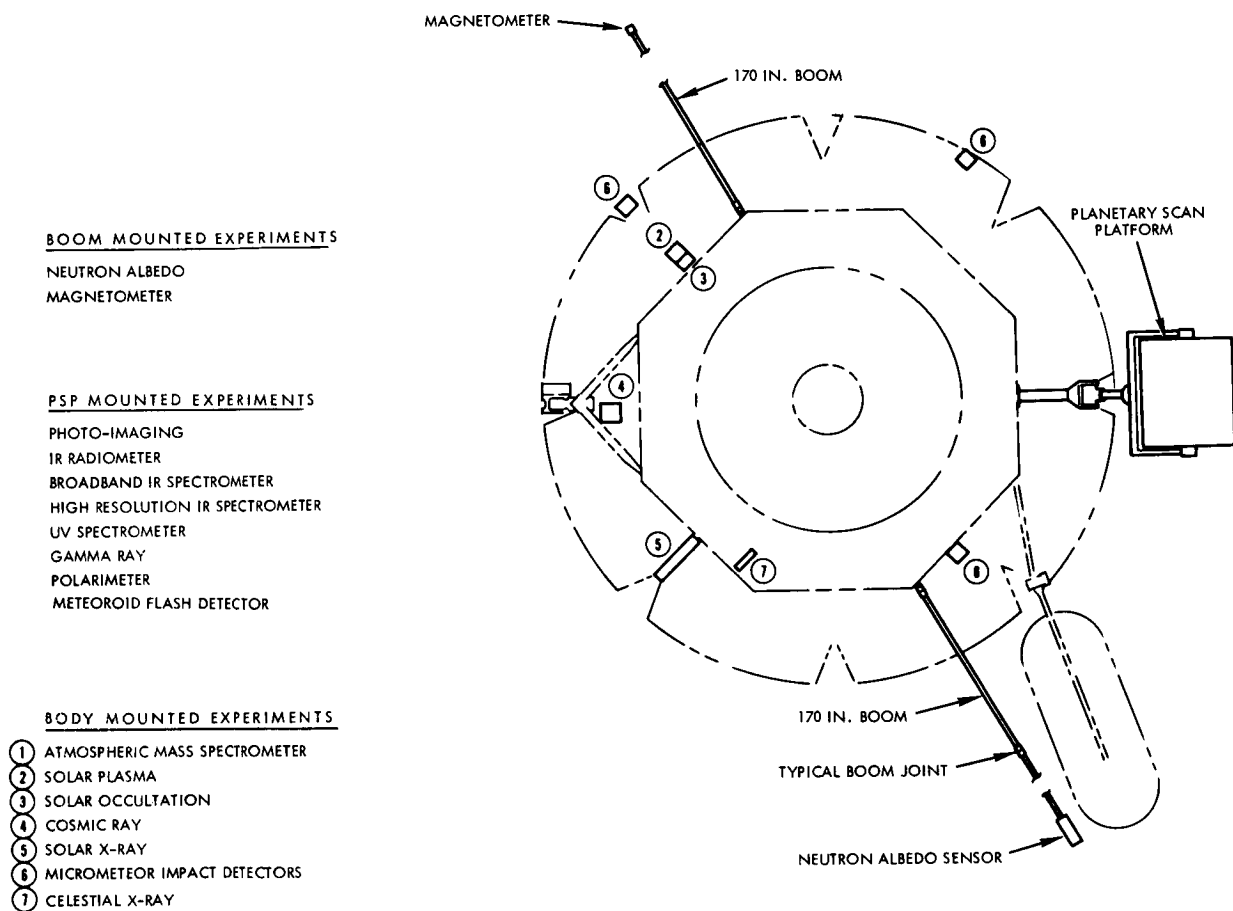


Figure 6-2

SIXTEEN EXPERIMENTS and a photo-imaging system can be accommodated on the recommended spacecraft without change in the basic configuration.

Table 6-1. Planetary Scan Platform Mechanical Interface Requirements, Core Experiments
plus Three Additional Experiments

Experiment	Sensor				Colocated Electronics	
	Field of View (deg)	Alignment (deg)	Aperture (in.)	Volume (in. ³)	Weight (lb)	Weight (lb)
UV spectrometer	2 x 6	0.1	4 x 4	1,536	20	10
Broadband IR spectrometer	5	0.5	10 diameter	1,848	20	-
High-resolution IR spectrometer	1 x 4	0.5	4 diameter	2,400	30	-
IR radiometer	120	0.1	5 diameter	220	10	10

Gamma ray spectrometer	45 sq	5.0	4 diameter	180	4	3.6
Meteoroid flash detector	1.5	0.5	2.5 diameter (3)	432	6	2
Atmospheric polarimeter	2.5 sq	0.5	2 diameter (2)	800	6	3

Mars sensors (2)	45 each side	0.05	8 x 3	70 each	5.5 each	6.0 each

Note:
Sensors are boresight mounted
Sensors require protective covers for micrometeorite erosion and thermal control
Sensors must not face the sun



6.2.1.4 Tubes

The boom is made up of two tubes hinged together to form a 170 inch long boom. Both tubes are 2-in. outside diameter and 0.028 in. wall 6061-T6 aluminum tubing. The size was determined by choosing the torque output of the deployment springs and then calculating the tube size required to absorb this energy in bending when the boom is deployed. A constant section is used because a tapered section would have less rigidity in the stowed position and the wall thickness is considered a minimum practical limit for this diameter. An additional boom section may be added if required. A detailed magnetics analysis of the specific spacecraft configuration will be performed to determine the required length.

6.2.1.5 Hinge Mechanisms

The hinge mechanisms consist of two hinge halves, two redundant drive springs, and a latch spring. The boom is supported from the edge member of the upper meteoroid panel and one of the longerons. The support bracket is a machined aluminum fitting which also serves as one hinge half. The hinge design is essentially the same as the OGO boom hinge design except that the hinge halves provide larger bearing surfaces at the pivot pin. The drive spring torque has been increased for more positive deployment and the spring diameters have been increased to improve coil clearance and minimize friction. The drive springs are spiral wound power springs and provide a total of 53 inch pounds of torque at the start of deployment and 40 inch pounds with the boom deployed.

The latch spring is a leaf spring attached to one hinge half. It has a rectangular hole in it which latches over a tong on the other hinge half. In latching, it rides up a sloped surface on the tong to take out the "slop" between the hinge halves. The stop for the deployed position is provided by machined surfaces on each hinge half.

6.2.1.6 Support Cradles and Secondary Release

There are two support cradles, one just above the upper hinge and the other at the lower hinge. The upper cradle is mounted from the face

of the release mechanism which is supported by a second bracket as shown in Figure 6-3. The cradle houses a kickoff spring to aid the initial separation.

The lower cradle is a machined aluminum fitting supported from a side panel of the spacecraft just above the solar array as shown. It not only serves as a cradle to support the boom but also supports a roller to which the boom is latched in the stowed position. The latching is accomplished by a hook mounted to the outboard hinge half as shown in Figure 6-3. After the ordnance release device is fired to initiate deployment the outer section of the boom rotates through 30 degrees at which time the hook is released from the roller to allow the inboard section of the boom to release.

6.2.1.7 Ordnance Release Device

The release device for the booms is the same device used on all other appendages. The description of this device is found in Volume 3.

6.2.1.8 Spacecraft Body Fixed Science Experiments

A number of nondeployable, body-fixed science experiments are mounted to the spacecraft's equipment module. Some of them are located internally and are mounted to the equipment panel doors where they are afforded the same structural and environmental support as the other spacecraft electrical equipment. Due to particular look angles and fields-of-view requirements, other science experiments are located externally to the equipment compartment forward of the annular solar array. These experiments are rigidly mounted to the equipment module structure — they are not physically attached to the annular solar array although they are located as close as possible to it for minimum interference with opened equipment mounting panel doors. Those sensors with 0 deg cone look angles (toward the sun) are oriented with their lines of sight parallel to the spacecraft roll axis and are located in areas where holes may be provided in the annular solar array with no loss of cells. Adequate areas which are free of cells are found at the array periphery and at the junctions of the eight solar panels which comprise the array.

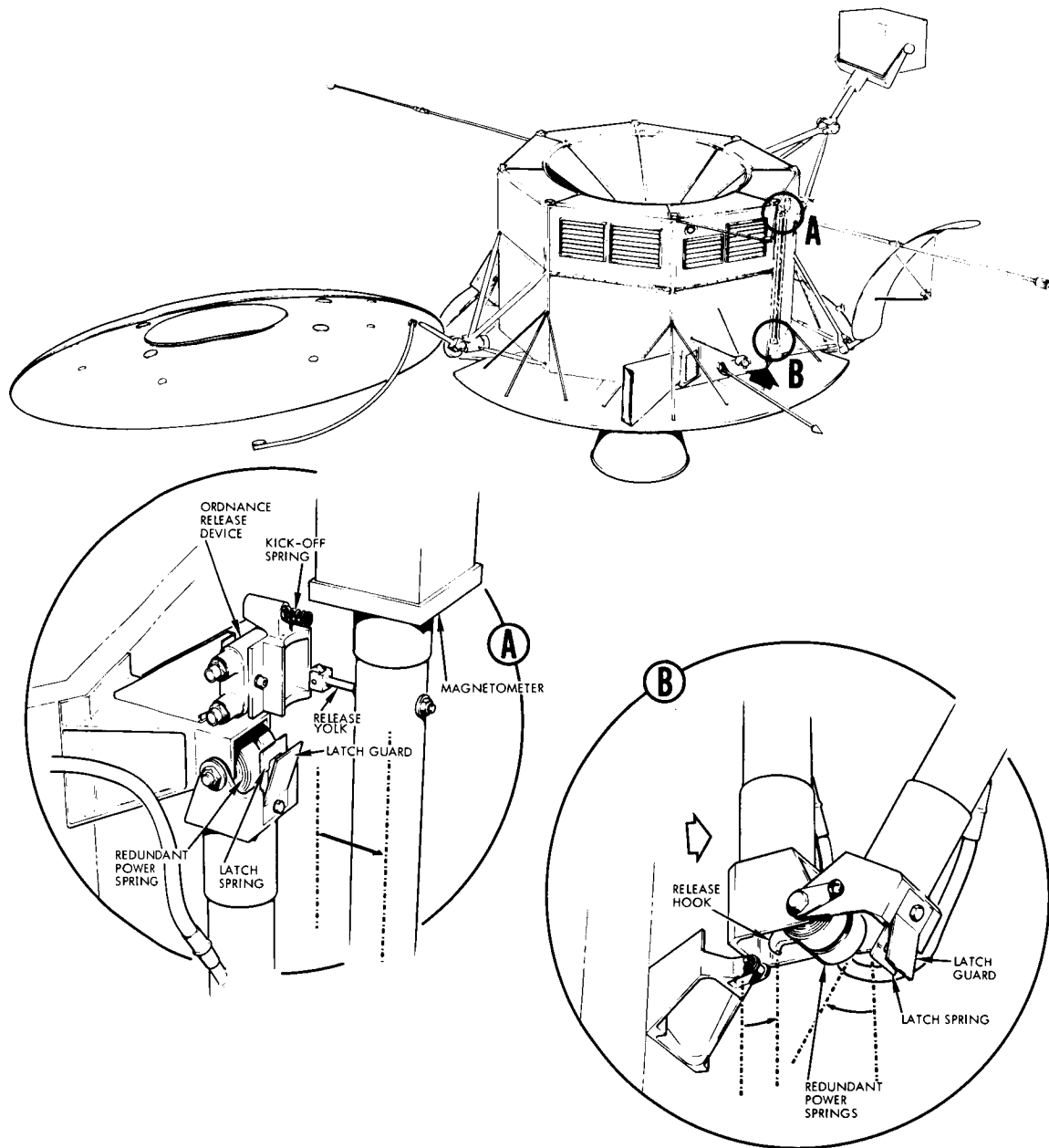


Figure 6-3
BOOMS FOR MAGNETOMETER and Neutron Albedo Experiments are released pyrotechnically and deployed by springs.



Internally and externally mounted experiments are provided with their minimum required look angles, fields of view, isolation and alignment accuracies. They are provided with adequate structural and electrical support, micrometeoroid protection and temperature control. Where required the experiments will be provided with adequate protective aperture covers and glint shields. Sensors and directly associated hardware are packaged together with consideration given to the effect on structure, meteoroid protection and temperature control. The remote hardware associated with experiment sensors is located on the equipment module's equipment panel doors.

Body fixed experiments and locations are described below and information relative to them is tabulated in Table 6-2.

- 1) The atmospheric mass spectrometer is located so as to point in the general direction of the spacecraft velocity vector near periapsis. It is bracket mounted from Panel I and located as far outboard as possible for maximum field-of-view.
- 2) The solar occultation experiment is located adjacent to Panel I and faces the sun in order to view the sun through the Martian atmosphere during eclipse periods.
- 3) The celestial x-ray experiment is bracket mounted on equipment Panel No. 3 inside the equipment module, at a clock angle of 335 deg and having a cone angle of 90 deg.
- 4) The solar plasma experiment views the sun and is located at Panel I immediately adjacent to and outboard of the solar occultation experiment. Both experiments are mounted to the same support bracketry.
- 5) The solar x-ray experiment has four sun-looking apertures and a hole in the annular solar array is provided for each aperture. The experiment is located forward of the solar panel joint at Panel III.
- 6) Micrometeor Impact Detectors. The three micrometeor impact detectors are located as follows: No. 1 is bracket mounted to the side of equipment panel No. 5 with 0 deg cone angle. No. 2 is bracket mounted to the side of equipment panel No. 1 at the edge of the solar array with clock angle of 65 deg and 90 deg cone angle. No. 3 is bracket mounted to the canopus sensor "B" glint shield at the edge of the solar array with clock angle 155 deg and 90 deg cone angle.

Table 6-2. Mechanical Interface Requirements, Body Fixed and Boom Mounted Science, Upgraded Spacecraft

Experiment	Sensors							Remote Hardware		
	Number Required	Total FOV	Alignment (deg)	Orientation Angles		Deployment	Volume (in. ³)	Weight Lbs	Volume (in. ³)	Weight Lbs
				Clock	Cone					
1. Atmospheric mass spectrometer	1	2 π steradians	$\pm 3.0^\circ$	50°	120°	NA	612	8.5	NA	NA
2. Solar occultation	1	$\pm 0.05^\circ$ max.	$\pm 0.1^\circ$	NA	0°	NA	504	18.0	NA	NA
3. Celestial x-ray	1	2 θ = 20°	$\pm 2.0^\circ$	335°	90°	NA	39.2	2.0	72	2.5
4. Solar plasma	1	$\pm 4.5^\circ$ parallel to ecliptic $\pm 12.5^\circ$ perpendicular to ecliptic	$\pm 0.5^\circ$	NA	0°	NA	336	5.0	NA	NA
5. Solar x-ray	1	0.1 steradians	$\pm 0.5^\circ$	NA	0°	NA	672 (assumed)	7.5	NA	NA
6. Micrometeoroid impact detector	3	2 π steradians	$\pm 1^\circ$	335° 245° 245°	90° 90° 0°	NA	216 each	5.5	NA	NA
7. Cosmic ray	1	$\pm 27.5^\circ$	$\pm 10.0^\circ$ (assumed)	NA	0°	NA	896	22	NA	NA
8. Neutron albedo	1	2 π steradians	2.0%	260°	90°	14 ft boom	198	5.5	244	6.5
9. Magnetometer	1	4 π steradians	$\pm 1-3^\circ$	80°	90°	14 ft boom	27	0.5	72	2.5
Note: NA denotes not applicable										

- 7) The cosmic ray experiment is mounted adjacent to Panel II and has a single sun-looking aperture.

6.2.2 Electrical Integration

The electrical integration of the additional experiments described in Section 5 takes two forms – one for planetary scan platform-mounted experiments, and another for those attached to the spacecraft body, either directly or by extendible booms. In the former the same techniques will be utilized as described in Section 3.3.1, regarding the hypothetical payload. The individual planetary scan platform-mounted experiment electrical interface requirements will be satisfied by the design of the interconnect cable between experiments and the remote decoding, switching, and multiplexing equipment contained within the planetary scan platform. The individual body mounted experiment electrical interface requirements will be satisfied by the design of the interconnect cable between the experiments and the electrical integration assembly (EIA). (See Section 6.2.2.3)

A special case among the additional experiments is the Cosmic Ray Detector, added to obtain correlation between production of albedo neutrons as a function of cosmic ray flux during solar flare events. The operation of the experiment will be continuous during initial orbital operations but may at some later time be operated only during period of solar flares. Subsequent discussions in regard to Science Data Handling, Science Sequencing and Electrical Integration Design do not include the detailed requirements for the Cosmic Ray Detector; however, flexibility of the various interfaces is such that its requirements can easily be accommodated. These are:

- Command – (4) discrete
- Telemetry – (6) 8-bit words
- Power – 5 watts

The following subsections cover:

- Science data handling
- Science sequencing
- Electrical integration design

6.2.2.1 Science Data Handling

Data Requirements. A telemetry list for the payload with all additional experiments is contained in Appendix B, Table 3. The list is considered to be typical of those experiments listed. The total number of data channels including primary science and engineering required for the core and additional experiments is 114 analog and 26 digital, as compared to 53 analog and 8 digital for just the core experiments.

A summary of orbital primary science data estimates for the additional experiments is contained in Table 6-3. The data acquisition assumptions are consistent with the science time lines for a noneclipse orbit illustrated in Section 6.2.2.2. The buffered data rates represent reasonable demands based on expected scientific requirements alone.

Data Formatting. The additional data handling equipment required for the payload with all additional experiments consists of a multiplexer under the control of the flexible format generator (FFG) for the body mounted experiments. The data handling interface requirements for the body mounted experiments is the same as described in Section 3.3.1.3. The output of the body mounted multiplexer will be mixed with the output of the planetary scan platform remote multiplexer and recorded in the data storage tape recorders for playback at a later time.

The planetary scan platform remote multiplexer and the body mounted multiplexer each contain 64 analog inputs and 16 digital inputs, all of which are wired to individual experiment data channels. The multiplexers are addressed by the FFG to establish the sequence of sampling of the individual data channels connected to the input of both multiplexers. The sequence stored in the FFG can be altered by ground command.

Figures 6-4 and 6-5 illustrate typical science data profiles for an eclipse and noneclipse orbit respectively. The data profiles are consistent with the science time lines illustrated in Section 6.2.2.2, based on the desirability of operating some of the additional experiments continuously at a rather low data rate (e.g. magnetometer, solar x-ray, gamma ray spectrometer).

One approach in regard to recording continuous low rate science data is to include the capability of servicing experiment data at two rates.

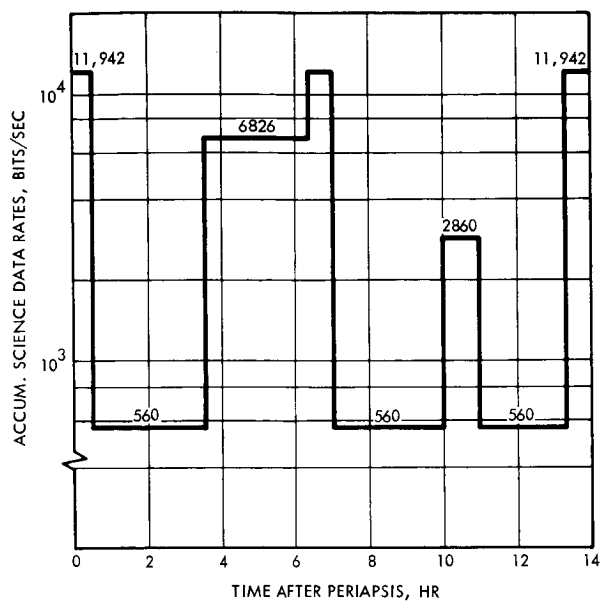


Figure 6-4
SCIENCE DATA PROFILE FOR ECLIPSE ORBIT shows representative accumulated data rates for core experiments and all additional experiments defined in section 5.0.

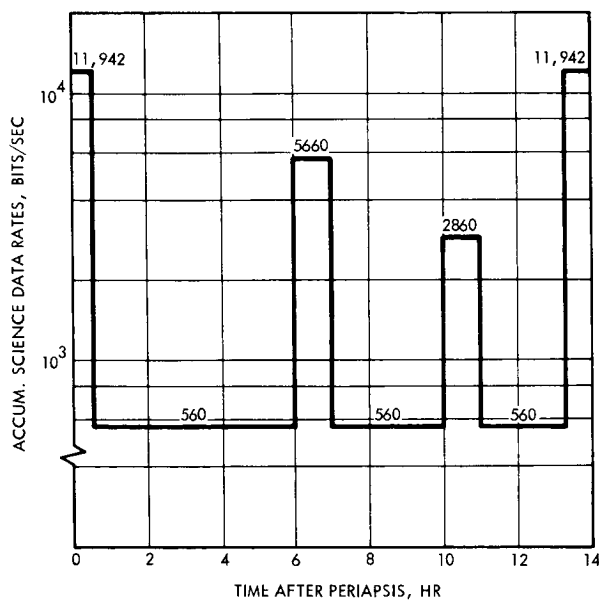


Figure 6-5
NONECLIPSE ORBIT DATA PROFILE for core experiments and all additional experiments show accumulated data rates for a typical science sequence throughout the nominal orbit.

Table 6-3. Orbital Science Data Estimates for the Additional Experiments

Instrument	Data Acquisition Assumptions	Buffered Data Rate (Bits/Sec)	Total Bits/Orbit
Atmospheric mass spectrometer	Data required continuously when spacecraft within ± 76 deg of periapsis. (Approximately 1.1 hr)	5100	2.02×10^7
Polarimeter	Data required continuously when spacecraft within ± 76 deg of periapsis. (Approximately 1.1 hr)	16	8.06×10^4
Solar occultation	Data required continuously through entire orbit (13.8 hr)	48	2.39×10^6
Gamma ray spectrometer	Data required continuously through entire orbit	16	7.95×10^5
Neutron albedo	Data required continuously when spacecraft within ± 76 deg of periapsis. (Approximately 1.1 hr)	16	8.06×10^4
Magnetometer	Data required continuously through entire orbit	240	1.19×10^7
Micrometeoroid			
a) Impact detector	Interrogate only once per orbit	--	--
b) Flash detector	a) Operate only during eclipse seasons	--	--
	b) Interrogate only once per orbit	--	--
Solar plasma	Data required continuously through entire orbit	256	1.27×10^7
Solar x-ray	Data required continuously through entire orbit	256	1.27×10^7
Celestial x-ray	Data required continuously through entire orbit	16	7.95×10^5
Subtotal - Additional Experiments			6.160×10^7
Subtotal - Core Experiments (From Section 3)			1.62×10^7
Grand Total			7.79×10^7



A high data recording rate (12.8 kbs) is utilized during periods when rates greater than 2800 bps are required. A low data rate (512 bps) is utilized during all other periods such that data at the low rate can be recorded for extended periods of time.

