X-692-75-91
PREPRINT

NASA TM X-70882

SMALL ASTRONOMY PAYLOADS FOR SPACELAB

(NASA-TM-X-70882) SMALL ASTRONOMY PAYLOADS FOR SPACELAB (NASA) 98 p HC $4.75 CSCL 03A
N75-23420
Unclas
G3/89 20425

EDITOR: RALPH C. BOHLIN

A WORKSHOP HELD 13-14 FEBRUARY 1975

Sponsored by the Astronomy Spacelab Payload Study

GSFC

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Foreword

In the 1980's Shuttle/Spacelab flights of 7 days duration or longer will offer astronomers the opportunity to utilize small, special purpose (sounding rocket class) experiments with integration times, spacecraft support facilities, and operational flexibility far exceeding that now availible in rocket astronomy. The Astronomy Spacelab Payloads (ASP) study at Goddard Space Flight Center is beginning to define feasible, concepts for the use of small payloads in Spacelab. We are particularly interested in establishing the requirements for hardware test facilities, interfaces, and program implementation procedures, which should provide the astronomical community with relatively simple and routine access to flight opportunities. A small, rocket-class payload is loosely defined as a payload with a minor impact on the complete spacelab system, a short lead time for development, and a relatively low cost. The weight limit will be about 400 kg and the size should be smaller than one Spacelab pallet element (3m length). Pointing and stabilization requirements are in the arc second range.

The First Workshop on Small Astronomy Payloads for Spacelab in the ultraviolet, optical and infrared disciplines was convened primarily for the purpose of acquainting the Goddard subsystem engineers with the payload requirements of scientists who have experience with flight hardware. Conversely, to inform the prospective users about the ASP study program, the agenda also included discussions of the Shuttle/Spacelab system, possible mission profiles, and the ongoing efforts at GSFC to define the necessary pointing and other subsystem capabilities. The list of invited scientists was restricted to the UV-optical-IR areas, because other groups are conducting similar studies in solar, atmospheric, and high-energy and X-ray astronomy. The proposed payloads will make comprehensive mission studies more realistic, as well as providing motivation for the design of support subsystems.

This report is not a transcript of the proceedings of the Workshop but is only a summary of the information presented. More detailed documentation on subsystems support will be available from the ASP study office after July 1975. Bruce Greer of Operations Research Inc. assisted in preparing this document.

INTRODUCTION TO SHUTTLE/SPACELAB

The Shuttle is a system comprised of an Orbiter, external fuel tank to power liquid fuel rocket engines, and solid booster rockets. Upon lift-off, all of the rockets fire in parallel, with the solid boosters dropping off soon after lift off. They are then retrieved, refurbished, and reused. The external tank continues to fuel the rockets in the Orbiter until just before orbit is obtained, at which time it is jettisoned. The Orbiter goes on to obtain orbit with fuel stored in its on-board tanks. This method of launching payloads into orbit is intended to be cost-effective, because the high-cost items, the vehicle itself with its many subsystems, and the rocket engines, are all used a number of times. The large external tank is discarded. The Orbiter has maneuvering capabilities when on orbit and can deliver and retrieve payloads. When its mission of up to 30 days is complete, it re-enters the earth's atmosphere, becomes a high-performance aircraft, and lands on a runway. On the ground the Orbiter is refurbished by removing the returned payload, transporting the craft to its launch site, checking its several subsystems, installing a new payload, attaching solid booster rockets and a new external tank, fueling, and re-launching.

The crew of Orbiter consists of the commander and the pilot. In addition to these two essential crew members are mission specialists and payload specialists as required by the particular mission. These members will receive the training necessary to meet the requirements of the particular mission. Accommodations for 28 man days of crew equipment and expendables are provided by the Orbiter. Thus, the requirements of a 4-man crew on a 7-day mission are met. Additional man days can be provided, but the provisions are payload chargeable.