A data format consistent with the format described in Section 3.3.1.4.1 can be utilized and altered as required for the low and high data rate periods by the FFG. For the high data periods, experiments requiring high sampling rates are switched to the main frame and continuous low rate science experiments are switched to the subcomm. For the low data periods, continuous low rate science experiments are switched to the main frame.

Two tape recorders are required for continuous science operations. One recorder is required to record data while the other is in playback. The second tape recorder used in support of science operations could be the engineering data recorder, since engineering data is normally recorded only during spacecraft maneuvers and earth occultation. As described in previous sections the recorded science experiment data is interleaved between transmissions of photo-imaging data for playback.

6.2.2.2 Science Sequencing

A list of commands exclusively for the operation and control of experiments is contained in Appendix A, Table 3 for the payload with all additional experiments. The command list represents an estimate of reasonable demands typical of those experiments listed. The total required for this payload is 22 power commands and 65 discrete commands, as compared to 7 power and 42 discrete commands for the recommended Film/TV payload and core experiments.

Figure 6-6 illustrates the basic interface with science experiments in regard to command and sequencing. The additional equipment required for the body mounted experiments is the electrical integration assembly and associate cabling.

A total of 67 discrete commands are required at the output at the remote decoder as compared to 56 for the basic payload. Twenty (20) of the 67 discrete commands are required for control of the switching of power (ON/OFF) within the remote science switching assembly for

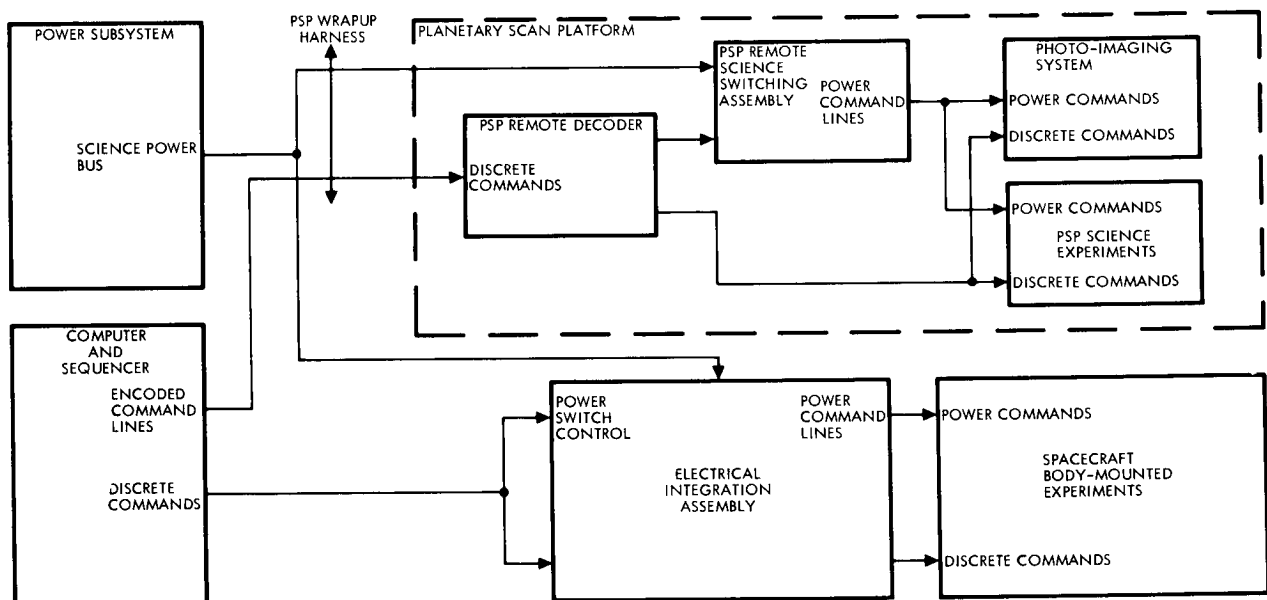


Figure 6-6
SCIENCE COMMAND AND SEQUENCING INTERFACE for the film/tv payload with core and all additional experiments requires an additional electrical integration assembly for body-mounted experiments.



individual experiments. The remainder will be utilized by experiments directly.

A total of 42 discrete commands are required at the input to the EIA. Twenty-four (24) of the 42 discrete commands are required for the control of the switching of power (ON/OFF) within the EIA for individual body mounted experiments. The remaining 18 discrete commands will be wired through the EIA and distributed to the body mounted experiments as required.

The distribution of primary science power from the power subsystem to body mounted experiments is accomplished within the EIA and the inter-connect cable to experiments. A power switch for each power line required by experiments will be contained within the EIA. The switch will be controlled (commanded ON or OFF) by discrete commands from the computer and sequencing subsystem.

Power estimates for the payload with all additional experiments are contained in Appendix C, Table 5, and Table 6 for eclipse and noneclipse orbits respectively. The total average power for this payload is 122.90 watts for an eclipse orbit and 100.33 watts for a noneclipse orbit, as compared to 84.99 watts and 64.16 watts for an eclipse and a noneclipse orbit respectively for the recommended Film/TV payload and core experiments.

Figures 6-7 and 6-8 illustrate typical experiment operations (data acquisition periods) for an eclipse and a noneclipse orbit respectively for the payload with all additional experiments. In addition to illustrating science operations as a function of altitude and true anomaly for the nominal orbit, the associated data tape recorder operations are also shown, as are the composite science power profile including the tape recorders and photo-imaging film readout for each orbit.

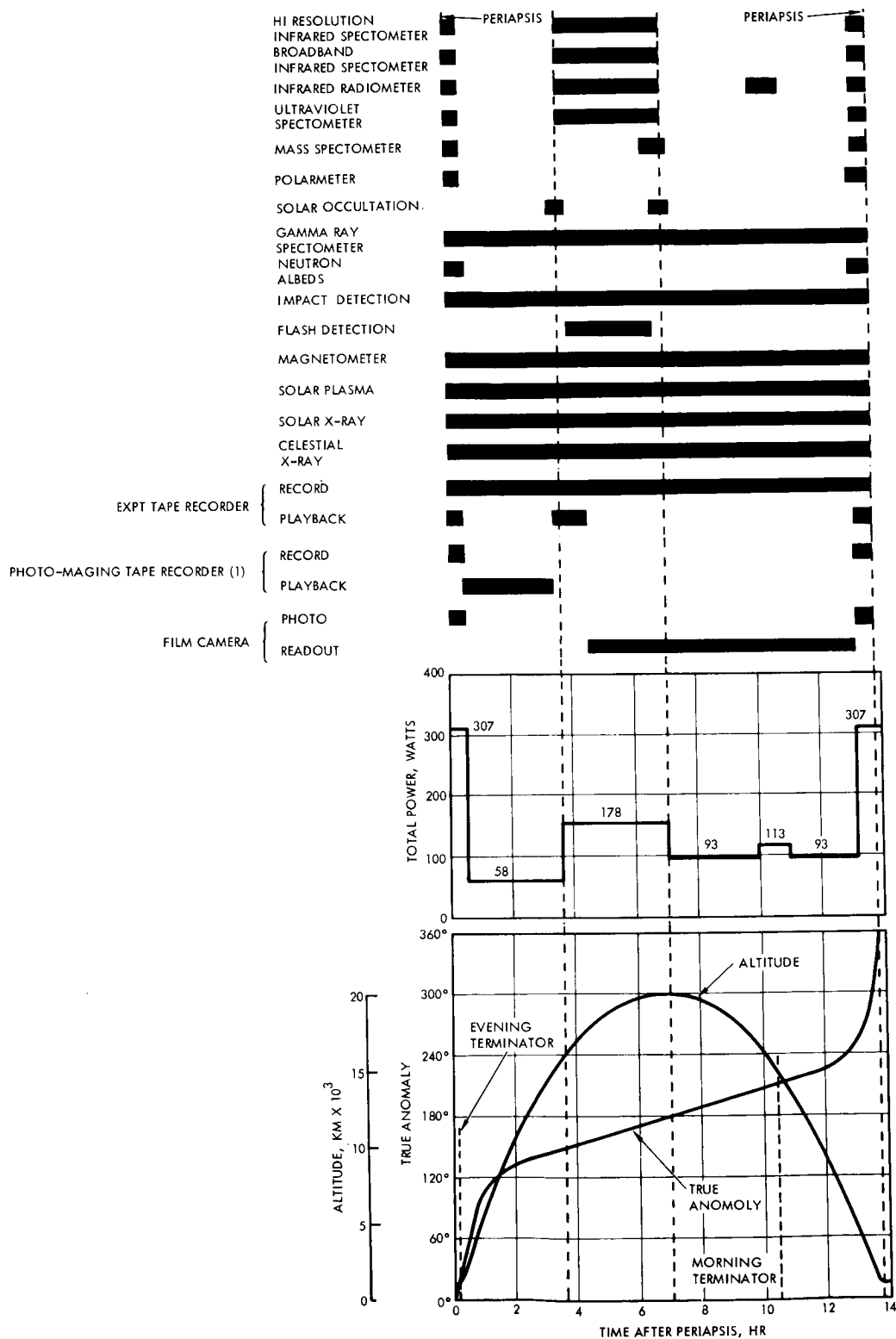


Figure 6-7
ECLIPSE ORBIT POWER PROFILES FOR ADDITIONAL EXPERIMENTS with film/TV and core experiments show peak power requirements around periaapsis and during eclipse period. The average power is 122.9 watts.

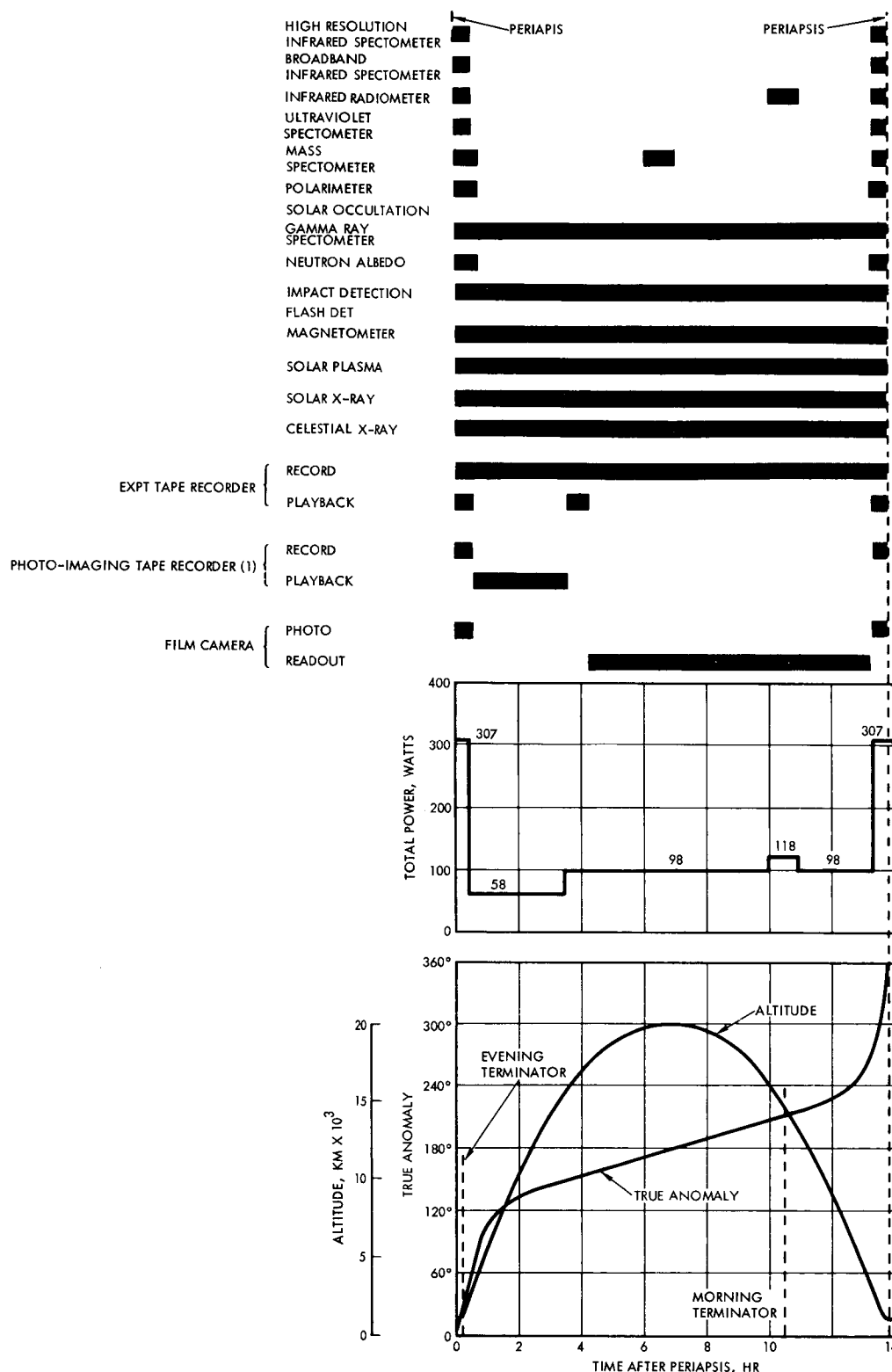


Figure 6-8

PEAK POWER FOR FILM/TV, core, and additional experiments occurs around periapsis for noneclipse orbits. The average power for the science experiments is 100.33 watts.

6.2.2.3 Electrical Integration Design

An electrical interface chart for the Film/TV payload with all additional experiments is contained in Appendix D, Table 3. The number of electrical signals required by individual experiments is specified and is considered to be representative of expected demands for those experiments identified. The detailed command and telemetry functions for each experiment is contained in Appendix A, Table 3 and Appendix B, Table 3 respectively.

The total number of wires for experiments contained within the planetary scan platform in this payload is derived in Appendix E, Table 3 based upon the interface chart. A total of approximately 289 wires is required in the planetary scan platform for the Film/TV payload with all experiments as compared to 227 for the recommended Film/TV payload with only core experiments. The use of the remote planetary scan platform units reduces this to approximately 30 wires down the shaft of the planetary scan platform in a nonredundant configuration. These numbers do not include wires for planetary scan platform support equipment such as the Mars sensor, gimbal drives, gimbal angle encoders, and thermal control heaters.

Figure 6-9 is a general block diagram illustrating the basic electrical interfaces between the Film/TV payload with all additional experiments and the spacecraft subsystems. The additional equipments required for the body mounted experiments are the science main body multiplexer and the electrical integration assembly (EIA).

The EIA provides maximum flexibility in the choice of boom and body mounted designs. The interconnect cabling between the spacecraft systems and the EIA can be designed and standardized with little consideration for the individual body mounted experiment interfaces. The

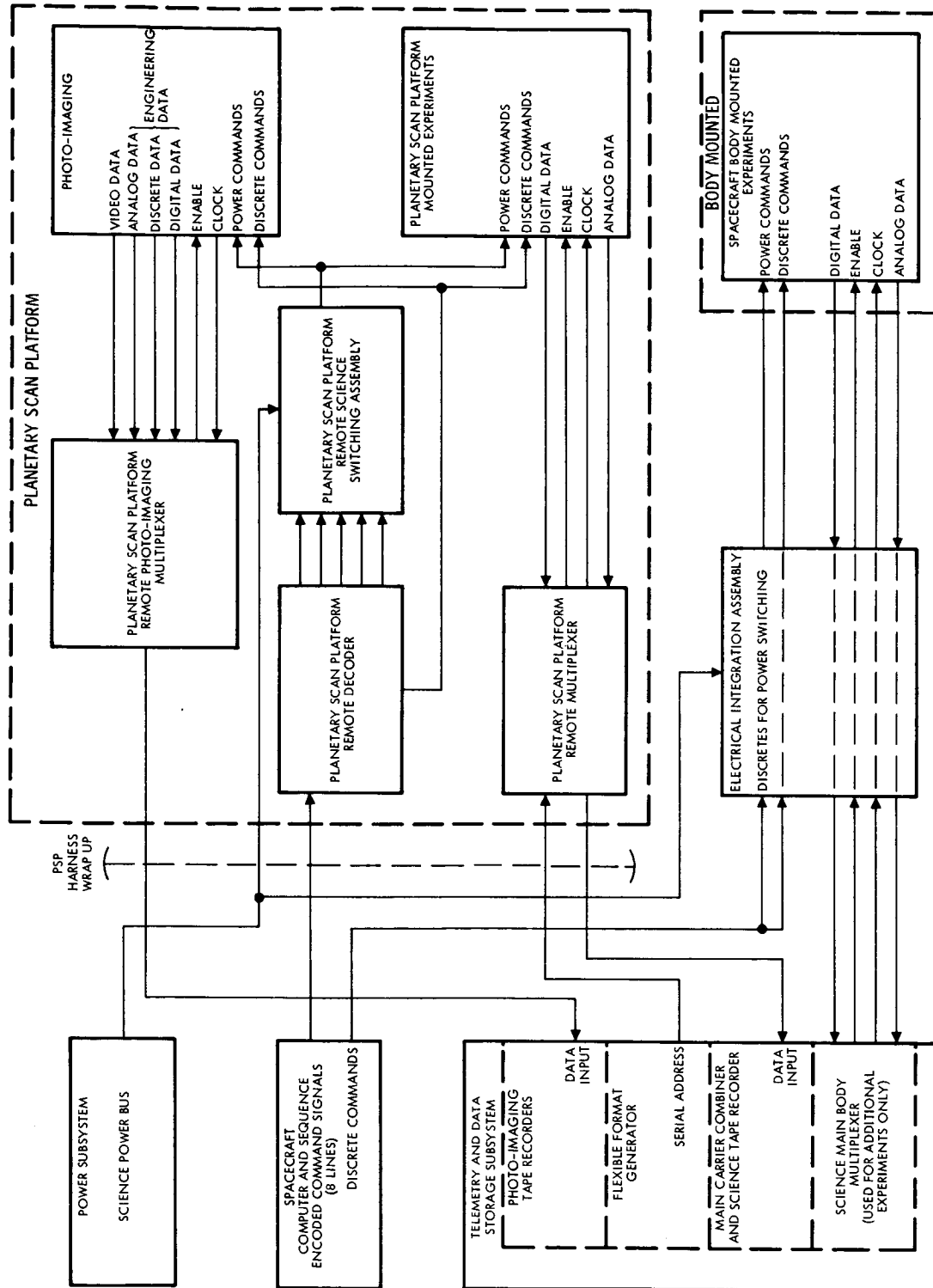


Figure 6-9
SCIENCE EQUIPMENT INTERFACE BLOCK DIAGRAM shows how installation of remote command decoders, switching assemblies and multiplexers on the Planetary Scan Platform reduces wiring between instruments and spacecraft; body-mounted experiments interface with the spacecraft through a science electrical integration assembly.

individual experiment electrical interface requirements can subsequently be satisfied by the design of the interconnect cable between experiments and the EIA.

6.2.3 Thermal

Temperature control of additional science equipment mounted within the planetary scan platform is accomplished by mounting the equipment directly to the scan platform structure and coating with Cat-a-lac black paint.

Temperature control of boom mounted equipment is accomplished by using thermostatically controlled heaters and insulation.

Body mounted science packages are thermally isolated from the structure to which they are attached. The science package temperature is controlled by use of insulation, surface coatings, and thermostatically controlled heaters.



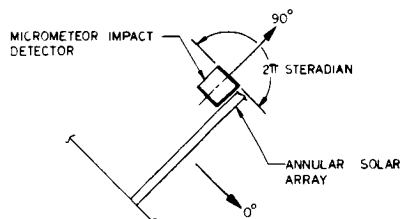
7. AUGMENTED PHOTO-IMAGING SYSTEM FOR UPGRADED SPACECRAFT

7.1 UPGRADED SPACECRAFT DEFINITION

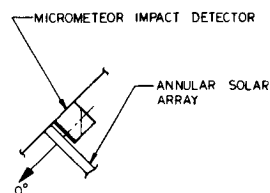
The configuration of the upgraded spacecraft is shown in Figure 7-1. This spacecraft is defined in detail in Volume 6. Basically, the upgraded spacecraft has a twenty foot diameter antenna, a 100 watt communications transmitter and additional solar power to support the higher power equipment and additional experiments. For this study it is also assumed that all of the additional experiments defined in Section 5 and all of the core science experiments defined in Section 2 are included in the science experiment payload. This assumption also includes an increase in the weight allocation to accommodate the total science payload.

As described in Section 6.1, additional experiments can also be carried on the recommended spacecraft, given the appropriate weight and power allocations. The main advantage of the upgraded spacecraft is the considerably higher communications data transmission rate available. The data rate for the upgraded spacecraft is approximately six times greater than that available on the recommended spacecraft. Since the data from the basic science experiments is not large, the main advantage for an upgraded spacecraft would be to exploit the photo-imaging aspects of the mission. The following section describes a proposed augmented Film/TV System which will exploit this capability of the upgraded spacecraft.

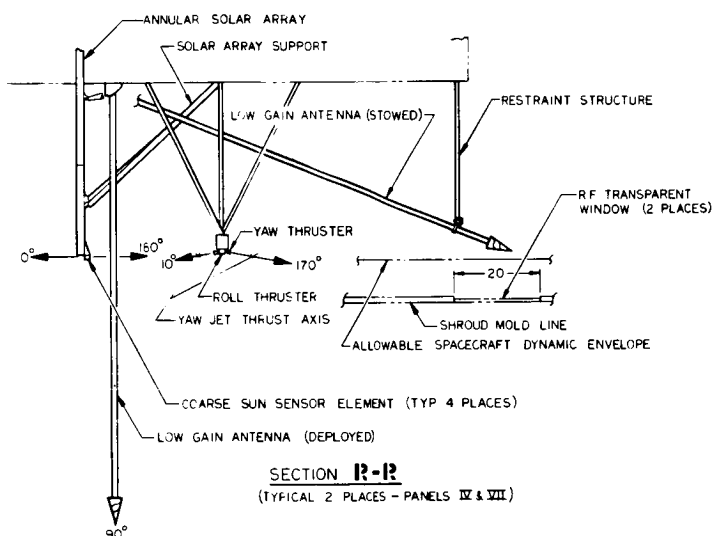
A weight statement for the science experiments and the augmented Film/TV System is given in Appendix G-III.



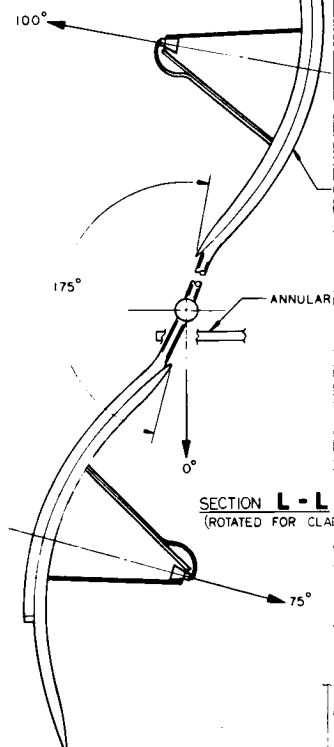
SECTION 0-0



SECTION Z-Z

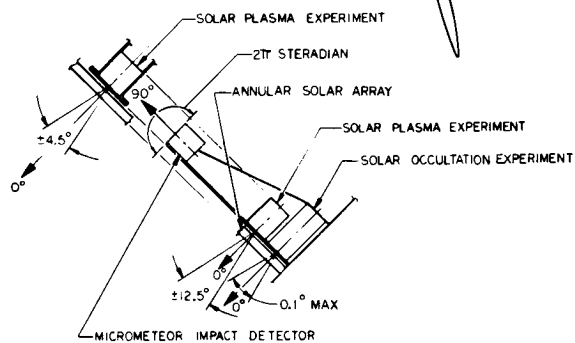


SECTION R-R
(TYPICAL 2 PLACES - PANELS IV & VII)



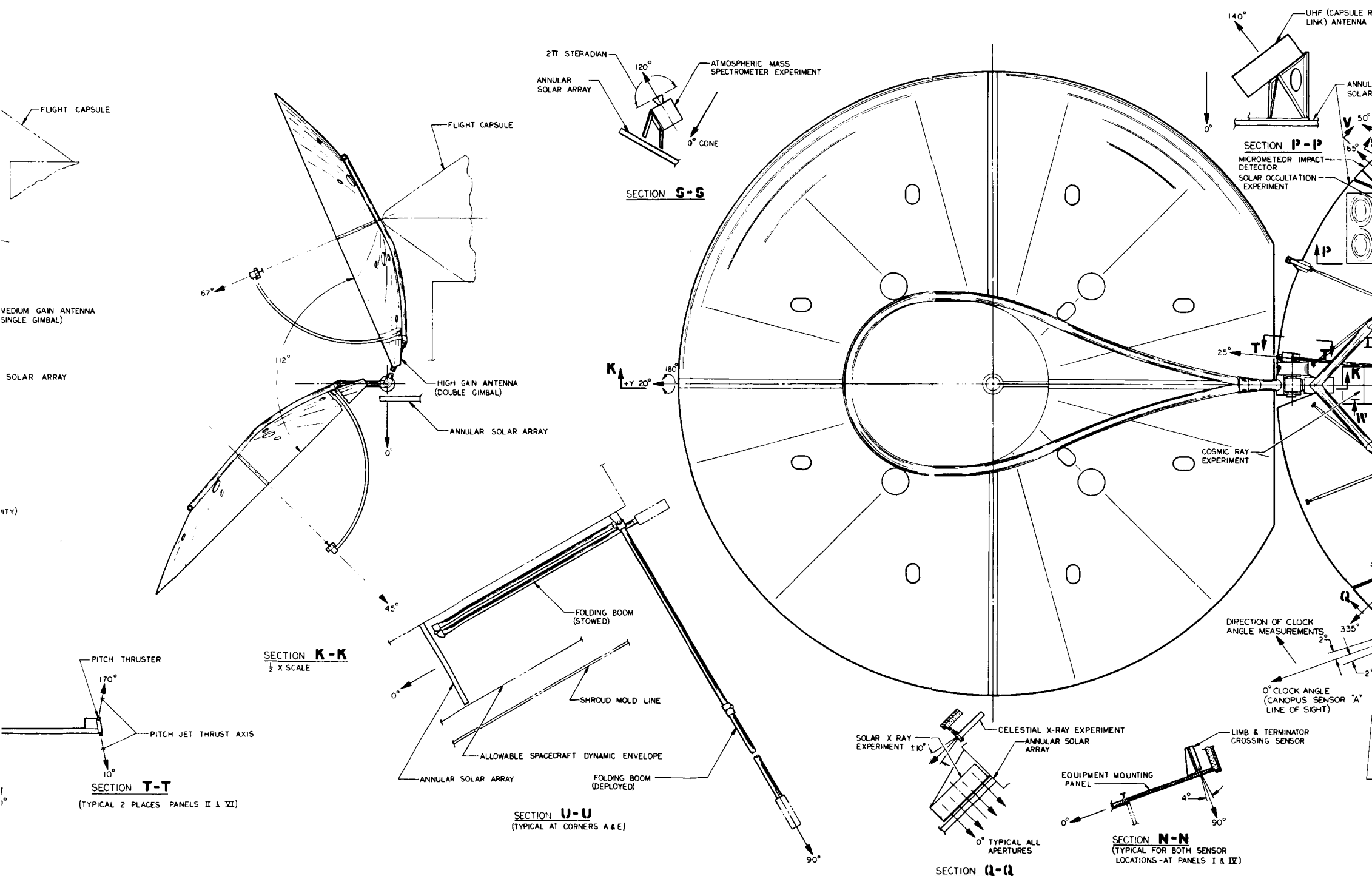
NOTES:

1. ALL ANGLES SHOWN IN ALL SECTIONS DEFINE THE CONE ANGLE AXIS FOR EACH SENSOR.
2. 0° CONE ANGLE IS DEFINED AS A LINE THRU THE SPACECRAFT POINTING AFT AT THE SUN.
3. ALL ANGLES IN MAIN VIEW DEFINE THE CLOCK ANGLE FOR EACH SENSOR AS MEASURED CLOCKWISE FROM THE LINE OF SIGHT OF THE CANOPUS 'A' SENSOR LOOKING AFT.
4. 0° CLOCK ANGLE IS DEFINED AS THE LINE OF SIGHT OF THE CANOPUS 'A' SENSOR.
5. DRAWING SHOWS SECTIONS J-J THROUGH Z-Z



SECTION V-V

AD7-164 1/10



FOLDOUT FRAME
7-2 B

FOLDOUT FRAME
7-2 C

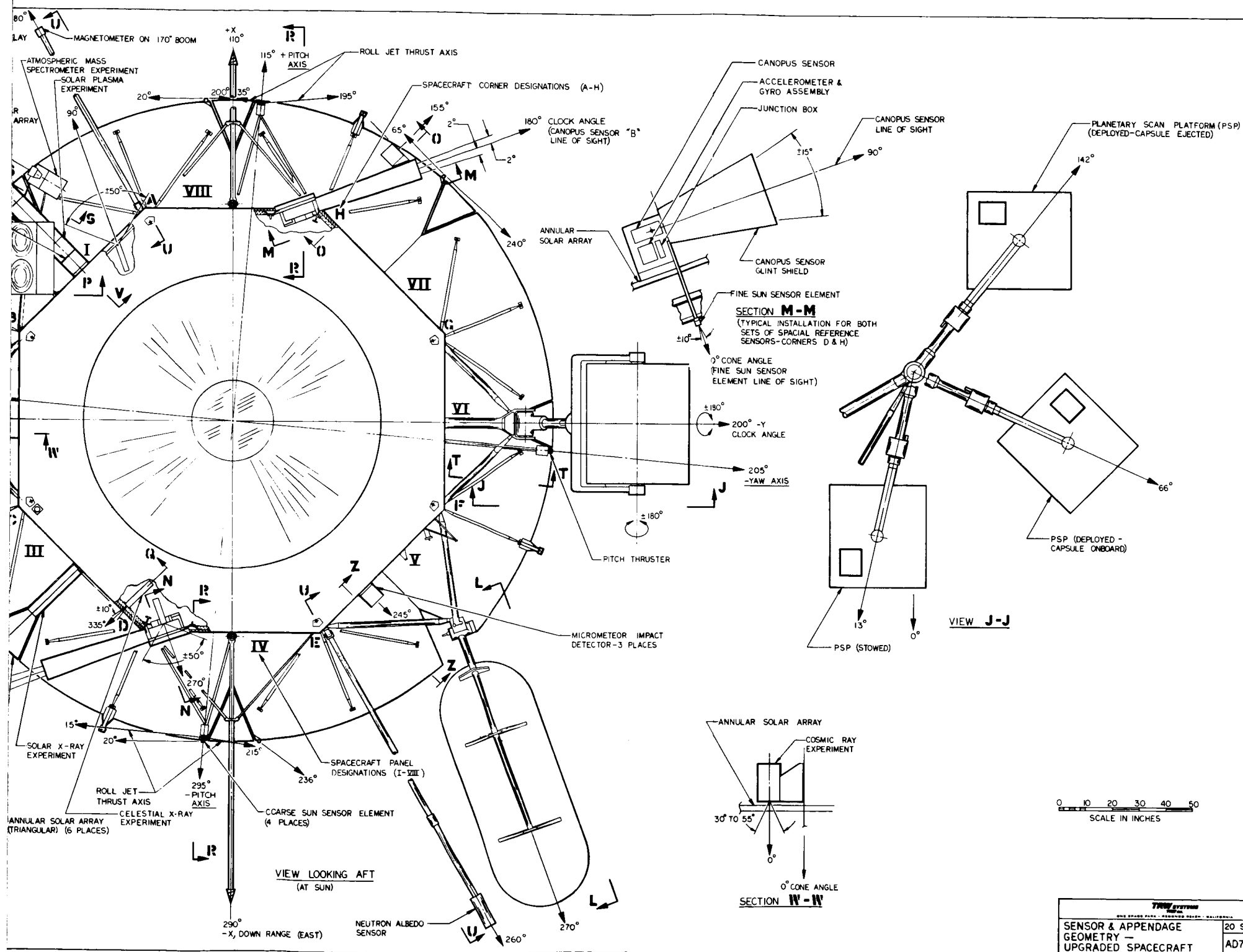


Figure 7-1
UPGRADED SPACECRAFT with film/TV core experiments and all additional experiments shows location of experiments and view angles.

FOLDOUT FRAME

7-20

7-2E / 7-3 FOLDOUT FRAME



7.2 DEFINITION OF AUGMENTED FILM/TV SYSTEM

In general, an increased total data allotment arising from increased data transmission rates or longer mission lifetime, or both, can be exploited by the photo-imaging system in two ways: through additional coverage at the same resolutions or similar coverage at higher resolution. A system capable of higher resolution can be obtained by higher intrinsic resolution capability of the sensor, by a longer focal length optical system, or a combination of these two. In addition, higher resolution usually entails greater demands on spacecraft stability and accuracy of image motion compensation.

Since the inherent resolution capability of the film is not likely to improve, higher resolutions must be achieved with longer focal length optics. A catadioptric imaging system similar to that specified for the recommended system with twice the focal length and 1.4 times the aperture, i.e., a 2000 mm focal length $f/5.6$ system, can probably be included in the augmented photographic instrumentation with a weight increase of about 30 pounds. With a nominal resolution of 5 meters at a periapsis altitude of 1000 kilometers, this system would have exposure times twice as long, and allowable image motions smears half as large, as the recommended system. Therefore, the image motion compensation must be four times as good, or accurate to about one percent. If it proved to be difficult to achieve this accuracy with the planetary scan platform gimbal drives, a closed-loop V/H sensor and IMC drive mechanism could be added to the film system, with a further increase of about 10 pounds in weight for the system. The resolution of the 100 meter resolution portion of the film camera could easily be improved by using a different lens weighing an additional 5 pounds to achieve a resolution of 50 meters.

The rate at which the film camera can acquire photography is limited only by orbital characteristics, and is greater than enough to fill the data link at the increased transmission rates, so the only modification necessary in the recommended medium and high resolution film system to obtain greater coverage is to increase the film supply.

The total cumulative data that can be transmitted by the upgraded spacecraft varies from about 1.2×10^{12} to 2.9×10^{12} bits, according to launch year and encounter date. To estimate the photographic coverage possible with the upgraded system a nominal value of 2.5×10^{12} bits allotted to the photo-imaging system has been taken. To provide enough film for this, with a safety factor of two, requires 600 feet of film, i. e., 470 feet more than in the original recommended system, for a total capacity of about 5×10^{12} bits. The added film with the Bimat processing web to go with it, weighs about nine pounds. Some additional radiation shielding is required, weighing another nine pounds. If only this change were made the recommended Film/TV System will increase from 220 to 238 pounds.

The total weight of the upgraded photo-imaging system on the planetary scan platform if both higher resolution lenses and greater film capacity were included increases from 220 pounds to 273 pounds (without the V/H sensor system which would add 10 pounds more).

The relative amount of information to be collected by each part of the photo-imaging system will differ somewhat from the recommended system. Low resolution color TV coverage is adequate in the recommended system, so should remain about the same in absolute amount but become a smaller fraction of the total in the upgraded system. After the first one or two missions, the importance of the low resolution photography will decrease. However, the recording of seasonal changes in color such as the advance and retreat of the polar caps will probably continue to be valuable during later missions. Other temporal changes such as variations in the blue haze and occurrence of yellow clouds will also warrant color coverage at relatively low resolution.

The shift of emphasis in the medium and high resolution photography from monoscopic to stereoscopic coverage will be more rapid for the upgraded system than for the recommended design, because fairly complete single coverage will be acquired sooner. Also, high resolution photography will assume increasing importance in later missions, since the need to devote a major fraction of the transmission time to general coverage will no longer exist as that coverage is obtained. Moreover,



previous information will indicate areas for which high resolution photography is desirable, eliminating the necessarily somewhat random coverage of the first one or two missions.

Table 7.2-1 gives an example of the photographic coverage possible with an augmented Film/TV system with resolutions of 1000 meters, 50 meters, and 5 meters. The values given in the Table are based on the nominal value of 2.5×10^{12} total bits for a mission. The relative distribution of transmission time (proportional to data bits) among the three systems is probably representative of an early mission. Great flexibility exists in the way the distribution of mapping resolutions could be changed to fit the requirements of a particular mission.

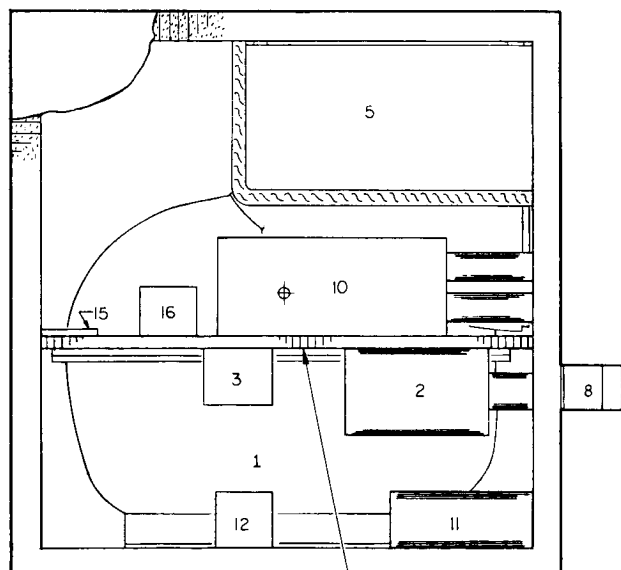
A layout of the Film/TV system in the PSP for the upgraded spacecraft is shown in Figure 7-2.

Table 7.2-1. Photographic Coverage of Mars
with Augmented Film/TV System

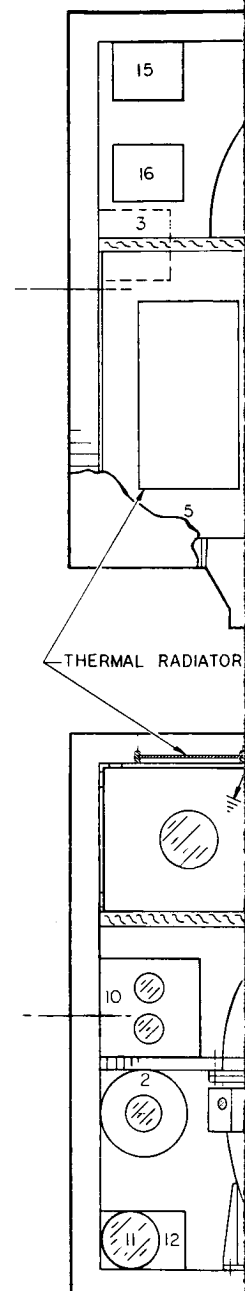
	Low Resolution TV (Color)	Medium Resolution Film	High Resolution Film
Ground resolution at periapsis	1000 m	50 m	5 m
Nominal area/ frame	$2.5 \times 10^5 \text{ km}^2$	$9 \times 10^4 \text{ km}^2$	$9 \times 10^2 \text{ km}^2$
Data bits/frame	8.58×10^6	1.24×10^9	1.24×10^9
Data Allotment (bits)	1.52×10^{10}	1.24×10^{12}	1.24×10^{12}
Number of frames	585 (color pictures)	1000	1000
Total area/mission (monoscopic)	$1.46 \times 10^8 \text{ km}^2$	$9 \times 10^7 \text{ km}^2$	$9 \times 10^5 \text{ km}^2$
Percent coverage (monoscopic)	100	62	0.62
Percent coverage (half monoscopic and half stereo- scopic)		20.6 mono 20.6 stereo	0.2 mono 0.2 stereo

EQUIPMENT LIST

1. FILM CAMERA SYSTEM
2. LOW RESOLUTION COLOR TV CAMERA
3. LOW RESOLUTION COLOR TV CAMERA ELECTRONICS
4. BROADBAND INFRARED SPECTROMETER
5. HIGH RESOLUTION INFRARED SPECTROMETER
6. INFRARED RADIOMETER
7. ULTRAVIOLET SPECTROMETER
8. MARS SENSOR (2 PLACES)
9. MICROMETEOR DETECTOR- LIGHT FLASH
10. POLARIMETER
11. GAMMA RAY SPECTROMETER
12. GAMMA RAY SPECTROMETER ELECTRONICS
13. PSP REMOTE PHOTO-IMAGING MULTIPLEXER
14. PSP REMOTE MULTIPLEXER UNIT
15. PSP REMOTE DECODER UNIT
16. PSP REMOTE SCIENCE SWITCHING ASSY
17. CRYOGENIC REFRIGERATOR



EQUIPMENT MOUNTING
PLATFORM



THERMAL RADIATOR

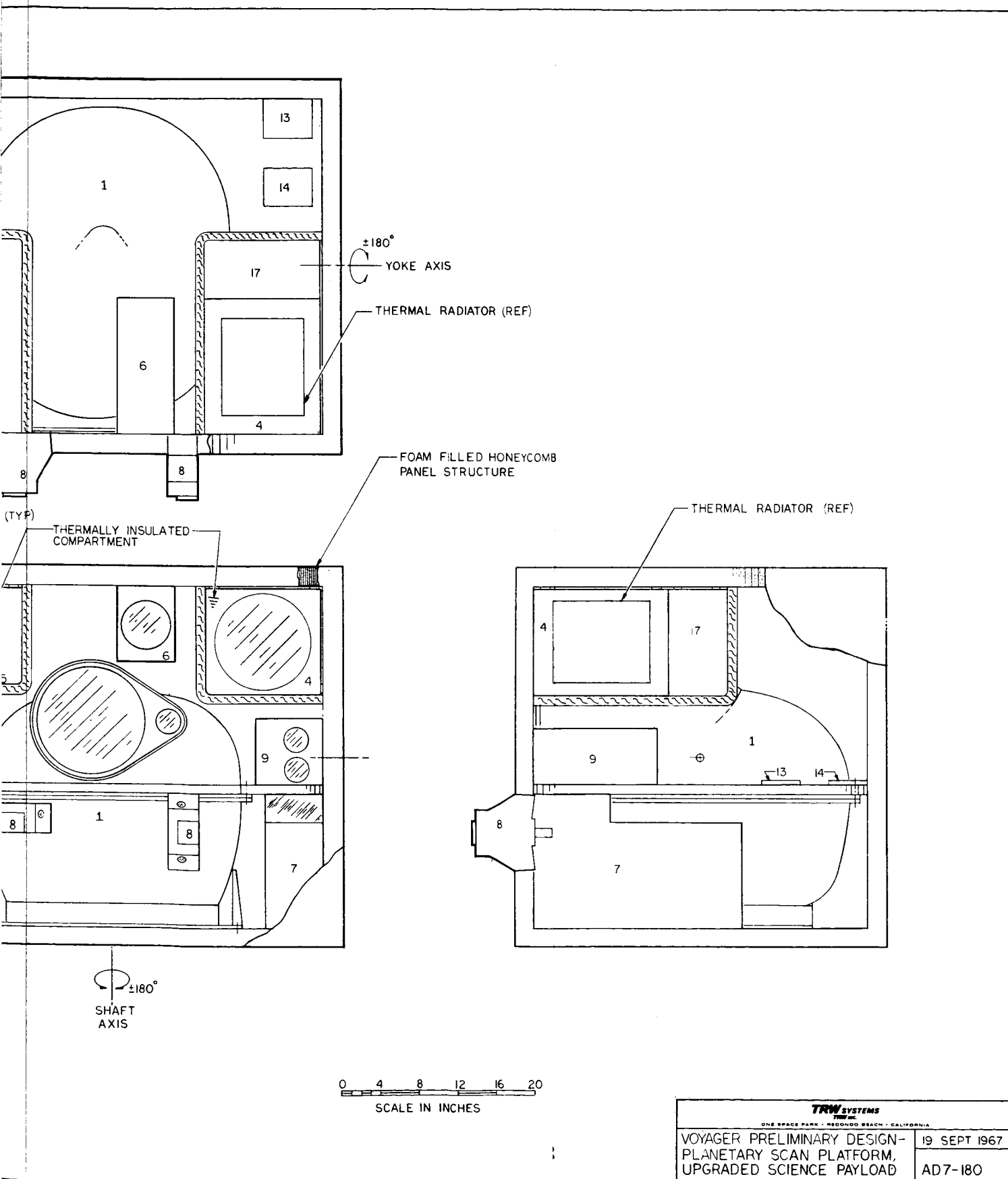


Figure 7-2
PSP FOR UPGRADED SPACECRAFT contains film/TV system, core experiments, and three additional experiments 1) Gamma Ray 2) Polarimeter 3) Micrometeor Detector-Light Flash.