With a gross mass of 950 metric tons, the Space Shuttle system is capable of lifting payloads of up to 29,500 Kg (65,000 lbs) and returning with a maximum of 14,500 Kg (32,000 lbs). The weights and dimensions of Shuttle and its components are shown in Table 1. The Orbiter provides many facilities for payloads. There are 13 structural points for attaching payloads to the Orbiter. A remote manipulator system, with a light and a TV camera mounted on the arm, allow payloads to be manipulated and inspected. Up to 50 kilowatt-hours of electric power are provided to the payload from the Orbiter. The avionics supply course pointing, communications, data transmission and reception, TV transmission, and onboard digital computations.

Within the Orbiter is Spacelab, the system providing support for experiments performed on-orbit. This system includes a number of elements and services necessary to the success of a payload. Briefly, Spacelab provides electrical interfaces and additional power. It provides data communications systems from experiment to the Payload Specialist Station, to the onboard computer, and to the Payload Operation Control Center. The Payload Specialist Station allows for a man-in-the-loop mode. Other systems and services of Spacelab include thermal control of equipment, pointing systems, mechanical mounting systems (pallets), and command and data management. The payload specialist's role includes setting-up, preparing and stowing all payload equipment. He may align, calibrate, and adjust instruments or point instruments at targets in the appropriate sequence. He will have responsibility in data management, determining whether data should be stored or used in real time. The specialist may also play a role in maintenance and repair, but the extent to which these functions will be performed are yet to be determined.

It is important to point out that physical systems are being constructed currently. The several parts of the Shuttle system are being designed and constructed by various contractors. Figure 1 shows the major parts of Space Shuttle and who is building them. Figure 2 illustrates the general design concept for Spacelab. The specific subsystems for support of UV-Optical-IR astronomy are discussed in detail later.

TABLE 1

SPACE SHUTTLE SYSTEM

APRIL 1974

Parameter	Metric value	English value
Overall Space Shuttle system		
Length	55.2 m	185 ft
Height	23.2 m	76 ft
Weight at launch	~1 860 000 kg	~4 100 000 lb
Payload weight into orbit		
Inclination (lowest), 28.5°	29 500 kg	65 000 lb
Inclination (highest), 104°	14 500 kg	32 000 lb
Solid-rocker booster		
Diameter	3.6 m	11.8 ft
Length	44.2 m	145.1 ft
Weight		
Launch	527 800 kg	1 163 500 lb
Inert	70 000 kg	154 300 lb
Thrust at launch, each	11 210 000 N	2 500 000 lb
External tank		
Diameter	8.4 m	27.5 ft
Length	46.9 m	153.9 ft
Weight		
Launch	739 800 kg	1 631 000 lb
Dry	31 900 kg	70 400 lb
Orbiter		
Length	37.5 m	123 ft
Wing span	23.8 m	78 ft
Height to extended landing gear	17.4 m	57 ft
Payload bay		
Diameter	4.8 m	15 ft
Length	18.3 m	60 ft
Cross range	2 038 km	1 100 n. mi.
Main engines (3)		
Vacuum thrust, each	2 090 700 N	470 000 lb
Orbital maneuvering subsystem engines (2)		
Vacuum thrust, each	26 700 N	6 000 lb
Reaction control system		
Engines (40)		
Thrust, each	4 003.4 N	900 lb
Vernier engines (6)		
Vacuum thrust, each	111.2 N	25 lb
Weight		
Dry	68 000 kg	150 000 lb
Landing	~82 000 kg	~180 000 lb