Table 1. Command List for Hypothetical Payload

<u>Instrument</u>	<u>Function</u>	<u>Command</u>		
		<u>Power</u>	<u>Discrete</u>	
Core experiments				
High-resolution IR spectrometer	Power on/off	1		
	Wavelength scan		1	
	Calibrate		2	
Broadband IR spectrometer	Power on/off	1		
	Cryogenic cooler on/off	1		
	Calibrate		2	
IR radiometer	Power on/off	1		
	Calibrate		1	
UV spectrometer	Power on/off	1		
	Calibrate		1	
Photo-imaging				
One low-resolution TV camera	Turn on and warm up	1		
	Operate shutter on/off		2	
	Scan and record on/off		2	
	Erase and neutralize target on/off		2	
	Set iris opening (6 position)		6	
	Internal calibration test on/off		2	
	Filter wheel position (4 position)		4	
	Three medium- resolution TV cameras	Turn on and warm up	3	
		Operate shutter on/off		6
Scan and record on/off			6	
Erase and neutralize target on/off			6	
Focus lens (6 position)			18	
Set iris opening (6 position)			18	
Internal calibration test on/off			6	
One high-resolution TV camera	Turn on and warm up	1		
	Operate shutter on/off		2	
	Scan and record on/off		2	
	Erase and neutralize target on/off		2	
	Set iris opening (6 position)		6	
	Internal calibration test on/off		2	
	Filter wheel position (4 position)		4	
	Subtotal - Core Experiments	5	7	
	Subtotal - Photo-imaging	5	96	
Grand Total	10	103		

Table 2. Command List Summary for Recommended Photo-imaging System Plus Core Experiments*

<u>Instrument</u>	<u>Function</u>	<u>Command</u>		
		<u>Power</u>	<u>Discrete</u>	
Photo-imaging (recommended)				
One low-resolution TV camera	Turn on and warm up	1		
	Operate shutter on/off		2	
	Scan and record on/off		2	
	Erase and neutralize target on/off		2	
	Set iris opening (6 position)		6	
	Internal calibration test on/off		2	
	Filter wheel position (4 position)		4	
	One Eastman-Kodak film camera	System power on/off	1	
Velocity/height on/off (optional)			2	
Camera on			1	
Readout electronics on			1	
Readout drive on/off			2	
Cut bimat			1	
Inhibit heaters			1	
Wind forward			1	
Solar eclipse on/off			2	
Fast camera rate			1	
Slow camera rate			1	
Frame count A			1	
Frame count A			1	
Frame count B			1	
Frame count B			1	
Subtotal - Recommended photo-imaging system		2	35	
Subtotal - Core experiments		5	7	
Grand Total		7	42	

* Core experiments include all experiments listed in Table 1, Appendix A excluding photo-imaging.



Table 3. Command List Summary for Additional Experiments
Plus Recommended Payload

<u>Instrument</u>	<u>Function</u>	<u>Command</u>		
		<u>Power</u>	<u>Discrete</u>	
Additional Experiments				
Atmospheric mass spectrometer	Power standby	1		
	Main power on/off	1		
	Filament select		1	
Polarimeter	Power on/off	1		
	Calibrate		1	
	Voltage level select		2	
Solar occultation	Power on/off	1		
	Calibrate		1	
Gamma ray spectrometer	Power on/off	1		
	Calibrate		1	
Martian neutron albedo	Power on/off	1		
	Calibrate		1	
	High voltage range set		2	
Magnetometer	Power on/off	1		
	Heater control	1		
	Range select normal/override	1		
	Range select manual		1	
Micrometeorite Impact detector	Main power on/off	1		
	Calibrate		1	
	Flash detector	Main power on/off and standby	2	
		Calibrate		1
Solar plasma	Power on/off	1		
	Multiplexor select		4	
Solar X-ray	Power on/off	1		
	Multiplexor logic select		4	
	Range select		2	
Celestial X-ray	Power on/off	1		
	Calibrate		1	
Subtotal - Additional experiments		15	23	
Subtotal - Recommended payload		7	42	
Grand Total		22	65	

* Recommended Payload includes Film/TV photo-imaging system plus core experiments.



Table 1. Telemetry List for Hypothetical Payload

Instrument	Function	Telemetry Channels		
		Analog	Digital	Discrete
Core Experiments				
High resolution IR spec- trometer	Multiplexer output		1	
	Multiplexer status	1		
	Grating position monitor	1		
	Calibration monitor	1		
	PbSe temperature monitor	1		
	Bias voltage level	1		
	AGC status monitor	1		
	Power Supply voltage monitor	1		
	Electronics temperature	1		
Broadband IR spec- trometer	Multiplexer output		7	
	Multiplexer status	1		
	Detector no. 1 tempera- ture monitor	1		
	Detector no. 2 tempera- ture monitor	1		
	Telescope and mon- chrometer temperature	1		
	Electronics temperature	1		
	Chopper temperature 1 and 2	2		
IR radiometer	Radiometer output	12		
	Electronics temperature	1		
	Detector no. 1 tempera- ture monitor	1		
	Detector no. 2 tempera- ture monitor	1		
	Mirror position	1		
	High/low sensitivity status	1		
	Calibration monitor	1		
UV spectrom- eter	Spectrometer output	12		
	High voltage monitor 1 and 2	2		
	Power supply tempera- ture monitor 1 and 2	2		
	Grating position monitor	1		
	Electronics temperature	1		
	Multiplexer status	1		
Photo-imaging (hypothetical)	Sequencer status	1		
One low-resolu- tion TV camera	Target temperature	1		
	Heater current	1		
	High voltage		1	

Table 1. Telemetry List for Hypothetical Payload (cont)

Instrument	Function	Telemetry Channels		
		Analog	Digital	Discrete
	High light level	1		
	Shutter position	1		
	Filter wheel position	1		
	Power supply voltages	6		
	Time reference (18 bits)		3	
	Power-on command (1 bit)			1
	Shutter command (1 bit)			1
	Scan/record command (1 bit)			1
	Erase command (1 bit)			1
	Trio command (1 bit)			1
	Calibrate command (1 bit)			1
Three medium-resolution TV cameras	Target temperature	3		
	Heater current	3		
	High voltage	3		
	High light level	3		
	Shutter position	3		
	Power transformer temperature	3		
	Power supply voltages (6)	18		
	Timing references (18 bits)		9	
	Power-on command (1 bit)			3
	Shutter Command (1 bit)			3
	Scan/record (1 bit)			3
	Erase command (1 bit)			3
	Focus command (1 bit)			3
	Trio command (1 bit)			3
	Calibrate command (1 bit)			3
One high-resolution TV camera	Target temperature	1		
	Heater current	1		
	High voltage	1		
	High light level	1		
	Shutter position	1		
	Power transformer temperature	1		
	Power supply voltages	8		
	Timing references (18 bits)		3	
	Power-on command (1 bit)			1
	Shutter command (1 bit)			1
	Scan/record (1 bit)			1
	Erase command (1 bit)			1
	Trio command (1 bit)			1
	Calibrate command (1 bit)			1
Subtotal - Core experiments		53	8	-
Subtotal - Hypothetical photo-imaging system		62	15	33



Table 2. Telemetry List for Recommended Photo-imaging System Plus Core Experiments

<u>Instrument</u>	<u>Function</u>	<u>Telemetry Channels</u>		
		<u>Analog</u>	<u>Digital</u>	<u>Discrete</u>
Photo-imaging (recommended)				
One low- resolution TV camera	Target temperature	1		
	Heater current	1		
	High voltage	1		
	High light level	1		
	Shutter position	1		
	Filter wheel position	1		
	Power supply voltages	6		
	Time reference (18 bits)		3	
	Power-on command (1 bit)			1
	Shutter command (1 bit)			1
	Scan/record command (1 bit)			1
	Erase command (1 bit)			1
	Trio command (1 bit)			1
	Calibrate command (1 bit)			1
One Eastman- Kodak film system	Take-up contents	1		
	Camera looper contents	1		
	Readout looper contents	1		
	Velocity/height ratio	1		
	Shutter operation			
	Platen operations	1		
	Velocity/height temperature	1		
	Camera temperature	1		
	Window temperature	1		
	Processor temperature	1		
	Dryer temperature	1		
	+10 converter output	1		
	LST cathode current	1		
	PM voltage	1		
	High voltage supply	1		
	Peak video output	1		
	Readout and fin temperature	1		
	Power supply pressure			
	Power supply humidity lower	1		
	Power supply humidity upper	1		
	Power supply environment temperature, lower	1		
	Power supply environment temperature, upper	1		
	Upper shell temperature	1		
	Pressure, nitrogen bottle	1		

Table 2. Telemetry List for Recommended Photo-imaging
System Plus Core Experiments (cont)

<u>Instrument</u>	<u>Function</u>	<u>Telemetry Channels</u>		
		<u>Analog</u>	<u>Digital</u>	<u>Discrete</u>
	Timing reference (18 bits)		3	
	Command verification (9 bits)			9
	Camera program setting (3 bits)			3
	Canera shutter setting (2 bits)			2
	Bimat clear (1 bit)			1
	Bimat take-up (1 bit)			1
	Subtotal - Recommend photo-imaging	36	6	22
	Subtotal - Core experiments (Table 1, Appendix B)	53	8	-



Table 3. Telemetry List for Additional Experiments
Plus Core Experiments

<u>Instrument</u>	<u>Function</u>	<u>Telemetry Channel</u>		
		<u>Analog</u>	<u>Digital</u>	<u>Discrete</u>
Additional Experiments				
Mass Spectrometer	Spectrometer output	8		
	Voltage monitor	1		
	Cone voltage monitor	1		
	Filament voltage monitor	1		
	Spectrometer gain indicator	1		
	Filament monitor	1		
	Electronics temperature	1		
	Scan mode	1		
Polarimeter	Polarimeter output	2		
	Power supply voltage monitor 1 and 2	2		
	Electronics temperature	1		
	Gain select status	1		
	Multiplexer status	1		
Solar Occultation	Detector output	4		
	Electronics temperature	1		
	Dynode voltage monitor	1		
	Velocity/height power supply voltage monitor	1		
	Detector temperature	1		
Gamma ray spectrometer	Detector output		2	
	Power supply voltage monitor	1		
	Dynode voltage monitor	1		
	Electronics temperature	1		
	Sensor temperature	1		
Neutron albedo	Detector output		3	
	Electronics temperature	1		
	Sensor temperature monitor	1		
	Integrating light pulser	1		
	Scintillator output	1		
	Velocity/height monitor	1		
	Log count rate monitor	1		
	Threshold level monitor	1		
Magnetometer	Detector output		3	
	Power supply voltage monitor	1		
	Sensor temperature monitor	1		
	Electronics temperature	1		

Table 3. Telemetry List for Additional Experiments
Plus Core Experiments (cont)

<u>Instrument</u>	<u>Function</u>	<u>Telemetry Channel</u>		
		<u>Analog</u>	<u>Digital</u>	<u>Discrete</u>
Meteorite flux (impact)	Detector output		2	
	Power supply voltage monitor	1		
	Sensor temperature	1		
	Vibration temperature monitor	1		
Meteorite flux (flash)	Detector output		2	
	Gain state monitor	1		
	Data reset status	1		
	Velocity/height power supply voltage monitor	1		
	Dynode voltage level monitor	1		
Solar plasma	Detector output		2	
	Stepped voltage level monitor	1		
	Power supply voltage monitor	1		
	Multiplexer status	1		
	Package temperature	1		
Solar X-ray	Detector output		2	
	Sensor temperature	1		
	Power supply voltage monitor	1		
	Package temperature	1		
Celestial X-ray	Detector output		2	
	Sensor temperature monitor	1		
	Power supply voltage monitor	1		
	Subtotal - Additional experiments	61	18	-
	Subtotal - Core experiments (Table 1, Appendix B)	53	8	-
	Total - Additional plus core experiments	114	26	-



Table 1. Power Estimate for Hypothetical Payload Including
Data Tape Recorders for Eclipse Orbit

<u>Instrument</u>	<u>Power (w)</u>	<u>Operating Time (hr)</u>	<u>Duty Cycle (%)</u>	<u>Average Power (w)</u>
Core Experiments				
High resolution IR spectrometer	20.0	3.8	27.5	5.50
Broadband IR spectrometer	20.0	3.8	27.5	5.50
IR radiometer	20.0	3.8	27.5	5.50
UV spectrometer	20.0	3.8	27.5	5.50
Data tape recorder				
Record	12.0	3.8	27.5	3.30
Playback	15.0	0.95	6.9	1.10
Photo-imaging system (hypothetical)				
One low-resolution TV camera	25	0.8	5.8	1.45
Three medium-resolution TV cameras	150	0.8	5.8	8.70
One high-resolution TV camera	20	0.8	5.8	1.16
Four data tape recorders				
Record — 20 watts per recorder	80	0.8	5.8	4.64
Playback — 15 watts per recorder	15	12.0	87.0	13.08
Subtotal — Core experiments and recorder				26.4
Subtotal — Photoimaging and recorders				29.03
Grand Total				55.43

Table 2. Power Estimate for Hypothetical Payload Including
Data Tape Recorders for Non-Eclipse Orbit

<u>Instrument</u>	<u>Power (w)</u>	<u>Operating Time (hr)</u>	<u>Duty Cycle (%)</u>	<u>Average Power (w)</u>
Core experiments				
High-resolution IR spectrometer	20.0	0.8	5.8	1.16
Broadband IR spectrometer	20.0	0.8	5.8	1.16
IR radiometer	20.0	0.8	5.8	1.16
UV spectrometer	20.0	0.8	5.8	1.16
One data tape recorder				
Record	12.0	0.8	5.8	0.7
Playback	15.0	0.2	1.5	0.23
Subtotal - Core experiments and recorders				5.57
Subtotal - Photo-imaging (From Table 1, Appendix C)				29.03
Grand Total				34.60



Table 3. Power Estimate for Recommended Photo-imaging System and Core Experiments Including Data Tape Recorders for Eclipse Orbit

<u>Instrument</u>	<u>Power (w)</u>	<u>Operating Time (hr)</u>	<u>Duty Cycle (%)</u>	<u>Average Power (w)</u>
Photo-imaging System (Recommended)				
One low-resolution TV camera	25	0.8	5.8	1.45
One medium/high-resolution film camera				
Photo mode	105	0.8	5.8	6.10
Standby to photo	77	0.5	3.4	2.60
Processing	110	-	-	-
Readout	62	9.0	65.0	40.50
One data tape recorder				
Record	20	0.8	5.8	4.64
Playback	15	3.0	21.8	3.30
Subtotal - Photo-imaging and recorders				58.59
Subtotal - Core experiments (From Table 1, Appendix C)				26.40
Grand Total				84.99

Table 4. Power Estimate for Recommended Photo-Imaging System
and Core Experiments Including Data Tape Recorders
for Non-Eclipse Orbit

<u>Instrument</u>	<u>Power (w)</u>	<u>Operating Time (hr)</u>	<u>Duty Cycle (%)</u>	<u>Average Power (w)</u>
<u>Core experiments</u>				
From Table 2, Appendix C				5.57
<u>Photo-imaging system</u>				
From Table 3, Appendix C				58.59
	Grand Total			64.16



Table 5. Power Estimate for Additional Experiments and Recommended Payload for Eclipse Orbit

<u>Instrument</u>	<u>Power (w)</u>	<u>Operating Time (hr)</u>	<u>Duty Cycle (%)</u>	<u>Average Power (w)</u>
Additional experiments				
Atmospheric mass spectrometer	22.0	2.1	15.2	3.35
Polarimeter	6.0	1.4	10.4	0.62
Solar occultation	20.0	1.2	8.7	1.68
Gamma ray spectrometer	4.0	13.8	100	4.00
Martian neutron albedo	6.0	1.4	10.4	0.62
Magnetometer	5.0	13.8	100	5.00
Micrometeoroid				
Impact detector	1.4	13.8	100	1.40
Flash detector	1.4	2.5	17.1	0.24
Solar plasma and x-ray				
Solar plasma detector	6.0	13.8	100	6.00
X-ray detector	10.0	13.8	100	10.00
Celestial x-ray detector	5.0	13.8	100	5.00
Subtotal - Additional experiments				37.91
Subtotal - Recommended payload and recorder (From Table 3, Appendix C)				84.99
Grand Total				122.90

Table 6. Power Estimate for Additional Experiments and Recommended Payload for Non-Eclipse Orbit

<u>Instrument</u>	<u>Power (w)</u>	<u>Operating Time (hr)</u>	<u>Duty Cycle (%)</u>	<u>Average Power (w)</u>
Additional experiments				
Atmospheric mass spectrometer	22.0	2.1	15.2	3.53
Polarimeter	6.0	1.4	10.4	0.62
Solar occultation	20.0	-	-	-
Gamma ray spectrometer	4.0	13.8	100.0	4.00
Martian neutron albedo	6.0	1.4	10.4	0.62
Magnetometer	5.0	13.8	100.0	5.00
Micrometeoroid				
Impact detector	1.4	13.8	100.0	1.40
Flash detector	1.4	-	-	-
Solar plasma and x-ray				
Solar plasma detector	6.0	13.8	100.0	6.00
X-ray detector	10.00	13.8	100.0	10.00
Celestial x-ray detector	5.0	13.8	100.0	5.00
Subtotal - Additional experiments				36.17
Subtotal - Recommended payload and recorders (From Table 4, Appendix C)				64.16
Grand Total				100.33

Table 1. Interface Summary for the Hypothetical Payload

Instrument	Location		Commands		Converter Sync	Telemetry					
	Planetary Scan Platform	Spacecraft Body	Power	Discrete		Analog (Video)	Analog	Quantitative	Discrete	Enable	Clock
Core experiments											
High-resolution IR spectrometer	X		1	3	1		8	1		1	1
Broadband IR spectrometer	X		2	2	1		7	7		7	1
IR radiometer	X		1	1	1		18				
UV spectrometer	X		1	1	1		20				
Photo-Imaging (hypothetical)											
One low-resolution TV camera	X		1	18	1	1	12	3	6	4	1
Three medium-resolution TV cameras	X		3	60	3	3	36	9	21	13	3
One high-resolution TV camera	X		1	18	1	1	14	3	6	4	1
Subtotal - Core experiments	X		5	7	4		53	8		8	2
Subtotal - Photo-imaging	X		5	96	5	5	62	15	33	21	5
Total	X		10	103	9	5	115	23	33	29	7



**Table 2. Interface Summary for the Recommended Photo-imaging System
Plus the Core Experiments (Recommended Payload)**

Instrument	Location		Commands		Converter Sync	Telemetry					
	Planetary Scan Platform	Spacecraft Body	Power	Discrete		Analog (Video)	Analog	Quantitative	Discrete	Enable	Clock
Photo-imaging (Recommended)											
One Low-resolution TV camera	X		1	18	1	1	12	3	6	3	1
One Eastman-Kodak film system	X		1	17	1	1	24	3	16	3	1
Subtotal - Photo-imaging	X		2	35	2	2	36	6	22	6	2
Subtotal - Core experiments (Table I, Appendix D)	X		5	7	4		53	8		8	2
Grand Total	X		7	42	6	2	89	14	22	14	4

Table 3. Interface Summary for Addition Experiments Plus the Recommended Payload

Instrument	Location		Commands		Converter Sync	Telemetry				
	Planetary Scan Platform	Spacecraft Body	Power	Discrete		Analog (Video)	Analog	Quantitative	Discrete	Enable
Additional experiments										
Atmospheric polarimetry	X		1	3	1		7			
Meteoroid flash detector	X		2	1	1		4	2		2
Gamma ray spectrometer	X		1	1	1		4	2		2
Atmospheric mass spectrometer		X	2	1	1		15			
Solar occultation		X	1	1	1		8			
Martian neutron albedo		X	1	3	1		7			
Magnetometer		X	3	1	1		3	3		3
Micrometeoroid impact detector		X	1	1	1		3	2		2
Solar plasma		X	1	4	1		4	2		2
Solar x-ray		X	1	6	1		3	2		2
Celestial x-ray		X	1	1	1		2	2		2
Subtotal - Additional experiments in planetary scan platform	X		4	5	3		15	4		4
Subtotal - Recommended payload (Table 2, Appendix D)	X		7	42	6		89	14		4
Grand Total - Planetary scan platform	X		11	47	9		104	18		6
										2



Table 1. Planetary Scan Platform Science Interface Wire Estimates
for the Hypothetical Payload

<u>Function</u>	<u>Wire Type</u>	<u>Number of Functions*</u>	<u>Number of Wires</u>	<u>Total</u>
Power commands	Twisted pair	10	(10 x 2)	20
Discrete commands	Coaxial	103	103	103
Converter sync	Twisted pair (shielded)	9	(9 x 2)	18
Telemetry				
Analog data	Coaxial	115	115	115
Digital data	Coaxial	23	23	23
Enable signal	Coaxial	25	25	25
Clock signal	Twisted pair (shielded)	7	(7 x 2)	14
Discrete	Coaxial	33	33	33
Analog video	Coaxial	5	5	5
Grounds				
Chassis	Hookup	9	9	9
Signal	Hookup	9	9	9
				374

* Number of functions taken from Table 1, Appendix D



Table 2. Planetary Scan Platform Science Interface Wire Estimate for the Recommended Photo-imaging System Plus the Core Experiments

<u>Function</u>	<u>Wire Type</u>	<u>Number of Functions*</u>	<u>Number of Wires</u>	<u>Total</u>
Power commands	Twisted pair	7	(7 x 2)	14
Discrete commands	Coaxial	42		42
Converter sync	Twisted pair (shielded)	6	(6 x 2)	12
Telemetry				
Analog data	Coaxial	89		89
Digital data	Coaxial	14		14
Enable signal	Coaxial	14		14
Clock signal	Twisted pair (shielded)	4	(4 x 2)	8
Discrete	Coaxial	22		22
Analog video	Coaxial	2		2
Grounds				
Chassis	Hookup	6		6
Signal	Hookup	6		6
				227

*Number of functions taken from Table 2, Appendix D

Table 3. Planetary Scan Platform Science Interface Wire Estimate for Additional Experiments Plus the Recommended Payload

<u>Function</u>	<u>Wire Type</u>	<u>Number of Functions*</u>	<u>Number of Wires</u>	<u>Total</u>
Power commands	Twisted pair	11	(11 x 2)	22
Discrete commands	Coaxial	47		47
Converter sync	Twisted pair (shielded)	9	(9 x 2)	18
Telemetry				
Analog data	Coaxial	104		104
Digital data	Coaxial	18		18
Enable signal	Coaxial	18		18
Clock signal	Twisted pair (shielded)	6	(6 x 2)	12
Discrete	Coaxial	22		22
Analog video	Coaxial	2		2
Grounds				
Chassis	Hookup	8		8
Signal	Hookup	8		8
				289

*Number of functions taken from Table 3, Appendix D



APPENDIX F

SCIENCE INTEGRATION DESIGN AND TEST ACTIVITIES

1. ELECTRICAL INTEGRATION

The tasks necessary to electrically integrate the science payload fall into two categories: those necessary to integrate the design of the experiments into the design of the spacecraft, and those necessary for the physical integration of the experiments into the spacecraft system.

To maintain the necessary continuity in all the activities associated with science electrical integration, all TRW Systems personnel assigned responsibility for science electrical integration design and system engineering must be assigned to integration, test, and launch phases of the program. The members of this group, the experiment responsible engineers, will normally each have total integration responsibilities for one experiment; some may be responsible for more than one. This will be determined after final selection of the Voyager science payload, at which time the specific experiment details will become known and assignments can be made.

The experiment responsible engineers will coordinate very closely with the principal investigators starting immediately after experiments are selected and continuing through the life of the program. Since the principal investigators cannot be present at TRW Systems during all design, integration, and system test operations, the experiment responsible engineers will act as the experimenters' representatives during those periods.

Following are major tasks to be performed by the experiment responsible engineers in regard to science integration coordination and design:

- The documentation and analysis of interfaces associated with the science payload
- The provision for a principal point of contact for exchange of information relating to science payload integration

- The preparation of specifications, drawings, and other data necessary to define and control the spacecraft science payload interface
- Assist in the definition of the interface between OSE and science support equipment
- Arrange for design studies necessary to evaluate alternative interface design solutions
- Participate in design reviews to ensure the maintenance of compatibility as the designs progress
- Preparation and maintenance of the science payload data book

Following are major tasks of the experiment responsible engineers with respect to science integration and test activities:

- Participation in test planning to establish integrated system test philosophy in regard to science subsystem testing
- Coordination between experimenters and spacecraft test planning and test personnel for all integrated system testing with respect to the science subsystem
- Coordination between experimenters and computer programmers to determine testing data processing requirements
- Monitor experiment unit testing at the experimenter's facility on all experiment models to acquire a detailed understanding of experiment operational and test characteristics
- Receive, handle, and store experiments and experiment EOSE at TRW Systems
- Provide laboratory space and equipment for experiment bench test operations and certification prior to integration
- Perform qualification and acceptance testing of experiment assemblies
- Support spacecraft test crew in integration and systems tests of experiments on the spacecraft
- Assist experimenters in calibration, troubleshooting, and repair of experiments



- Preliminary processing and analysis of experiment test data
- Reporting experiment test results to experimenters
- Support spacecraft test crew during launch operations
- Design and develop spacecraft simulators to perform experiment bench, environmental and certification tests consistent with integrated system test procedures

A general flow diagram of major tasks associated with science integration is contained in Figure F-1. A detailed science integration plan is contained in Volume 8, Implementation Plan.

2. ELECTROMAGNETIC COMPATIBILITY

To ensure electromagnetic compatibility (EMC) between individual experiments and between experiments and the spacecraft a comprehensive EMC program will be implemented. Following are major tasks that will be performed by EMC specialists under the direction of the program director and his staff:

- Define EMC requirements for the science payload interface specification. Includes definition of EMC generation and susceptibility levels, shielding, grounding, and bonding
- Through examination of experiment block diagrams, experimenter discussions, analysis of circuitry, etc., determine the maximum levels of generated interference and minimum threshold of susceptibility, both to conducted and radiated fields for each experiment
- Derive empirically and analytically these same parameters for the spacecraft equipment
- Using the information obtained above, determine the conducted and radiated spectrum which the various spacecraft equipment should be subjected to during its engineering model EMI susceptibility testing. This will define the types of fields, frequency spectrum, and levels which will be purposely generated to determine the equipment susceptibility problems

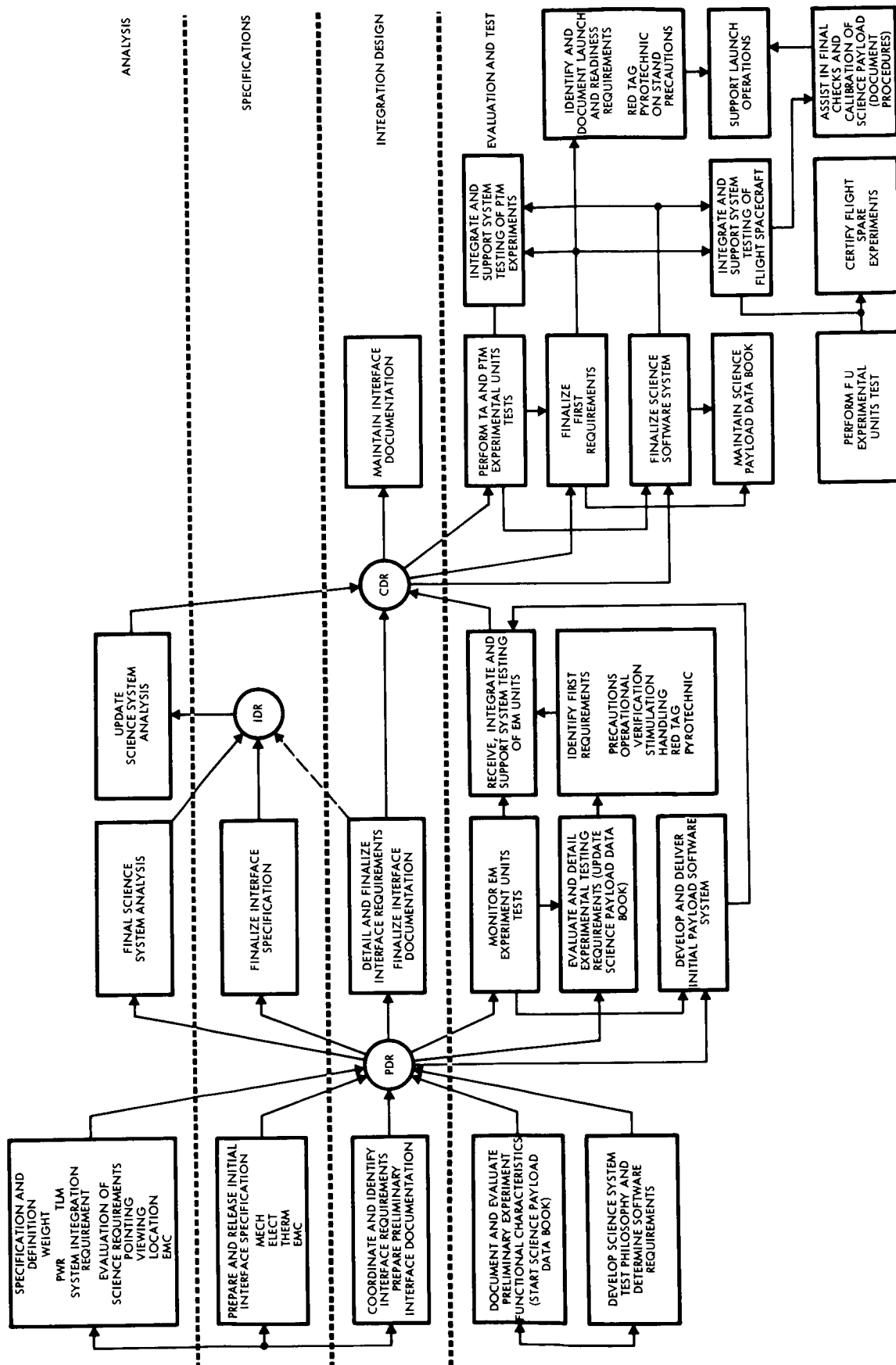


Figure 1-1
MAJOR SCIENCE INTEGRATION TASKS are performed or coordinated by equipment-responsible engineers under direction of the Project Office.



- Prepare a test matrix identifying the types of susceptibility tests to be performed on each spacecraft equipment (certain nonsensitive equipments may not require testing)
- Prepare a similar test matrix for the experiments and advise the respective experimenters to perform significant tests
- Prepare a master test plan and an addendum for each spacecraft equipment. The addendum would contain the detailed test procedures as well as the criteria which define the equipment's malfunction or degradation of performance
- Perform susceptibility tests in accordance with the test plan
- Analyze the data from this testing and data received from testing of the respective experiments and determine compatibility problems
- Recommend "fixes" to the respective experiments to reduce the interference generation to levels which will provide a margin of compatibility with spacecraft and other experiments
- Incorporate these fixes
- Retest each spacecraft equipment to determine proper implementation of "fixes"
- Prepare spacecraft/experiment interface control document specifying the maximum interference levels permitted on the respective interference lines
- Perform a conducted interference test on the experiment/spacecraft interface lines during the engineering model spacecraft test program to verify that the conducted interference levels are acceptably low.

3. SCIENCE PAYLOAD ASSEMBLY LEVEL TESTS

The experiment pre-integration testing phase covers the period from receipt of the experiment until it is certified for integration into the spacecraft. Each experiment PTM and flight model is subjected to essentially the same operations. This phase includes the environmental testing with PTM experiments undergoing qualification tests and flight experiments undergoing the acceptance tests. The main difference between qualification tests and the acceptance tests are the levels and durations of the environmental exposures.

Upon receipt of an experiment, routing and operation (R and O) cards are assigned, as is the experiment log book. Receiving inspection is then performed for possible transportation damage. The assembly level test sequence will begin with a unit bench test utilizing experiment operational support equipment to verify operational status of the experiment. The test also serves to further familiarize the experiment responsible engineer in the operation of the experiment OSE.

When its operational status is verified, the experiment will be mated to the spacecraft simulator for the purpose of checking the compatibility of experiment operation in an environment which has interfaces identical to the spacecraft. The test also serves to check the compatibility and adequacy of experiment data processing computer programs. The programs are developed to facilitate testing and data analysis and will be used on all future assembly and system level tests to maintain consistency and easy identification of data trends.

Upon completion of the operational check with the spacecraft simulator, the experiment will proceed through the environmental test sequences. If a failure should occur in any of the tests or if a noncompliance to specification is verified, it shall be cause for termination of testing. Corrective action will be determined and testing will be recycled as required through the environmental test sequence.



When all test operations have been completed, a flight certification test will be performed as a final operational test of the experiment to verify its flight worthiness prior to integration on the spacecraft.

ETO decontamination may also be required for spacecraft components in order to meet planetary quarantine constraints. If the decontamination is required, science instruments would also be required to pass ETO decontamination processing. It is also probable that clean room conditions as specified in Volume 9 will have to be maintained throughout the fabrication, assembly, and testing phases of instrument production as well as the equivalent phases of experiment integration and checkout with the spacecraft. Detailed requirements for planetary quarantine constraints would be specified in planning documents, based on the total system requirements and applied to each subsystem as required.

4. SPACECRAFT INTEGRATION AND TESTING

4.1 IN-PLANT OPERATIONS

The spacecraft integration and test phase operations involving experiments covers the period from integration of the experiments into the spacecraft through shipment of the spacecraft to the launch site. The detailed operations during this period are described in the Assembly and Integration section (Section 4) of the Implementation Plan (Volume 8), and will not be discussed in detail here.

After the mechanical installation is completed, experiment electrical integration will begin. The integration will be performed serially with each experiment taking an average of two days to complete. The detailed checks to be performed at this time are:

- Verify spacecraft side of the electrical interface
- Measurement of power transients
- Measurement of power profile
- Perform functional data checks during which the experiment will be operated in each mode of operation. Sensor stimulation, if required, will be performed at this time utilizing experimenter's OSE.

The integrated systems test (IST) is a functional check of all the spacecraft subsystems and will be performed before and after environmental exposures. The experiment tests during the IST will be more limited than those during the integration period. The experiment portion of the test procedure will be designed to verify the integrity of the experiments and that experiment data outputs are within pre-determined limits.

4.2 LAUNCH OPERATIONS

The launch operations phase includes the period from shipment to the launch site through launch of the spacecraft. Experiment testing during the launch operations again will be limited to testing similar to that conducted during the IST.

Special tasks associated with final launch preparations are:

- Arming of ordnance devices
- Cleaning of thermal surfaces
- Removing sensor covers
- Removing nonflight jumpers.

Each experimenter will be required to submit a list of all such "red-tag" items to be cared for during this period.



APPENDIX G
WEIGHT ESTIMATES

1. Hypothetical Science Payload

			<u>Weight (lb)</u>
1.1	<u>Planetary Scan Platform Mounted Instruments</u>		
Core Experiments	Photo-imaging Equipment		230
	1 - low-resolution color TV	20 lb	
	3 - medium-resolution TV	55 lb/ea	
	1 - high-resolution TV	45	
	(High-resolution IR spectrometer		30
	(Broadband IR spectrometer		20
	(IR radiometer		20
	(UV spectrometer		<u>30</u>
			(330)
	1.2	<u>Planetary Scan Platform</u>	
	Platform and box structure		78.5
	Support structure		15.0
	Thermal control		5.1
	Cryogenic control		10.0
	Control electronics		7.5
	Mars sensors (2)		14.0
	Drive mechanisms and hardware (3)		47.9
	Attachments		<u>5.0</u>
			(183.0)
		Subtotal	<u>513</u>
1.3	<u>Supporting Data Storage</u>		
	Tape recorders (4) (Photo-imaging)	20/ea	(80)
		System Total	<u>593</u>

It will be noted that the hypothetical science payload as configured by TRW is estimated to weigh approximately 513 pounds including the planetary scan platform structure, drives, etc. Throughout other volumes of this report a science payload of 400 pounds (based on previous JPL guidelines) was used for performance calculations and general weight allocation purposes. A science payload meeting this 400 pound allocation could be configured by removing one of the medium-resolution TV cameras and the low-resolution color TV camera and reducing the size of the high-resolution lens systems. In addition, since the planetary scan platform for the hypothetical payload has been configured to allow installation of the Film/TV System plus additional instruments for growth, it is probable that the 183 pounds estimated for the planetary scan platform could also be reduced. The following list summarizes the changes for a 400 pound science payload:

	<u>Weight (lb)</u>
1.4 <u>Planetary Scan Platform Mounted Instruments</u>	
Photo-imaging system	155
High-resolution IR spectrometry	30
Broadband IR spectrometer	20
IR radiometer	20
UV spectrometer	30
	<u>(255)</u>
1.5 <u>Planetary Scan Platform</u>	
Platform structure	58
Platform support structure	10
Thermal control	16
Cryogenic control	10
Control electronics	5
Mars sensors	12
Drive mechanisms	14
Electrical harness	10
Attachments	10
	<u>(145)</u>
Subtotal	<u><u>400</u></u>



2. Recommended Spacecraft - Film/TV with Core Experiments

			<u>Weight (lb)</u>
2.1 <u>Planetary Scan Platform Mounted Instruments</u>			
Core Experiments	Film/TV System		220
	Film	200	
	TV	20	
	(High-resolution IR spectrometer		30
	(Broadband IR spectrometer		20
	(IR radiometer		20
	(UV spectrometer		30
			<u>(320)</u>
2.2 <u>Planetary Scan Platform</u>			
	Platform and box structure		74.6
	Support structure		15.0
	Thermal control		5.1
	Cryogenic control		10.0
	Control electronics		7.5
	Mars sensors		14.0
	Drive mechanisms and hardware		47.9
	Attachments		5.0
			<u>(179.1)</u>
Subtotal			<u>499.1</u>
2.3 Supporting Data Storage			
	Tape recorder (1) (Photo-imaging)		<u>(20)</u>
System Total			<u>519.1</u>

3. Upgraded Spacecraft

		<u>Weight (lb)</u>
3.1	<u>Planetary Scan Platform Mounted Instruments</u>	
Core Experiments	Film/TV system - augmented	238.0
	(High-resolution IR spectrometry	30.0
	(Broadband IR spectrometry	20.0
	(IR radiometer	20.0
	(UV spectrometer	30.0
	Polarimeter	6.0
	Gamma ray spectrometer	13.0
	Meteoroid flash detector	8.5
		<u>(365.5)</u>
3.2	<u>Body Mounted Instruments</u>	
	Mass spectrometer	8.5
	Solar occultation	18.0
	Cosmic ray detector	22.0
	Micrometeorite impact	16.5
	Solar Plasma	5.0
	Solar x-ray	7.5
	Celestial x-ray	4.5
		<u>(82.0)</u>
3.3	<u>Boom Mounted Instruments</u>	
	Magnetometer	3.0
	Magnetometer boom and mechanisms	8.5
	Neutron albedo	12.0
	Neutron albedo boom and mechanisms	8.5
		<u>(32.0)</u>

-Continued-



		<u>Weight (lb)</u>
3.4	<u>Planetary Scan Platform</u>	
	Platform structure	74.6
	Support structure	15.0
	Thermal control	5.1
	Cryogenic control	10.0
	Control electronics	7.5
	Mars sensors	14.0
	Drive mechanisms and hardware	47.9
	Attachments (est)	5.0
		<u>(179.1)</u>
	Subtotal	<u><u>258.6</u></u>
3.5	<u>Supporting Data Storage</u>	
	Tape Recorder (1) (Photo-imaging)	(20)
	System Total	<u><u>678.6</u></u>



APPENDIX H

PLANETARY SCAN PLATFORM CONTROL SYSTEM DESIGN

1. DESIGN IMPLEMENTATION

The design of the planetary scan platform control system shown in Figure 3-14 is implemented with a maximum of space flight proven hardware. Component and circuit redundancy is applied where necessary to enhance the control system reliability. Special attention is focused on an optimum design to provide minimum weight and volume.

The equipment provided for the planetary scan platform control system is:

- Mars trackers and associated electronics
- Planetary scan platform drive electronics
- Planetary scan platform gimbal drives

A description of this equipment together with the basic principles of operation and the preliminary specifications is given in the following paragraphs.

1.1 MARS TRACKER AND TRACKER SERVO SYSTEM

1.1.1 Mars Tracker

The planet sensor selected for tracking Mars is a modified version of the horizon sensor which has been successfully flight tested in the Orbital Geophysical Observatory (OGO). It utilizes four infrared search-track units, shown in Figure H-1 to track the Mars horizon in four planes 90 degrees apart in azimuth. The four search units scan the Mars horizon and measure the elevation angle: ϕ_A , ϕ_B , ϕ_C , ϕ_D , as shown in Figure H-1.