FIGURE 1

STATUS OF SPACE SHUTTLE CONTRACTING

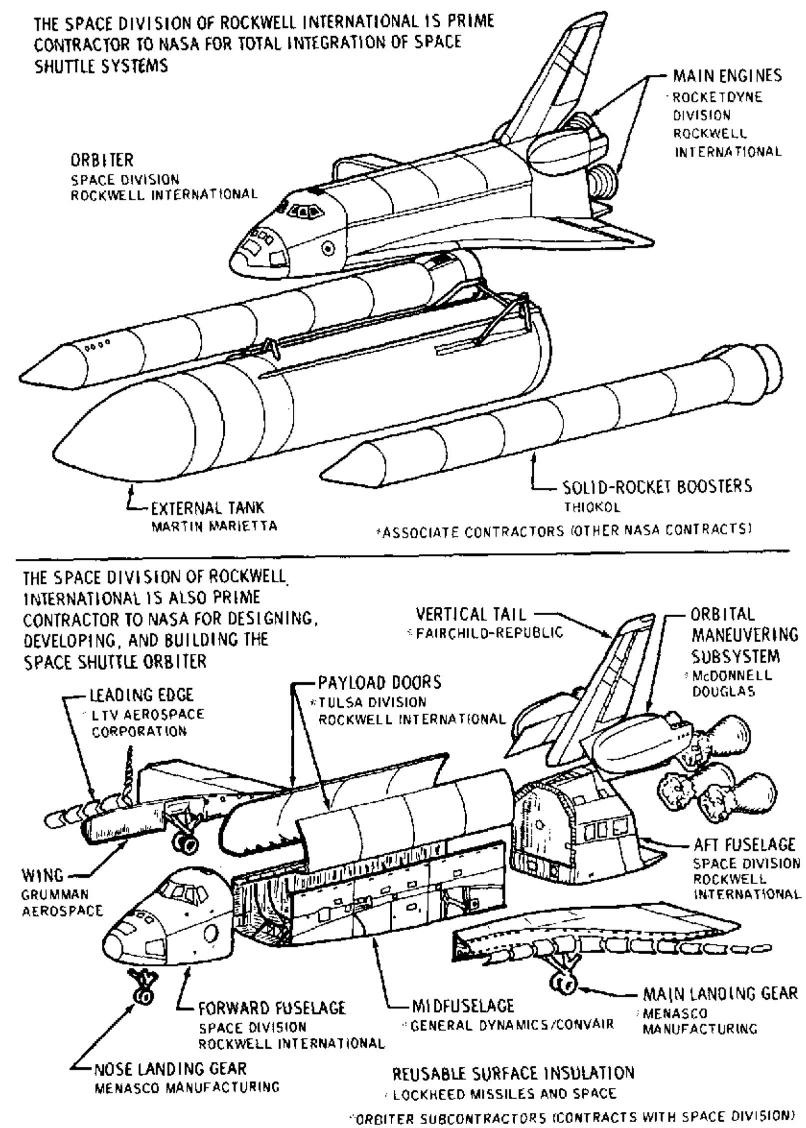
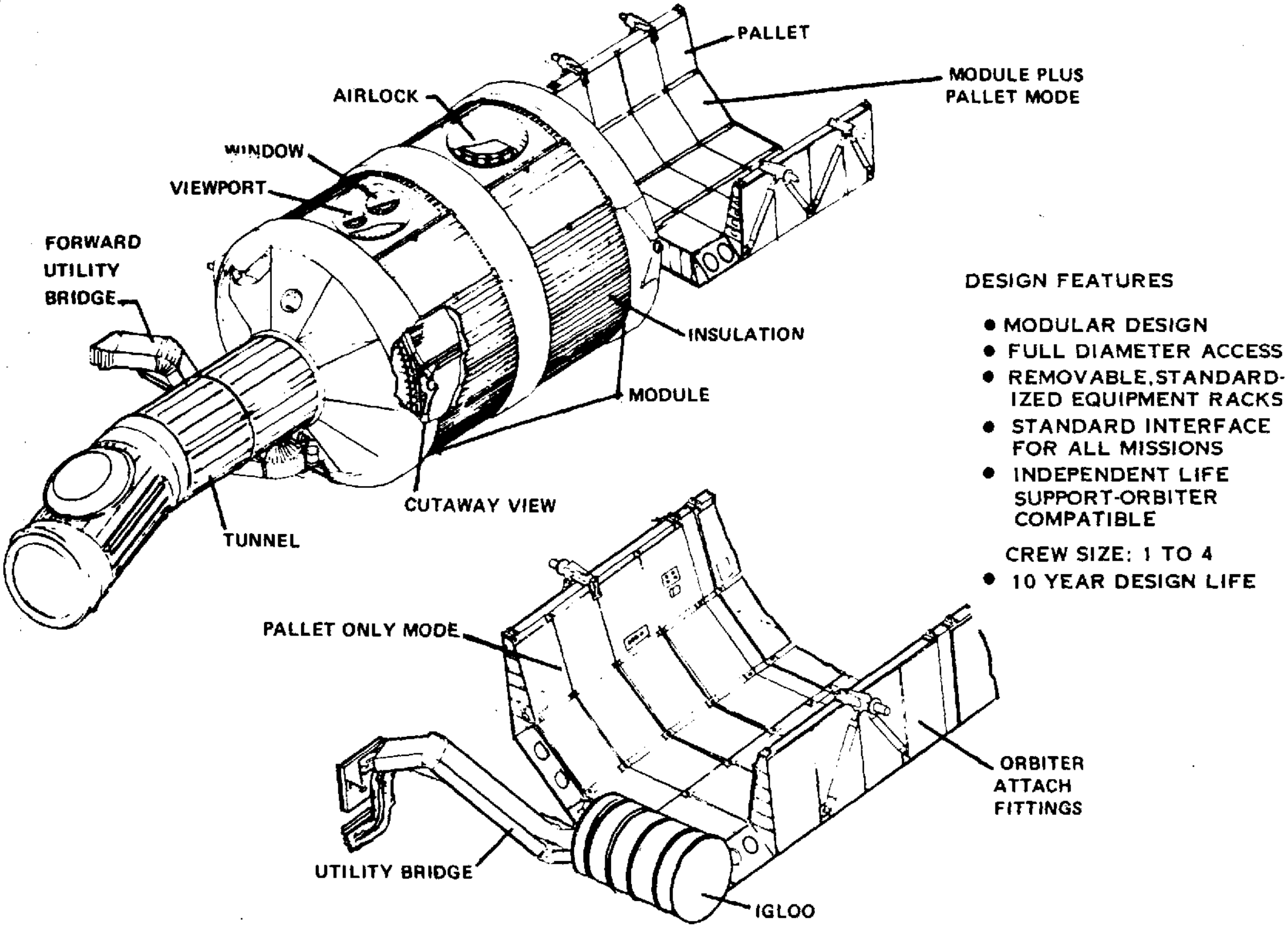