These elevation signals in analog units are fed to a switching matrix where appropriate signals for pitch and roll computation are selected. Selection is based upon sun presence and track-check information from each tracker. If the sun is in the tracker field-of-view, or if the scanner is not tracking, that tracker signal is disqualified and the

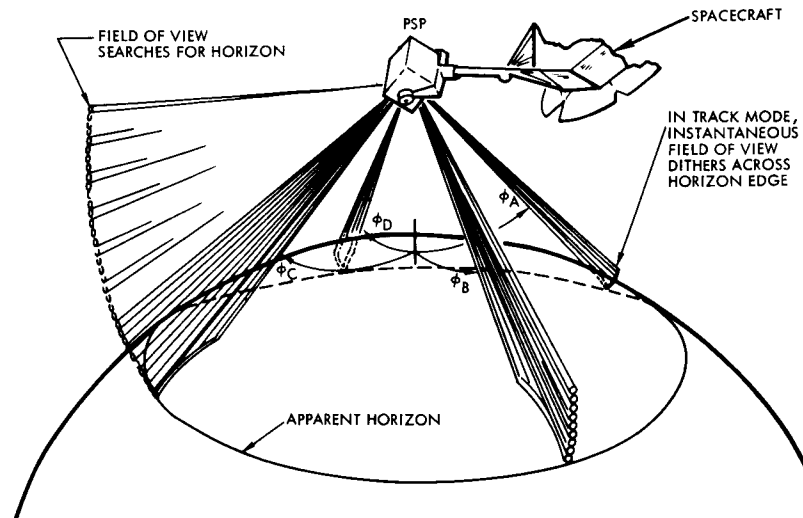


Figure H-1
HORIZON EDGE-TRACKING SCHEME is employed in the PSP control system for the Mars tracking mode.

remaining three signals are utilized in the computation. Thus, if the C tracker was disqualified, the pitch and roll information would be computed as follows:

$$\text{Pitch} = \frac{1}{2} (\phi_B + \phi_D) - \phi_A$$

$$\text{Roll} = \frac{1}{2} (\phi_B - \phi_D) \quad .$$

1.1.2 Tracker Servo System

The closed loop Mars tracking servo shown in Figure H-2 consists of a telescope, bolometer, signal preamplifier, signal amplifier, Schmitt trigger, drive amplifier and a flexure mounted mirror driven by a d'Arsonval type mechanism called a positor. The positor oscillates at 20 Hz with a 2.5 degrees peak-to-peak amplitude, causing the telescope field of view to dither about the planet horizon. If the horizon is centered within the field of view's oscillation pattern then the output of

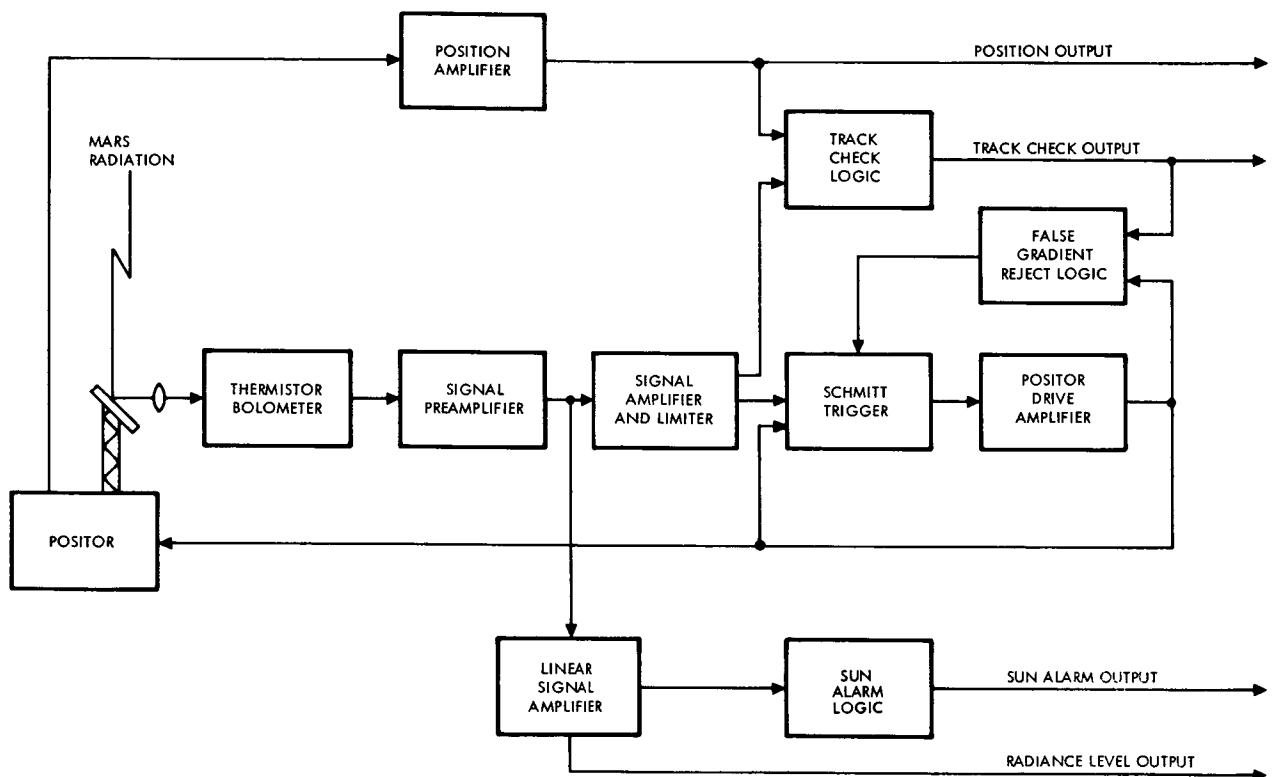


Figure H-2
MARS TRACKING SERVO is used in the Mars Track mode with a pointing accuracy of ± 0.25 DEGREE.

the Schmitt trigger will have a zero DC component. If the horizon is displaced from the center of oscillation, the resulting DC component in the Schmitt trigger output when applied thru the drive amplifier will cause the positor to recenter the oscillating field of view on the planet horizon. The position amplifier output is derived from a position pickoff coil on the positor and is proportional to the angle, ϕ , between the center of field of view oscillations and the planetary scan platform pointing axis as shown in Figure H-1. As described in the previous paragraph, the position outputs from the four trackers are fed to the central electronics and processed for the derivation of pitch and roll information. A detailed description of the design of the tracker is given in Section H-2 of this appendix.

1.1.3 Preliminary Specification of Mars Sensor

The preliminary specification of the Mars sensor is given in Table H-1.

Table H-1. Preliminary Specification of Mars Sensor

Function

To sense the infrared radiation from the planet's horizon and compute the pointing error between the planetary scan platform axis and the planet center.

Design Configuration

Two sensor head assemblies are mounted orthogonally on the planetary scan platform. Signals from these assemblies are processed in a central electronics unit. The primary components of the sensor are:

(4) Detectors:	Silicon immersed bolometer
(4) Scan mechanisms:	Vibrating mirror (Positor)
Electronics:	Signal processing Positor sweep Voltage regulator Summing matrix

Physical Characteristics

Size:	Sensor Heads	70 in. ³ ea
	Central Electronics	170 in. ³
Weight:	Sensor Heads	5.5 lb ea
	Central Electronics	6.0 lb
Power:	12 watts	

Performance

Track Range:	±83 degrees
Altitude Range:	500 to 30000 km
Linear Interval:	±30 deg (low altitude) ±10 deg (high altitude)
Spectral Range:	18 to 35 microns
Instantaneous Field-of-View	1.2 deg sq
Accuracy:	±0.25 deg (ea axis)



1.2 PLANETARY SCAN PLATFORM DRIVE ELECTRONICS

1.2.1 Function

The basic function of the planetary scan platform drive electronics in the PSP control system is to process the position or rate command signal and to impress a voltage on the AC motor to drive the planetary scan platform. In the position control mode, the Computer and Sequencer or command system commands the gimbal angle in the shaft or yoke axis by loading an input storage register. A digital comparator compares the output of the input register to that of a shaft position register. The output of the digital comparator is converted to analog form to control an analog controller (motor driver) which provides the motor drive current. Gimbal position feedback is obtained from an incremental encoder mounted on the gimbal axis trunnion. The output pulses from the encoder are entered into the feedback register, which accumulates the gimbal angle. This information is then used for comparison in the digital comparator and for telemetry. Rate damping is obtained by tachometer feedback.

Since the use of an incremental encoder requires that the feedback register be loaded serially, false gimbal angle indication may occur if a pulse is lost. To minimize this problem, a feedback reset pulse is provided from the encoder to reset the register each time the encoder passes through null.

During the Mars tracking mode, the output of the Mars tracker is fed into the motor driver through the planetary scan platform drive electronics to cause the planetary scan platform to track the center of Mars. For pointing in directions away from the planet local vertical, a bias signal proportional to the desired offset angle is added to the Mars tracker output.

During image motion compensation the inputs are initial angle settings followed by a rate signal from a programmed sequence. This rate signal is obtained by opening the encoder feedback signal and inserting a rate command in the input register. This signal is summed with the tachometer feedback to produce the desired angular motion.

For all modes of operation, both gimbal angle and motor shaft speed are obtained from the feedback register for telemetry back to earth.

1.2.2 Design

The planetary scan platform drive electronics consists of three independent subsystems, each composed of a position counter and control, an input register, an adder and a digital-to-analog converter, run in parallel from three sets of independent encoder amplifier outputs to three operational-amplifier summing junctions. The three summing operational amplifiers are connected at their outputs to form a majority voting circuit with the resulting analog error signal sent to the motor drive circuit.

The adder chain, input register, analog-to-digital converter, the position counter and its control logic are shown in Figure H-3. The position counter accepts two phase square-wave signals from the incremental encoder amplifiers and its output is compared with the input register so as to provide the desired planetary scan platform position.

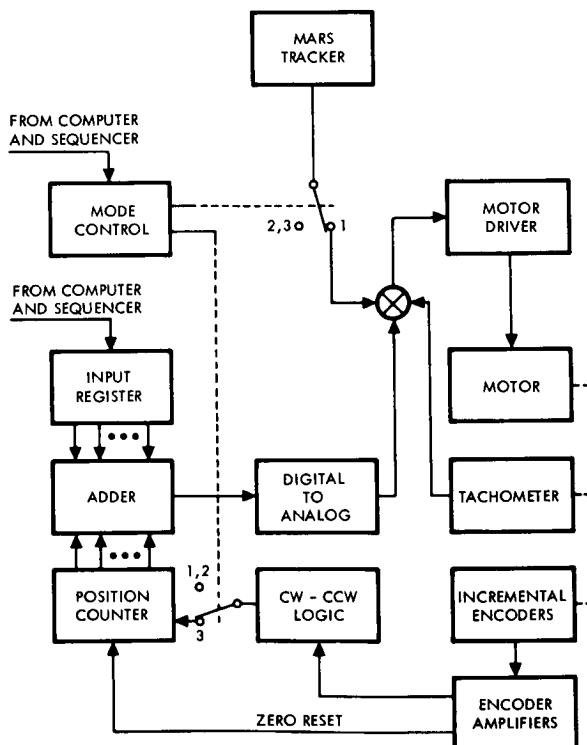


Figure H-3

PSP DRIVE ELECTRONICS accepts the command inputs in three distinct modes of operation; Mode 1 - Mars Track with or without offset; Mode 2 - Rate Command and Mode 3 - Commanded Position.



Since the encoder is quantized to 0.1 deg and the planetary scan platform may have a motion of ± 180 deg from null (3,600 position increments), the position counter accepts 12 bit information allowing a range of 4,096 position increments.

A more detailed description of this design with alternate approaches is given in Section H-3 of this appendix.

1.2.3 Preliminary Specifications of the Planetary Scan Platform Drive Electronics

Table H-2 delineates the preliminary specifications for the planetary scan platform drive electronics.

Table H-2. Preliminary Specification of Planetary Scan Platform Drive Electronics

Function

Processes inputs from computer and sequencer, or Mars tracker and provides drive power to the gimbal assembly actuators. Gimbal position readout about all axis is provided to the telemetry system.

Physical Characteristics

Size	Power	50VDC	50VAC
7x6x6 inches			
Weight	Shaft Axis (avg)	10 watts	15 watts
7.5 lb total	Hinge Axis (avg)	10 watts	20 watts
for all axis	Yoke Axis (avg)	10 watts	17 watts

Operational Characteristics

Mode 1: Mars Tracking. This mode is normally entered from Mode 3; it utilizes error signals from a Mars tracker to maintain pointing along the planet local vertical. For pointing in directions away from the local vertical the output of the tracker is biased accordingly.

Mode 2: Image Motion Compensation. During photographic sequences, this mode provides image motion compensation by causing the boresight axis of the planetary scan platform to point at a fixed point on the planet surface. Angles and/or single rate commands are issued by the computer and sequencer subsystem. Normally, Mode 2 would be

entered from Mode 1 just before photographic exposure to provide image motion compensation during the exposure period.

Mode 3: Gimbal Angle Command. The planetary scan platform is pointed in selected directions based on gimbal angle commands from the computer and sequencer.

Planetary Scan Platform
Drive Electronics Inputs

Computer and Sequencer and
Mars Tracker about two axis

Planetary Scan Platform
Drive Electronics Outputs

Motor drive signals to gimbal axis
actuators. Telemetry position pick-
offs of gimbal axis actuators

1.3 PLANETARY SCAN PLATFORM GIMBAL DRIVES

1.3.1 Design

To meet the operational position and rate control requirements of the science payload, the PSP is provided with three gimbal drives. These drives are:

- Hinge axis drive
- Shaft axis drive
- Yoke (or platform) axis drive.

A mechanical schematic of the arrangement of these drives is shown in Figure H-4. The hinge axis drive is used for deployment of the planetary scan platform from the stowed position against the spacecraft. This drive is also used to position the main shaft of the planetary scan platform perpendicular to the orbit plane if it is desired to control the planetary scan platform in a three-degree of freedom orientation mode (in conjunction with a commanded spacecraft roll maneuver.)

The other two drives in the shaft and yoke axes function as two orthogonal gimbal-type drives and are used to point the planetary scan platform at the planet in a closed loop control mode using the Mars sensor or in commanded open loop modes. The open loop modes are controlled by the computer and sequencer subsystem to perform such functions as image motion compensation for the photo-imaging equipment or to point at or track certain locations off the local vertical

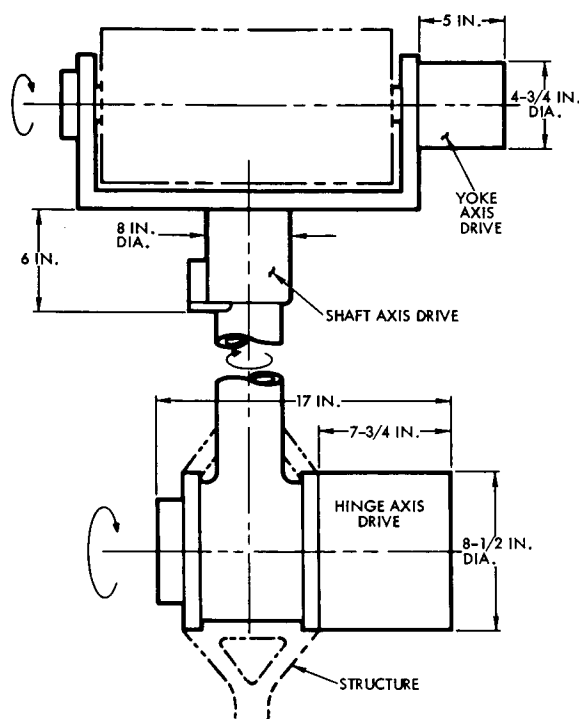


Figure H-4
PLANETARY SCAN PLATFORM GIMBAL DRIVES used in PSP control system for Yoke, Shaft and Hinge axis drive. All use wobble gears with static seals of metallic bellows to provide isolation from the space environment.

center for stereo photography or for viewing parts of the planet as required by certain science equipment.

The gimbal drives of all axes employ AC motors and wobble gear drives. A typical wobble gear drive with the metallic bellows static seal which provides isolation of the motor and gear train from the space environment is shown in Figure H-5. Rate operation between ± 4.05 milliradian/sec with an accuracy of ± 0.15 mr/sec is achievable by tachometer feedback while position control in the open-loop is obtained by feedback signal from the incremental encoder attached to the planetary scan platform gimbal drive output. The attainable position control accuracy is about 0.14 degree. A single axis gimbal drive functional block diagram is shown in Figure H-6.

A detailed analysis of the operation of all three axes is given in Section H-4 of this appendix.

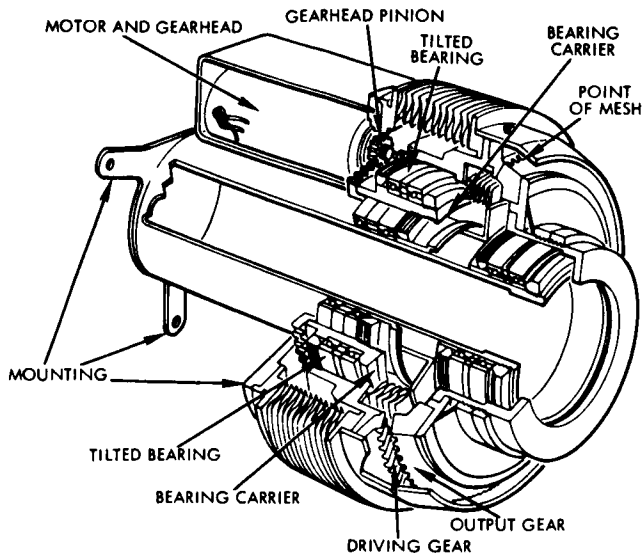


Figure H-5
WABBLE GEAR DRIVE used in the gimbal drive of the PSP Control System. The equipment has been successfully flight tested in the OGO Spacecraft and proved its reliable operation in the space environment.

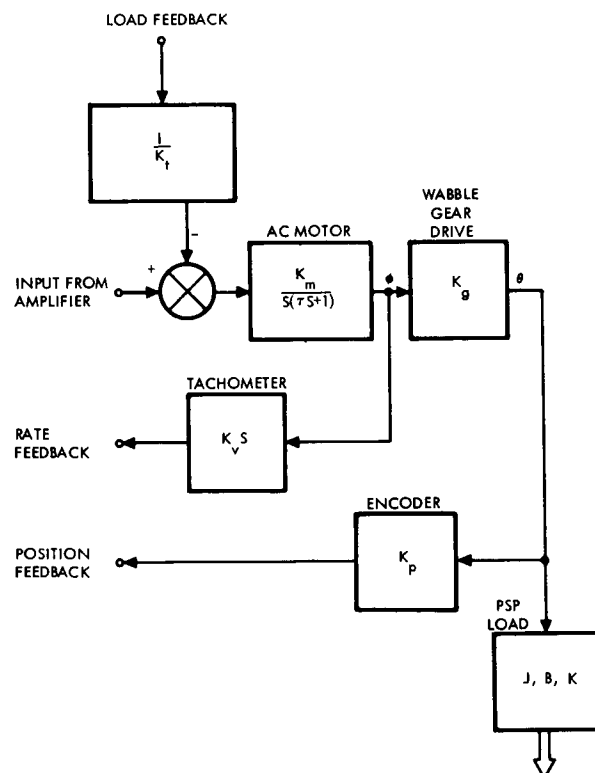


Figure H-6
PSP GIMBAL DRIVE, one for each axis of the PSP Drive Control System, incorporates an AC magnetic tachometer for feedback to augment stability in the position control mode and to control the PSP rate in the rate or IMC mode. It uses a digital incremental encoder to feedback gimbal angle for open loop PSP position control. Both outputs are telemetered back to earth for operational as well as engineering data analysis.



1.3.2 Preliminary Specifications for Planetary Scan Platform Gimbal Drives

The preliminary design specifications for the planetary scan platform gimbal drives are given in Tables H-3 and H-4

Table H-3. Preliminary Specifications of Planetary Scan Platform Yoke Axis Gimbal Drive

Gimbal travel	±180 deg
Image motion compensation	This gimbal shall be driven at a rate of ±10 mr/sec ±0.15 mr/sec
Gimbal rates	
Slew	10 mr/sec maximum
Angular acceleration	0.5 mr/sec ² (nominal)
Stability	0.075 mr/sec maximum random velocity
Load torque	195 in-lb, nominal load at 10 mr/sec. Capable of withstanding an external load of 4000 in-lb at zero speed.
Input voltage	36 volts, 2φ, 400 Hz for drive 28 VDC for heater
Power	12.5 watts, running and peak
Pointing accuracy	±0.143 deg relative to the Mars sensor reference
Gimbal axis encoder	0.1 deg (three digital encoders will be provided for redundancy)
Rate generator accuracy	±1% of absolute rate or 0.1 mr/sec, whichever is greater
Drive and encoder weight	6.0 lb maximum total
Seals	All high speed elements will be hermetically sealed within an inert atmosphere

Table H-4. Preliminary Specifications of the Planetary Scan Platform
Shaft and Hinge Axis Gimbal Drive

Planetary scan platform control axes

Shaft hinge axis	Axis of this drive about which planetary scan platform is deployed from spacecraft
Shaft axis	Axis of this drive about which the planetary scan platform boom is rotated
Image motion compensation	The planetary scan platform shaft axis can be driven at a rate of -10 mr/sec to +10 mr/sec, ± 0.15 mr/sec
Gimbal travel	
Hinge axis	± 100 deg plus stowage
Shaft axis	± 180 deg
Gimbal rates: stability	0.075 mr/sec (maximum random velocity each axis)
Slew	0.35 mr/sec ± 0.15 mr/sec (minimum, shaft axis only), 10 mr/sec maximum
Angular acceleration	0.5 mr/sec^2 (minimum each axis)
Load torque (maximum)	
Hinge axis	41,000 in - lb (design load carrying capability at zero speed only at Mass Orbit Insertion)
Shaft axis	195 in - lb (nominal load at 10 mr/sec. Capable of withstanding an external load of 4,000 in - lbs at zero speed)
Input voltages	
Drives	36 volts, 2 ϕ , 400 Hz
Heaters	28 VDC
Power	
Hinge axis	15 w running and peak
Shaft axis	11 w running and peak



Table H-4. Preliminary Specifications of the Planetary Scan Platform
Shaft and Hinge Axis Gimbal Drive (Continued)

Design parameters

Pointing accuracy	± 0.143 deg relative to the Mars sensor reference
Gimbal axis encoder	± 0.1 deg (three digital encoders will be provided for each axis for redundancy)
Rate generator accuracy	$\pm 1\%$ of absolute rate or 0.1 mr/sec, whichever is greater
Cable	A cable wrapup configuration will be provided
Drive weight	20.8 lb total (both drives)
Seals	High speed elements hermetically sealed in inert atmosphere

2. DESIGN ANALYSIS OF MARS TRACKER

This section presents a design analysis of the horizon edge tracker based on a Mars radiance model and the spectral bandwidth selection. The results of the analysis led to the choice of the OGO earth sensor for Mars tracker. Details of the physical characteristics and pointing accuracy of the selected tracker are given. Alternate approaches are discussed in the last paragraph of this section.

2.1 MARS RADIANCE MODEL AND SPECTRAL BANDWIDTH SELECTION

The spectral passband of the sensor has been chosen to be the interval 18 to 35 microns, for the following reasons:

- 1) Water vapor density is low and too uncertain for using an H_2O absorption band near 33 microns.
- 2) Energy in the water vapor window, 8 to 12μ , is high, but radiance level variations over the planet disc are about 30 to 1. This is based upon a black-body emittance model with surface temperature varying from $175^\circ K$ to $300^\circ K$.

- 3) Presence of CO₂ is well established, but the frequency of variations in the CO₂ horizon are unknown. Using the same temperature model, the energy emitted in the 14 - 16μ CO₂ absorption band varies by a factor of 10 over the planet disc.
- 4) Planetary surface radiation in the 18 - 35μ region is still high and variation is only six to one. The frequency of cloud occurrence is low. The use of surface radiation rather than atmospheric absorption is suggested.

The effective planet radiance within the spectral band of the Sensor may be computed from the following formula:

$$N_e = \int_0^{\infty} N_{\lambda} R_{\lambda} d\lambda \quad \text{watts/m}^2/\text{ster},$$

where

N_{λ} = Planet spectral radiance (watts/m²/ster/μ)

R_{λ} = Sensor spectral response

λ = Wavelength

N_e = Effective radiance

Using the appropriate radiance levels for a black body source temperature of 175°K and the sensor's response as shown in Figure H-7 results in a value for N_e of 2.1 watts/m²/ster. The equivalent value for the high temperature extreme of 300°K is 9.0 watts/m²/ster. The expected contrast ratio is 4.2 to 1.

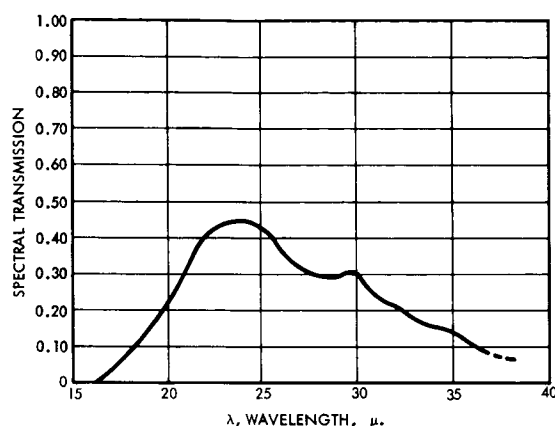


Figure H-7
SPECTRAL TRANSMISSION OF SENSOR OPTICS in Mars tracker was selected such as to minimize the contrast seen by the sensor.

2.2 SELECTION OF THE OGO SENSOR FOR MARS TRACKER

OGO Earth Sensor test data show that the tracker will not lock onto a false gradient minimum of 4.2 to 1 within the planet's disc as long as the minimum radiance level is sufficiently high. The typical minimum radiance level for OGO operation at these contrast ratios is around 3 to 4 watts/m²/ster. Circuit parameters do not allow operation at the 2.1 watts/m²/ster minimum level for the Mars application. However, successful operation at levels below 2 watts/m²/ster are possible using a modified version of the OGO Earth Sensor called the AOGO Sensor. The AOGO sensor is designed to operate at lower radiance levels since its spectral band is limited to only 14 to 16 microns. Therefore, it is adaptable to the Mars mission by proper choice of circuit parameters and sensor spectral acceptance band.

2.3 PHYSICAL CHARACTERISTICS

All electronics in the horizon scanner system, with the exception of components internal to the positor and internal to the telescope

detector, are packaged in 33 modular units. Most of these modules consists of components stacked in a logwood fashion, with leads extending through etched circuit boards on each end of the stack. The leads are soldered to the boards and beryllium-copper studs provide posts for wiring interconnections between modules. The modules are held in place in the scanner by countersunk flathead screws, entering from the backside of the dual tracker heads and central electronics unit. In addition, the modules are epoxy-bonded to the scanner structural members to provide more thermal dissipation.

The telescope is mounted into a bracket assembly and fastened by screws to the dual-tracker-head baseplate. The dual-tracker-heads are optically aligned by rotation of the telescope's assembly about a line in the surface of the mirror.

The positor scanning mechanism is mounted in the dual-tracker-heads by four mounting screws, rigidly attaching them to the rear surface of the baseplate.

Six separate connectors are supplied with the horizon scanner assembly, one on each dual-tracker-head and four on the central electronics unit.

2.4 ACCURACY

Error sources contributing to uncertainties in the pitch and roll readouts are listed in Table H-5. Treating these as statistically independent, the RSS (3σ) error is computed to be ± 0.25 degree for each axis.

Table H-5. Error Tabulation - Mars Tracker

<u>Error Source</u>	<u>Tolerance Assumption</u>	<u>Error Magnitude</u>
Readout Nonlinearity	$\pm 0.25\%$	± 0.10 deg
Schmitt Trigger Threshold	$\pm 50\%$	± 0.05 deg
Schmitt Output Offset	$\pm 1.0\%$	± 0.03 deg
Noise (0.6 Hz Bandwidth)	± 0.02 deg	± 0.02 deg
Spacecraft Alignment	± 0.05 deg	± 0.05 deg
Temperature Variance	$0^\circ\text{F} - 113^\circ\text{F}$	± 0.12 deg

RSS (3σ) Total for Two Scanners = ± 0.25 deg each axis



2.5 ALTERNATE APPROACHES

Other systems that could be used for tracking Mars are compared with the modified OGO in Table H-6. Specified accuracies are based upon previous analysis for each sensing mission and are appropriately scaled for application to Mars tracking. Selection of the AOGO system with minor modifications is based upon its estimated accuracy and flight-proven capability. Alternate approaches may conceptually satisfy mission requirements, but probably require a longer development lead time. The alternate configurations, along with other devices such as those sensors manufactured by Quantic Company, can be re-evaluated later as their development status advances. the AOGO Sensor's margin of performance at the lowest radiance extreme is not as great as would be desirable. If later data shows that the contrast ratio or minimum radiance level is much less than assumed, then the alternate approaches should be investigated in greater detail.

Table H-6. Mars Tracker Candidate Systems

<u>Sensor Model</u>	<u>AOGO</u>	<u>*LPHS</u>	<u>Edge Tracker</u>
Manufacturer	TRW Systems	Barnes Engineering Co.	Barnes Engineering Co.
Development Status	**Operational	Engineering Model	Engineering Model
Sensor Type	Edge Tracking	Sampled Detector Array	Edge Tracking
Pitch and Roll Accuracy	±0.25 degree	±0.45 degree	±0.30 degree
Output Type	Analog	Digital	Analog
Size (Ft ³)	0.174	0.095	0.161
Weight (lbs)	17.0	15.0	14.0
Power (watts)	12	5	5
Reliability (1 yr)	0.95	0.97	0.97

* LPHS - Lunar and Planetary Horizon Sensor

** The system, excluding the Proposed Modifications, is operational.

3. DESIGN ANALYSIS OF THE PLANETARY SCAN PLATFORM DRIVE ELECTRONICS

3.1 DETAILED DESIGN OF THE PLANETARY SCAN PLATFORM DRIVE ELECTRONICS

The functional block diagram of the planetary scan platform drive electronics is shown in Figure H-3. It consists of the input register, adder, digital-to-analog converter, the position counter and its control logic shown in more detail in Figure H-8 through Figure H-12.

The contents of the position counter and input register (a conventional shift register) are compared in a binary adder to provide a digital error signal proportional to the difference between the position counter contents and the input register contents. This error signal is in turn fed into a weighted-resistor operational amplifier digital-to-analog converter to provide an analog signal to the summing junction. Since a twelve bit digital-to-analog converter (corresponding to the adder output) is complex to mechanize in practice and since it is not necessary that the error signal be linear over its full range, a six bit digital to analog converter which saturates for an input error equivalent to more

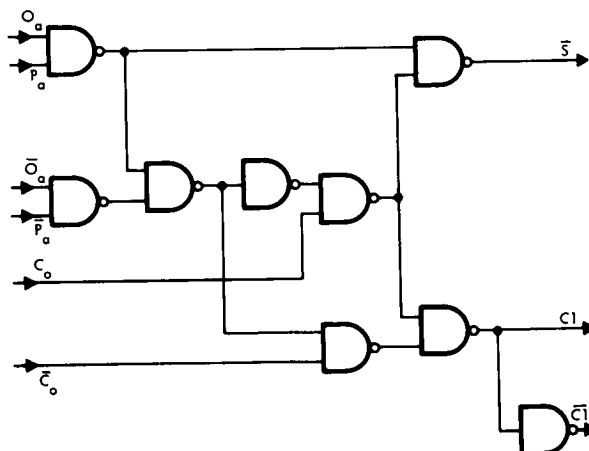


Figure H-8
ADDER ELEMENT is used in the PSP Control System Drive Electronics.

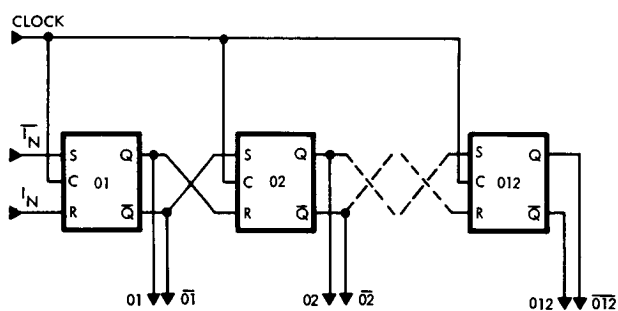


Figure H-9
INPUT REGISTER (12 bits) is used in the PSP Control System Drive Electronics.

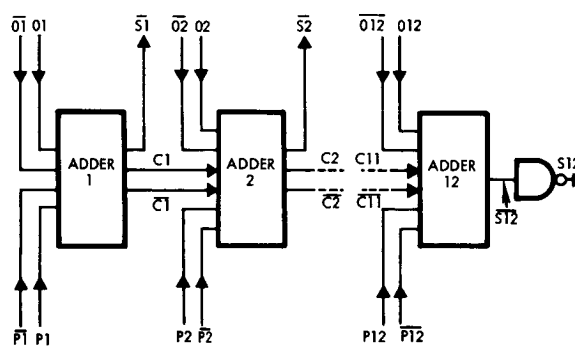


Figure H-10
ADDER CHAIN (12 bits) is used in the PSP Control System Drive Electronics.

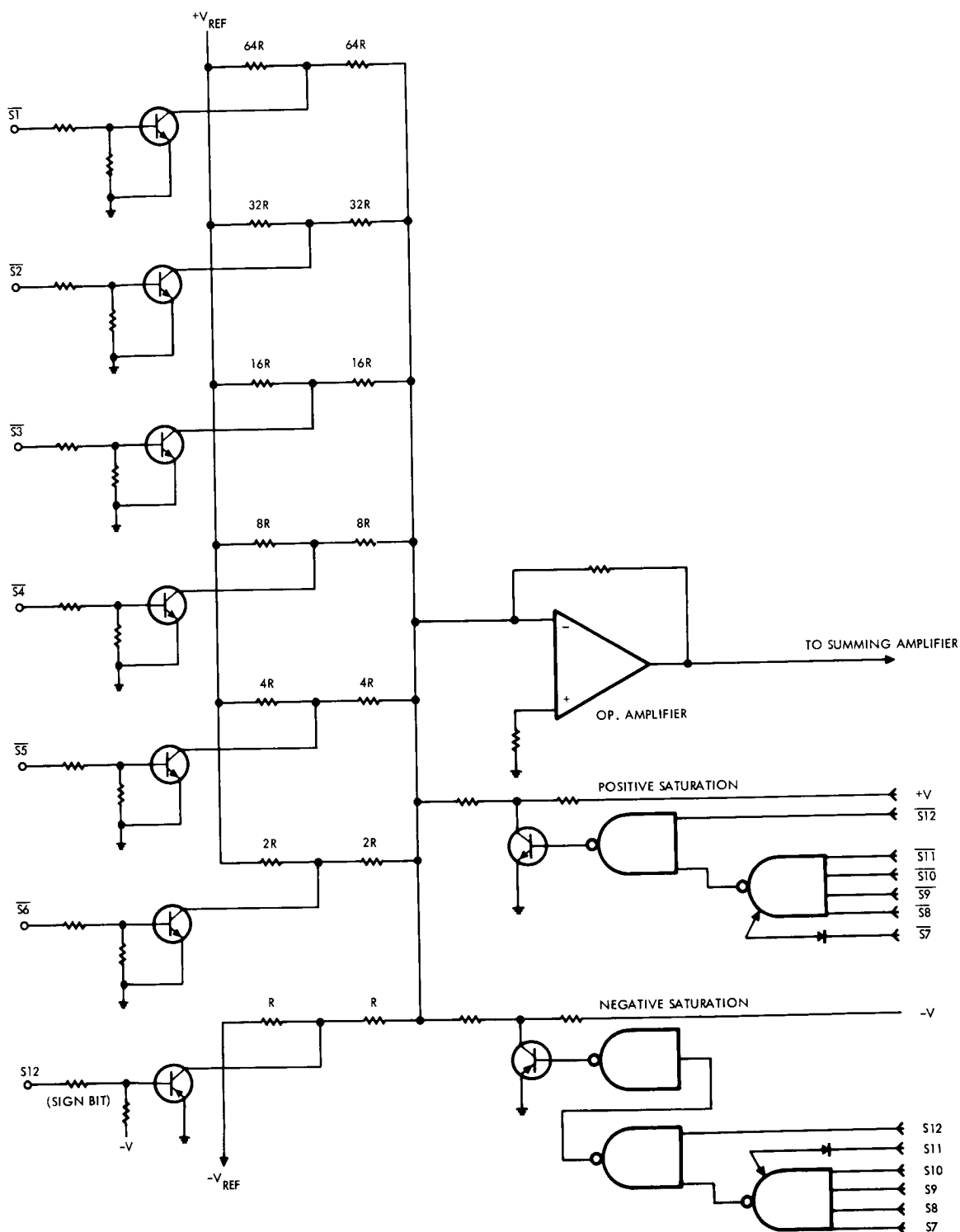


Figure H-11
SATURATING DIGITAL TO ANALOG CONVERTER is used in the PSP control system drive electronics

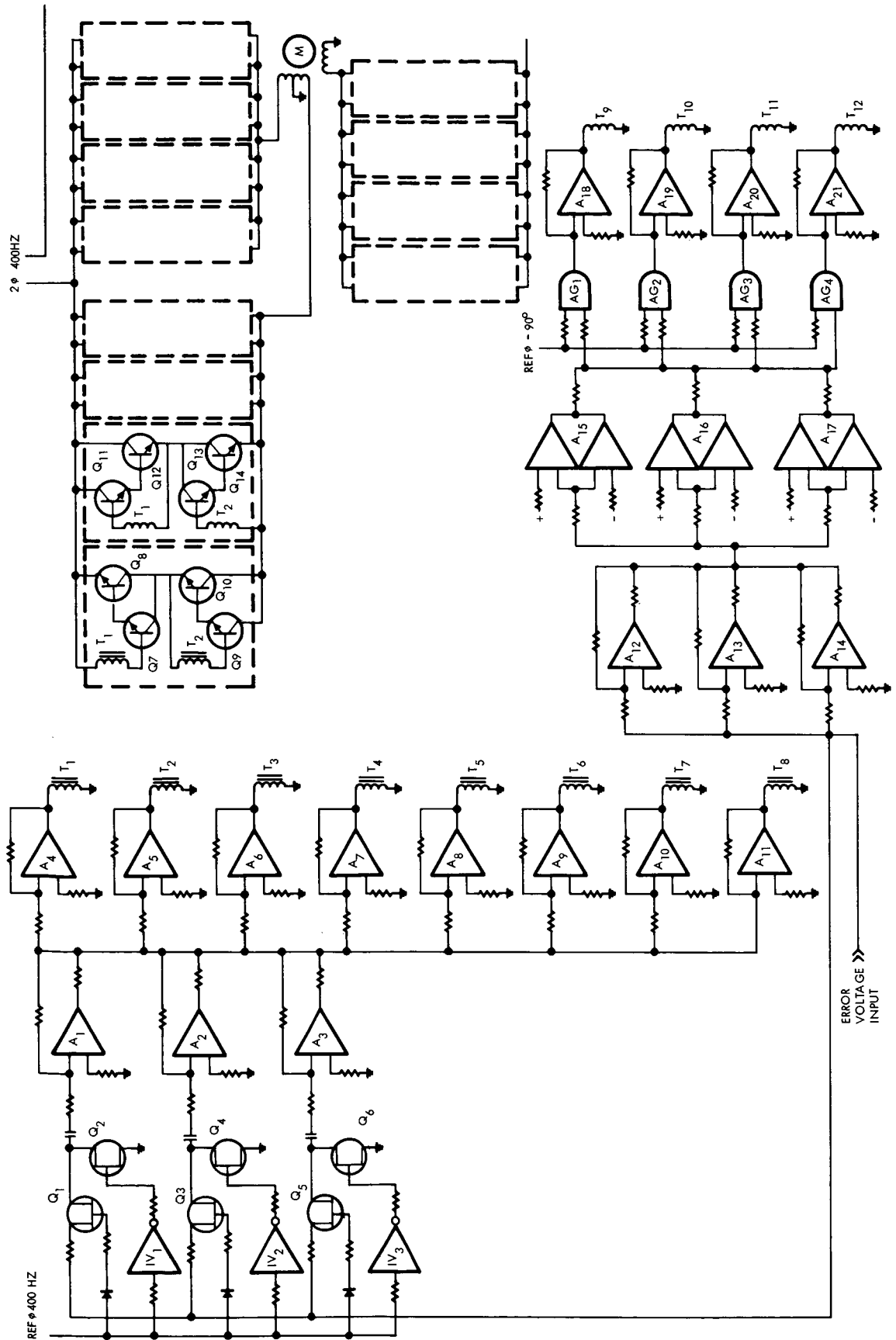


Figure H-12
PSP MOTOR DRIVER is used in the PSP control system drive electronics.

than ± 6.4 degree is used. The saturation level could, if necessary, be extended beyond this limit. As shown in Figure H-11, the six least significant bits of the adder output, together with the sign bit, are used to control the linear portion of the digital-to-analog converter. Any larger errors are sensed and gating circuits are used to drive the operational amplifier into saturation. Thus, an easily mechanized seven bit digital-to-analog converter provides the required 0.1 degree accuracy near null and an error signal of the correct polarity for errors over the range of ± 180 degrees.

Figure H-8 shows an adder element. Each information bit (P_a) and its negation ($\bar{P}_a$) from the position counter is compared to the corresponding bit (0_a) and its negation ($\bar{0}_a$) from the input register shown in H-9. Such adder elements are combined in the adder chain (Figure H-10) which produces an output equal to the difference in the binary numbers generated by the input and position registers. If there is a one-to-one correspondence in all bit positions from 1 to 12, the angular position of the corresponding planetary scan platform axis is precisely where it has been commanded. If a difference in bits exists, the output of the adder will be a measure of such difference and will be used to develop an error signal that will command the corresponding planetary scan platform motor drive to null such difference.

Position feedback information is obtained from an incremental encoder mounted on the planetary scan platform output. The position outputs from the incremental encoder amplifiers are two square-wave trains shifted 90 degrees relative to each other provided that the direction of the planetary scan platform rotation is sensed by the phase relationship of the two signals. The position counter counts either up or down depending on the direction of planetary scan platform motion.

The output pulses from the encoder amplifiers are entered into the feedback counter register which will accumulate the absolute planetary scan platform angle. Since the planetary scan platform encoder is quantized to 0.1 degree, the position counter must accept 12 bit information. A zero reset capability is provided by the encoder. When the encoder presses through null, a pulse is emitted, amplified and fed into the reset input of the register.



The planetary scan platform motor driver power amplifier, shown in Figure H-12, amplifies the DC error voltage existing from the motor control loop summing junction and provides bidirectional drive to a 400Hz two phase servo motor. The DC error voltage is chopped at a 400 Hz rate by three parallel operated series-shunt FET choppers (Q_1 thru Q_6) and referenced to a single phase of two phase power. Since the FET's must be switched alternately, integrated-circuit inverting gates IV_1 , IV_2 and IV_3 are provided. The output of each of the series-shunt choppers is amplified by an integrated-circuit operational amplifier and outputs are summed together. The summed output of the amplifiers thus provides a 400 Hz square wave voltage in phase with the switching-reference drive for one polarity of DC error voltage and 180 degrees out of phase for the other polarity of the DC error input voltage. In each case the amplitude of the square wave is proportional to the magnitude of the error voltage. Because of the parallel arrangement of the chopping network, no single failure of any component within the channel will significantly affect the output. The in-phase or out-of-phase square wave is amplified by integrated-circuit operational power amplifiers A_4 thru A_{11} , to a power level sufficient to drive gating transformers T_1 thru T_8 . The secondaries of each gating transformer drive darlington-configured switching transistors. Transformers T_1 thru T_4 drive series-parallel switching transistors which apply 400 Hz power to the motor for one drive polarity. Similarly transformers T_5 thru T_8 gate 400 Hz power to the motor for the other drive polarity. In each case the voltage applied to the motor is proportional to the amplitude of the square wave and the DC error input voltage. The series-parallel configured drivers provide sufficient redundancy such that no single transistor or gating transformer in any failure mode will modify the power applied. The reference winding of the 400 Hz servo motor is powered identically with the exception that full voltage must be supplied to the winding when any amount of DC error voltage exists. The DC error voltage is first amplified by integrated-circuit operational amplifiers A_{12} , A_{13} and A_{14} mechanized in a self-redundant configuration. The amplified error is first applied to three integrated-circuit dual comparators which supply positive voltage logic levels at their outputs when the DC

amplified error, equivalent to less than one bit of positioning error, goes positive or negative. The summed output is applied to one input of two input redundant "and" gates AG_1 thru AG_4 . The other input to the "and" gates is a logic level equivalent to the 90 degree reference 400 Hz square wave. When a DC error input is sensed, the 400 Hz -90 degree phase square wave is gated to the input of the integrated circuit operational power amplifiers as discussed before. Thus full voltage is applied to the reference winding power whenever the sensed error signal exceeds a threshold equivalent to less than one bit of positioning error.

3.2 PHYSICAL CHARACTERISTICS

The planetary scan platform drive electronics is packaged in a manner similar to the slice and loaf technique used on previous TRW control systems. The assembly is made up of slices which contain units such as shown in Figure H-13. A slice is a chassis which contains two circuit boards electrically interconnected. The slices are bolted together and connected through external electrical connectors. Such units, each forming an assembly for one PSP axis drive, will be built and tested as an entity. This packaging scheme facilitates fabrication, checkout and systems test. Therefore, its development time is substantially reduced.

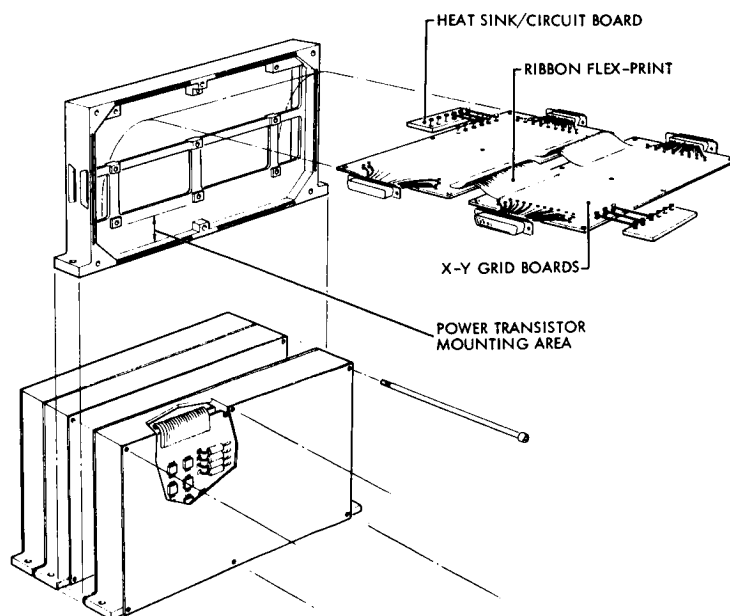


Figure H-13
SLICE AND LOAF PACKAGING is employed in the PSP Drive Electronics.