FIGURE 2
SPACELAB DESIGN CONCEPT



ASTRONOMY PAYLOADS

The eight science presentations provided the central focus of the Workshop. Without active participation by the experimenters who are currently flying astronomy payloads, the subsystems development could be incomplete and without long-term direction. The Workshop was well attended and enthusiasm for additional meetings on a yearly basis was expressed. The only invitees, not attending, were either in the hospital or in the field launching payloads.

The astronomers were asked to propose payloads for Spacelab in the spirit of the rocket program, where costs and paperwork are minimized, short lead time and rapid turnaround are emphasized, and some degree of risk is accepted for each individual launch. The presentations by the astronomers are their interpretation of these guidelines. The following two tables summarize some of the important parameters for the payloads discussed by the eight different groups.

SMALL UV-OPTICAL PAYLOADS - SUMMARY OF REQUIREMENTS

	Dimensions (cm)	Mass (kg)	Power (W)	Temp (°C)	Spectral Range (Å)	Resol-ution (Å)	Limit Mag.	Pointing Abs. (min)	Pointing Stab. (sec)	Data Rate (Kbps)	Field Diam. (deg)	Non-standard Requirement
Princeton												
High Resol.	15 m bay	<50	small	TBD	912-1100	0.003	∿5	TBD	180	750	TBD	Dedicated Pointing 15 m light path
NRL												
Schmidt 1	30x55x115	80	25	20$\pm$15	1250-2000	Con	18	60	10	Film	11	RCS and gas Venting inhibited.
Schmidt 2	30x55x115	80	25	20$\pm$15	1050-1600	Con	18	60	10	Film	11	Has pointing mount.
JSC												
UV-Tel.	100x100x250	400	500	Con	2000-3400	0.1	8	Con	1-Con	48	TBD	
Sky Survey	120x200x220	700	1000	10$\pm$10	UV-Vis	TBD	24	6	0.1-Con	Film	5	Large mass and size. 2.5 sec roll stability.
Echelle	100x100x200	300	TBD	TBD	TBD	0.05	TBD	0.02	0.3	Film	TBD	Pointing
AFCRL												
IR-Tel.	51Dx137	170	150	7$\pm$30	4-30x10^4	Con	---	1	20	28	TBD	Scan mode LHe on gimbal
Berkeley												
EUV-Image.	44Dx250	150	12	24$\pm$19	100-1000	Con	TBD	60	3600	20	TBD	No SIPS needed
EUV-Spect.	18x27x65	16	10	7$\pm$47	250-1200	TBD	TBD	120	7200	40	TBD	Scan mode
X-Ray	43Dx180	160	40	12$\pm$30	3-100	TBD	---	60	3600	200	1-2	Gas flow detector
Wisconsin												
Photom.	38Dx200	91	100	30$\pm$70	925-3400	50	TBD	2	5	1	0.5	New Moon
GSFC												
Schwartz.	38Dx190	162	100	20$\pm$15	1200-3000	2	---	30	2	Film	11	Side looker 15 sec stab-3rd axis
Colorado												
Microch.	16x27x107	17	30	20$\pm$40	450-3100	2.5	13	1	30	400	0.1	
Polarim.	19Dx34	3	2	20$\pm$40	1050-7000	200	19	1	60	0.1	Con	
High Resol.	38Dx200	TBD	10	20$\pm$20	1050-3100	0.05	11	0.02	1	Film	.001	Absolute Pointing