3.3 ALTERNATIVE APPROACHES

Tradeoff studies were made in the area of motor drive power generation. One approach used AC drive power and the other used DC. It was determined that the use of AC power and switching connections to the motor control phase to control direction provided a system with a better output conversion efficiency. Since this to a great extent determines subsystem efficiency, this approach was considered to be the better choice from a standpoint of power consumption and total count.

4. DESIGN ANALYSIS OF PLANETARY SCAN PLATFORM GIMBAL DRIVE

4.1 PLANETARY SCAN PLATFORM GIMBAL DRIVES

The planetary scan platform gimbal drive consists of the drive motor, wobble gear drive, tachometer and digital incremental encoder. The drive motor and wobble gear drive are housed together in a sealed unit. There are no dynamic seals in this design. All rolling and sliding surfaces of the motor, gearhead, associated gears and ball-bearings are hermetically sealed in pressurized inert gas for protection against the effects of the space environment, except for one pair of gears and one ball-bearing pair specially designed to operate at low speed in vacuum. These parts are fabricated of special materials and employ solid lubricant coatings not subject to cold welding or sublimation. Sealed lubricants are radiation resistant.

Sealing is accomplished by two bellows which are installed between the high speed parts of the mechanism and the output driving gear. The unique feature of the drive mechanism is the use of a pair of specially cut wobble gears for the output stage. Action of the bearing carrier and a tilted bearing internal to the unit produces a non-rotating conical nutation motion (or wobble) of the driving gear at the end of the main bellows. This motion causes rotation of the output gear and shaft by sequential engagement of a limited number of gear teeth. Typical wobble gear drives (both with motor external and internal) are shown in Figure H-5.

The motor is a 36-volt, 400 Hz, 2-phase AC servomotor (a high rotor resistance, induction motor, often referred to as a 2-phase control motor). It consists of a stator with two windings displaced 90 electrical degrees from each other, and a squirrel-cage rotor (or equivalent). Each winding is excited by its own voltage source; these voltages differ in phase by 90 degrees. The first phase is called the fixed phase and the voltage is of constant amplitude and constant frequency. The second phase is called the control phase and the voltage is of the same frequency, but varies in amplitude.

The hinge axis drive provides for a cable wrap-up to carry power and signals to the shaft axis drive which, in turn, provides the same function to the yoke axis drive. Heaters are located in the wobble gear housing to maintain drive and cable wrap-up temperatures above 10°F during operation.

The position pickoff, used for control system feedback is a magnetic incremental encoder which sends digital feedback signals to the planetary scan platform drive electronics to null the error. Three encoders are incorporated into each drive mechanism to provide feedback signals to three redundant amplifiers in the electronics. The tachometer is an AC voltage generator. It provides the rate feedback signal to control the planetary scan platform drive speed. The tachometers are mounted in tandem with the motor.

The load on the planetary scan platform yoke axis drive is made up of the following components:

Inertial load	1.2 in.-oz. (negligible)
Bearing friction	100 in.-oz. (estimated)
Cable wrap-up	3000 in.-oz.
Total	3100 in.-oz.

The yoke axis loads are primarily due to the cable wrap-up stiffness, and are identical to the planetary scan platform shaft axis loads, and only slightly lower than high-gain antenna shaft axis loads. It is therefore proposed to use a common design for the yoke axis, the shaft axis and the high-gain antenna shaft axis, except that



it would be designed to drive the planetary scan platform as shown in Figure H-4. The power requirements are 12.5 watts, running and peak. The AC servomotor will be a size 11 connected to a gear reducer. The total gear reduction from the motor to the wobble gear output will be 10,000:1.

The weight of the (yoke axis) drive and encoders will be 6.0 pounds, maximum, and is broken down as follows:

Wobble gear drive	4.2 lb (including drive motor)
Encoders	1.8 lb
Total	6.0 lb

The outline dimensions are shown in Figure H-4.

The load on the shaft axis drive is made up of the following components:

Inertial load	1.2 in. -oz. (negligible)
Bearing friction	100 in. -oz. (estimated)
Cable wrap-up	3000 in. -oz
Total	3100 in. -oz

For a size 15 AC servomotor, a 6000:1 gearing ratio, and assuming a gearing efficiency of 50 percent, the motor will have to provide 1.0 in. -oz. of torque. This will result in a power requirement of 11.0 watts, running and peak. The control winding amplifier must supply 4.9 watts.

The load on the hinge axis drive for planetary scan platform deployment consists of the following components:

Inertial load	1.24 in. -oz. (negligible)
Bearing friction	100 in. -oz. (estimated)
Cable wrap-up	3000 in. -oz. (estimated)
Mass unbalance	492, 900 in. -oz (during Mars Orbit Insertion)
Total	496, 000 in. -oz

Using a gear ratio of 60, 000:1 and an estimated gearing efficiency of 50 percent, the required motor torque is 16.5 in. -oz. Using a

6-inch diameter AC servomotor, the resulting peak and running power requirements are 212 and 60 watts respectively during orbit insertion only. The amplifier used to drive the servomotor control winding must provide 110 watts peak power. The peak and running power in orbit about Mars is 15 watts.

The total weight of the two-axis shaft and hinge drive is 20.8 pounds, broken down as follows:

Shaft axis:	Wobble gear drive 4.2 lb (including drive motor)
	Encoders 1.8 lb
	Total 6.0 lb

Hinge axis:	Wobble gear drive 13.0 lb (including drive motor)
	Encoders 1.8 lb
	Total 14.8 lb

The size and shape of both drives are shown in Figure H-4.

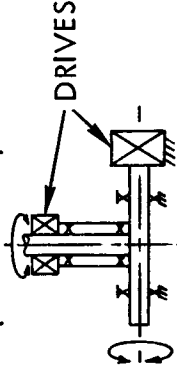
4.2 ALTERNATE APPROACHES

The major planetary scan platform drive elements which were considered in tradeoff studies were: 1) the gimbal system, 2) the drive mechanism, 3) the prime mover, and 4) the shaft position transducer. The tradeoff studies are summarized below.

Table H-7 shows the tradeoff between the standard cross gimbal and the differential gimbal. In this application the complexity and size of the differential gimbal make it less attractive than the cross gimbal which make a more rugged and compact design package. Therefore, the latter was selected as the recommended design.

Table H-8 shows the tradeoff for six drive mechanisms: 1) the wobble gear, 2) the harmonic drive, 3) the spur gear drive, 4) the linear ball screw drive, 5) the worm gear, and 6) the direct drive. The wobble gear drive was chosen because: 1) it has been successfully flight tested in the Orbiting Geophysical Observatory (OGO) spacecraft, 2) fast moving surfaces are sealed from the space environment, and 3) it makes a rugged, compact package.

Table H-7. Gimbal Tradeoff Summary

Gimbal Arrangements	Tradeoffs		Complexity
	Reliability	Performance Characteristics	
Standard Cross Gimbal (Preferred) 	Number of Critical Parts Bearings: 4 Gears: 0 Structural: 3	Movement about any one axis requires input from one drive only. Failure of one critical part causes failure of one axis only.	Minimum complexity. Simple to package, manufacture, and assemble. Minimum manufacturing cost.
	Bearings: 10 Gears: 4 Structural: 3	Movement about any one axis requires that both drives operate simultaneously. Failure of one critical part causes failure of both axes.	Highly complex. Difficult to package, manufacture, and assemble. High manufacturing cost.

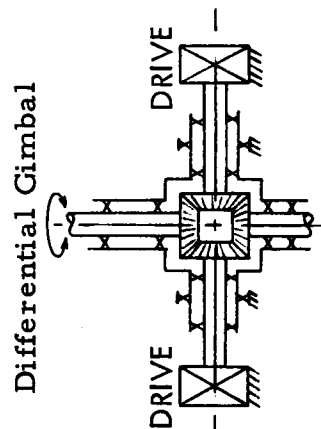


Table H-8. Sealed Drive Tradeoff Summary for Planetary Scan Platform Gimbal Drive Systems

Sealed Drive Candidates	Performance	No. of Gears Based on 100:1 Reduction	No. of Rotating Seals	Size and Weight	Packaging
Wobble Gear Calculated Reliability Single Axis: .9933 Two Axis: .986	High efficiency High torque transmission Good accuracy	Gears: 2	None Uses bellows and static "O" ring or hermetic seals	Small and light	Internal design moderately simple. Packages extremely well with gimbal.
Harmonic Drive	High efficiency Moderate torque transmission Good accuracy	Gears: 2	None Uses thin flexible metal tube which is exposed to space. High speed elements are sealed.	Small and light	Internal design moderately simple. Very difficult to package with gimbal. Requires additional gears and bearings
Spur Gear Drive	Moderate efficiency High torque transmission Good accuracy	Gears: 8 (Based on 4:1 reduction per gear mesh)	Two sets of seals	Small and light	Internal design moderately simple. More parts req'd than the two above. Can be designed to package well with gimbal
Linear Ball Screw Drive	Moderate to high efficiency, depending on no. of gears used. High load transmission. Good accuracy. Some losses of torque occur due to differential gas pressure acting on the bellows	Gears: 2 - 8 plus ball screw assembly	None Uses bellows and static "O" ring or hermetic seals	Large and heavy due to large stroke req'd to obtain gimbal travel	Internal design could range from moderately simple to complex. Will not package well with gimbal. Requires long stroke to achieve gimbal travel. Not suited to this application
Worm Gear	Low to moderate efficiency. High torque transmission Good accuracy	Gears: 2 (Gears have large sliding surfaces)	Two dynamic shaft seals	Small and light	Internal design moderately simple. Can be designed to package well with gimbal
Direct Drive	Highest efficiency High torque transmission Good accuracy	No gears	Two sets of dynamic seals	Small size and weight for drive; motor diameter prohibitive due to torque requirements	Internal design may be simple. Package can be designed for gimbal

Table H-9. Prime Mover Tradeoff Summary for PSP Gimbal Drives

	Performance	Life and Reliability	Size and Weight	RFI	Magnetic Field	Comments
DC Torque Motor	High torque to inertia ratio. High coupling stiffness. Fast response	May be short lived because of brush wear. Otherwise, high reliability	Moderate size and weight due to lack of gearing to load	High unless filtered	Large stationary magnetic field	Constant torque required out of motor to hold load
Stepper Motor	Constant speed. High resolution. Fast response	Not as long life as AC servo-motor	Moderate to large size due to slow rotor speed, inertial loads	No RFI generated by motor. Driver pulsed output can be troublesome	Uses permanent magnet rotor	This is a digital device and does not require D/A conversion in a computer system. Needs complex electronics for driver
AC Servo Motor	High speed, low torque device. Slow response due to high gearing ratio to load	Long life, rugged and reliable	Small size and light	No RFI generated by motor	No stationary magnetic field	High speed components need to be sealed and pressurized



Table H-9 summarizes the tradeoff study for the three prime movers: 1) DC torque motor, 2) stepper motor, and 3) AC servomotor. The AC servomotor was selected because of the absence of brushes, the low magnetic field, low weight and small size, and its ruggedness and higher reliability.

The magnetic incremental encoder was chosen over the other types of position transducers, such as potentiometers, optical pickoffs and capacitive devices because it offers high accuracy in a compact and reliable package.

5. PLANETARY SCAN PLATFORM CONTROL SYSTEM PERFORMANCE ANALYSIS

5.1 PLANETARY SCAN PLATFORM OPERATING MODES

A single axis planetary scan platform gimbal control system is shown in Figure H-14. Each platform gimbal control has the following three operational modes:

5.1.1 Mode 1 - Mars Track Mode

Error signals from the Mars tracker are used to maintain pointing along Mars local vertical. As the vehicle orbits, an apparent rotation of the local vertical is sensed by the Mars tracker. A typical response near perigee (where the rate of local vertical rotation is the largest) is illustrated in Figure H-15. The results indicated that the steady-state rate and position errors are zero degrees per second and 0.06 deg, respectively.

5.1.2 Mode 2 - Rate Command Mode (including IMC mode)

Rate commands are issued by the computer and sequencer or ground and stored in register 1 as shown in Figure H-14. Register 2 is reset and the position feedback loop is open in this mode. An electronic integrator is switched in to reduce steadystate velocity errors to zero. The command amplifier gain is reduced to 0.0095 v. sec/deg so that the resolution of the rate commands in register 1 can be 0.001 deg/sec per bit. The response to a 0.25 deg/sec rate command is shown in Figure H-16. The position error accrued is a function of the cable wrap-up torques. In this sample analysis, the position error is estimated about 0.065 deg for a 3100 in oz torque.

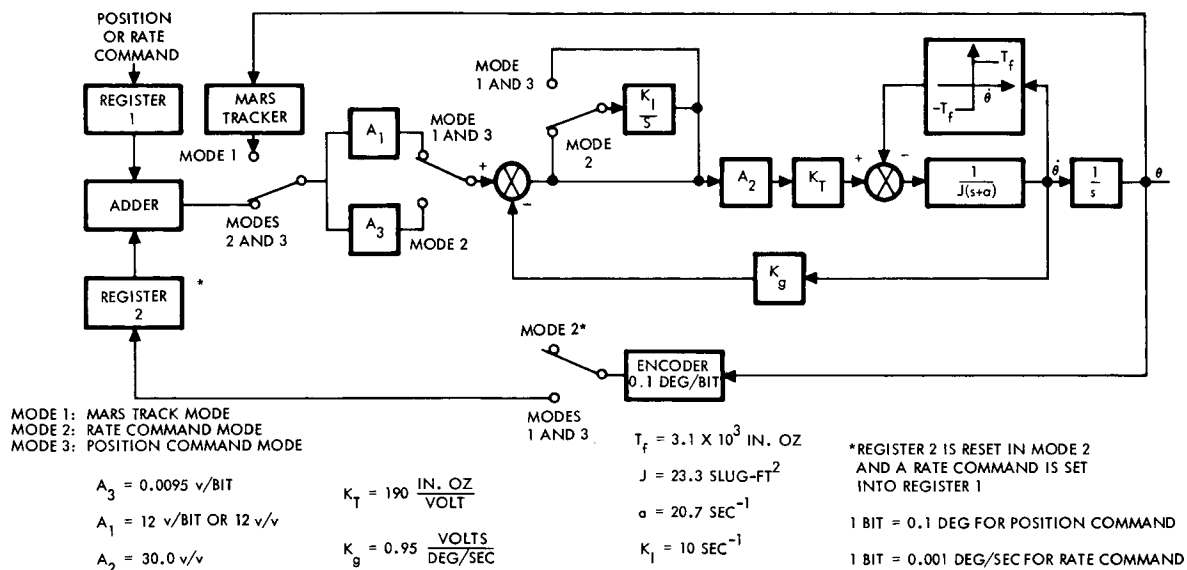


Figure H-14

CONTROL MODE ANALYSIS BLOCK DIAGRAM was used to determine parameter values for desired stability and control of PSP.

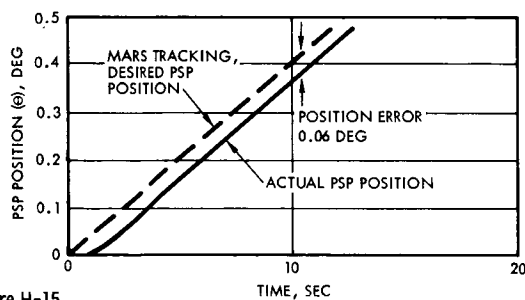


Figure H-15

PSP RESPONSE DURING THE MARS TRACKING MODE is shown by the time history of the PSP position.

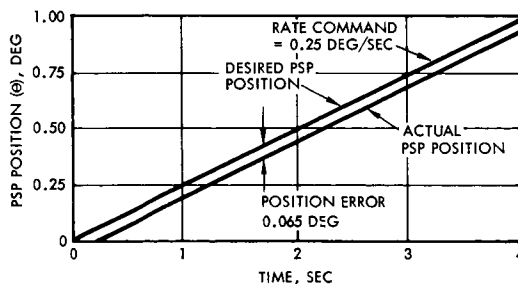


Figure H-16

PSP RESPONSE TO A RATE COMMAND is shown by the time history of the PSP position.

5.1.3 Mode 3 - Position Command Mode

The gimbal angles are commanded by the computer and sequencer or ground. The control system positions the gimbal to within 0.1 deg of the desired angle.

The selected command is stored in register 1 and added to the contents of feedback register 2 to provide the position error signal. The error signal is passed through a digital-to-analog converter, amplified and applied to the motor through the motor driver. A single error bit provides maximum rated torque; therefore, the motor operates in an "on-off" fashion. The gimbal slews at constant rate until the gimbal is within 0.1 deg of the desired gimbal angle; then the error signal is removed. The gimbal stops within 0.01° from the time of error removal. A typical response is illustrated in Fig. H-17.

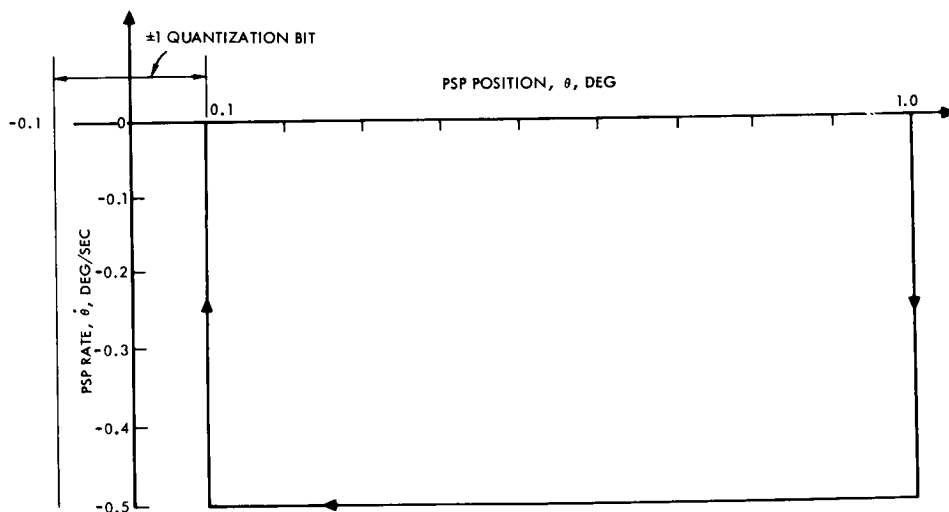


Figure H-17
PSP RESPONSE TO POSITION COMMAND is shown in a phase plane diagram.



5.2 PLANETARY SCAN PLATFORM POINTING ACCURACY

From the control standpoint, the planetary scan platform is controlled in two basic modes: the first mode is a closed loop mode which is controlled by error signals from the Mars tracker, the second mode is an open loop mode which is controlled by commands from the computer and sequencer subsystem or ground. The open loop control is used for functions such as image motion compensation for the photo-imaging system and for pointing to a specific location on the planet away from its center.

During the open loop mode operation of the planetary scan platform control system, the platform itself is caused to drive at a fixed angular rate in response to signals generated by the computer and sequencer. In this mode, the accuracy of pointing of the planetary scan platform is 0.486 deg (3σ) each axis with respect to the celestial reference frame. For the closed loop mode of operation, planetary scan platform pointing error is ± 0.292 deg (3σ) each axis including local vertical determination errors generated by the Mars sensor. The components of these pointing errors are indicated in the Table H-10.

Table H-10. Planetary Scan Platform Pointing Error Analysis

<u>Closed Loop Pointing</u>	
Mars horizon scanner accuracy	± 0.25 deg (3σ) each axis
Gimbal drives accuracy relative to horizon scanner	± 0.143 deg (3σ) each axis
Alignment of camera lenses, etc. to planetary scan platform mounting platform	± 0.05 deg (3σ) each axis
RSS	0.292 deg (3σ) each axis
<u>Open Loop Pointing</u>	
Celestial reference accuracies	± 0.10 deg (3σ) each axis
Gimbal drive pointing relative to commanded angle	± 0.143 deg (3σ) each axis
Alignment of drives and mounting platform to celestial references	± 0.15 deg (3σ) each axis
Alignment of lenses to mounting platform	± 0.05 deg (3σ) each axis
RSS	0.236 deg (3σ) each axis
Attitude control limit cycle error	± 0.25 deg (3σ) each axis
Total	0.486 deg (3σ) each axis

5.3 RELIABILITY ASSESSMENT

The reliability estimate for the planetary scan platform control equipment is 0.955. The reliability equation and block diagram is shown in component data presented in Volume D₂ Table Ell-1.

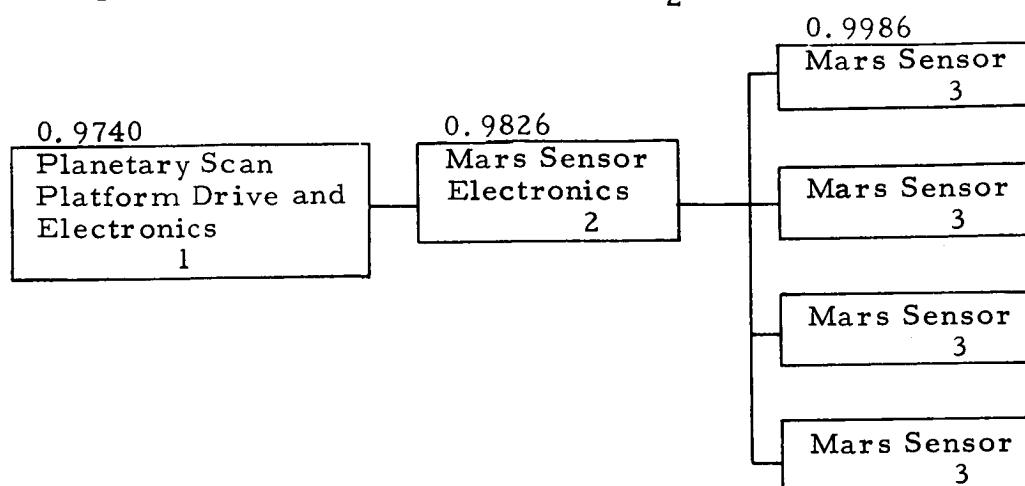


Figure H-18. Reliability Block Diagram of Planetary Scan Platform Equipment

$$R_{PSP} = R_1 R_2 \left[R_3^4 + 4R_3^3 (1-R_3) \right] = 0.955$$