Notes: D- Diameter
TBD - To be determined
Con - Controlled by experimenter.

COST AND DELIVERY

	Cost Guestimate (thousands of 1975 dollars)	Lead Time (years)
Princeton		
High Resolution Spectrometer	300	2
NRL		
2 Schmidt Cameras	100	1.5-2
JSC		
UV Telescope (modify)	490	3
UV Telescope (copy)	1030	3
Sky Survey	2000	3
Echelle	500-1000	3-4
AFCRL		
IR-Telescope	1900	2.5
Berkeley		
EUV Imaging Telescope	40	0.25
EUV Spectrometer	125	3
X-Ray	45	0.25
Wisconsin		
UV Photometer	300	1
GSFC		
Schwarzschild Camera	200	1.5
Colorado		
Microchannel Spectrometer	150	0.7
UV Polarimeter	200	1.5
High Resolution Spectrograph	200 + telescope	1.5

The cost figures are only educated guesses and, in many cases, are not broken down or itemized in any way. The relative costs are also unreliable, because such things as travel, manpower support, and number of flights are not treated uniformly. The lead time is defined as the time between funding and the beginning of test and evaluation at Goddard.

1. A VERY HIGH RESOLUTION UV SPECTROGRAPH FOR INTERSTELLAR MATTER RESEARCH

E. Jenkins and D. York, Princeton University Observatory

Objectives

In 1959 prior to the development of space astronomy, Spitzer and Zabriskie predicted that the study of absorption features appearing in the far ultraviolet spectra of hot stars would afford us a very powerful means to analyze the composition and physical state of the interstellar gas. The foundations of that prediction even understated the enormous wealth of material and the growth in our understanding which has been precipitated by the observations from the Copernicus (OAO-3) satellite. We may anticipate that the International Ultraviolet Explorer (IUE) and Large Space Telescope (LST) should significantly widen the scope of ultraviolet observations by collecting spectral information at a much faster rate and with greater sensitivity. These two instruments, plus the proposed Spacelab 1-meter telescope facility, should be able to execute a comprehensive ultraviolet observing program leading to data not only on interstellar matter, but also on the actual targets observed - stars, galaxies, solar system objects, etc.

Worthwhile objectives for more specialized, new instruments for Spacelab include classes of observations which are outside the grasp of the relatively powerful, general purpose instruments just mentioned. One such program is the recording at substantially higher wavelength resolution the spectra of relatively bright stars. An increase by a factor of ten in resolving power to $\lambda/\Delta\lambda = 3 \times 10^5$, which corresponds to 1 km s^{-1} in radial velocity, permits us to address the following crucial problems in interstellar matter research.

a) Kinetic temperature of the diffuse gas intercloud medium

While there are several approaches to learning about the temperature within dense accumulations of gas, such as observing 21-cm emission and absorption, H_2 rotation temperatures, and C I fine structure populations, temperature measurements for the more tenuous un-ionized material have been elusive. Comparisons of emission and absorption by broad velocity components at 21-cm seem to indicate temperatures ranging from 600 to 9000°K, but the interpretation of the results is somewhat controversial. High resolution measurements of widths for weak absorptions in the ultraviolet should show the thermal motion of the atoms. A dispersion in radial velocity produced by either turbulence or gradients in bulk velocities can be separated from thermal broadening by observing constituents of different mass. For instance, a temperature as low as 100°K will produce unsaturated absorptions by atomic hydrogen whose apparent widths are at least 44% wider than those from elements heavier than carbon, if the non-thermal broadening is no larger than 1 km s^{-1} and the half-width of the instrumental profile is 1 km s^{-1}. The measured temperatures of H I at relatively low densities bear directly on our theoretical understanding of the heat balance in the gas, as well as on the nature of thermal instabilities and phase separations.

b) Velocity separation of absorption components from H I and H II regions

Many atomic and ionic species arise from both H I and H II regions, and at low resolution the components are blended. The ability to separately consider contributions from the different regions has obvious advantages in the interpretations of abundances and physical conditions. For example, absorptions from H I regions caused by ions requiring more than 13.6 eV ionization energy for their production could be isolated. Experience with Copernicus data suggest H I and H II region velocities can have typical separations of 6 to 10 km s^{-1} for nearby stars - a velocity difference barely resolved by Copernicus but easily separated at the proposed 3 x 10^5 resolving power.

c) Separation of velocity components of H_2 with high and low rotation temperatures

Early studies of H_2 absorptions suggested an increase in velocity dispersion for absorptions by H_2 in realtively high levels of rotational excitation ($J \geq 4$ or 5). More precise observations by Spitzer and Morton revealed that the apparent increase was due to a superposition of components at different velocities, rather than a symmetrical increase in the velocity spread of a single component. The ability to unravel these contributions would clarify our understanding of the rates of formation, destruction, and rotational excitation of interstellar H_2 under substantially diverse conditions.

d) Gas in the solar vicinity

Most O and B stars are on the order of 100 or more pc away, however a number of bright M and K giants are much closer. In the ultraviolet these stars exhibit strong chromospheric emission lines which may show narrow interstellar features in absorption. For instance, observations of the Lα absorption to nearby stars by Copernicus has revealed that the local neutral hydrogen density is only around 0.05 atoms cm^{-3}, considerably lower than average for our galaxy. Our confidence in the accuracy of this technique for measuring hydrogen can be significantly enhanced by going to higher resolution, since our present inability to see the precise shape of the emission is a principal source of uncertainty. Even more gain may be realized for elements other than hydrogen, where the emission lines and matching interstellar absorptions are much narrower.

The specific research possibilities listed above are in themselves strong justification for observations at high resolution. In addition, the principal uncertainties in column densities derived from moderately (but not fully) saturated lines can be virtually eliminated by directly integrating optical depths over velocity instead of applying curve of growth techniques. In short, the value of high resolution profiles becomes obvious by reviewing the detail exhibited by lines in the visible spectrum recorded by Hobbs at a resolution of ~ 1 km s^{-1}.

Instrumentation

Simultaneous detection of the many adjacent wavelength bins is almost imperative, especially at high resolution, since the observing time on a shuttle mission is limited. This introduces an imaging detector as a necessary component of the system. Photoelectric devices capable of imagery have limited spatial resolution, however, which imposes the severest constraint on instrument design concepts when one requires high wavelength resolution.

A grating spectrograph with a focal length ℓ will have a resolving power given by $\lambda/\Delta\lambda = \ell$ (sec r) (sin i + sin r)/Δx where i and r are the angles of incidence and reflectance, respectively, and Δx is the width of a resolution element of the detector. Although high resolution can be achieved by having r approach 90°, blaze efficiency or effective beam collecting area of the grating is sacrificed. Another approach, which is the choice we adopt here, is to increase ℓ to a very large value. One can magnify or fold the dispersed beam to limit the physical dimensions of the configuration, but this is undesirable since the attenuation of the light flux is large, owing to the poor efficiency of optical elements in the far ultraviolet. On the other hand, we can capitalize on the generous length of the shuttle payload bay and have an uninterrupted beam from the grating at one end focused on the detector at the opposite end. If ℓ is as large as 15 m, the length of the Spacelab pallet assembly, and Δx is 50μ (a realistic value), we can achieve $\lambda/\Delta\lambda = 3 \times 10^5$ if the combined trigonometric terms in the equation are about unity (which gives reasonable angles). An additional benefit of a long focal length is the reduction of high order optical aberrations. For efficiency and simplicity, a concave grating used in a Wadsworth configuration seems most desirable.

A conventional approach for recording a spectrum is to allow the imaging device to accumulate and store the photon counts over the time of integration. While this has obvious advantages for economical data management, it requires elaborate and very precise compensation precedures over the whole integration time to eliminate drifts in wavelength caused by (1) guidance errors, (2) flexure of the shuttle or instrument and (3) variations in projected orbital velocity. To avoid these complications, we prefer to allow the spectrum to move and use very short integration times. The detectors will be operated in a photon counting mode, and the position of each photoevent will be recorded. Position offsets will be recorded using an image disector which senses the star's flux from a mirror rigidly attached (but with a small tilt) to the grating cell. The subsequent analysis of the data to produce a spectrum will compensate for the different forms of drift. The major shortcoming of this method, of course, is the wide bandpass of about one mega Hz needed to record the rapid flow of photoevent coordinates.

Pointing and Other Spacelab Requirements

The entire shuttle vehicle must be oriented properly for each target star. No drift greater than about 0.1° is acceptable during periods of a half to one hour. The availability of control moment gyros may be essential for this experiment, because contaminants with column densities as low as 10^9 cm^{-2} are detectable and begin to interfere with observations of interstellar lines. The worst contaminant is H_2 and other bad species include OH, CO, H_2O, O_2 and N_2. Another solution would be to gate the experiment off during gas firings, if the column densities are large only for a small fraction of the time. The need for dedicated shuttle pointing will impact mission operations and sacrifices in observing efficiency for this or other experiments may result from conflicts. A limited capability for independent pointing could be acquired by allowing the grating to articulate along two axes, but the increased complexity and changing instrument characteristics would make this choice somewhat undesirable.

In several respects the proposed payload is essentially of the "sounding rocket class" in that it is lightweight, inexpensive, conceptually simple and is designed to accomplish a special class of observations outside the capability of a general telescope facility. In one other respect, however, it differs from normal small instruments: it is far from being compact, because the light beam traverses almost the entire length of the shuttle bay. Somewhere within the bay we must have an unobstructed line of sight for the light beam traveling between the grating and the detector. How difficult a problem this will be is unclear until it is known what the dimensions of other systems sharing the flight will be. In all likelihood some unobstructed path will exist, or alternatively, compromises could be made. (For instance, for a few of the pointing directions of SIPS, conflict may occur and the high resolution observations would occasionally be interrupted). If serious interference with other payloads seems inevitable, then it may be preferable to operate this system on a mission carrying free-flyer satellites for which an empty payload bay would be available after release.

2. SCHMIDT CAMERA/SPECTROGRAPH FOR FAR-ULTRAVIOLET SKY SURVEY

G. Carruthers and C. Opal, NRL

Objectives

The primary objectives of the proposed experiment are to obtain far-ultraviolet imagery and intermediate-resolution spectra, in the 1050-2000 Å wavelength range, of stars and stellar objects (early type Pop. I stars, and Pop. II objects such as the faint blue stars at high galactic latitudes), emission and reflection nebulosities, planetary nebulae, relatively nearby external galaxies, and the brighter Seyfert galaxies and quasars. The stellar spectra will also provide information on the distributions of interstellar dust, atomic hydrogen, molecular hydrogen, and (for the more distant and/or reddened stars) atomic oxygen, nitrogen, and carbon.

It is desired to cover as much of the sky as possible, to the faintest possible limiting magnitudes with high photometric quality. The ultimate goal is a complete sky survey, reaching (in 20-minute exposures) unreddened BO stars (or equivalent) as faint as $m_V = 18$ in direct imagery, and as faint as $m_V = 11$ in the objective-spectrograph mode (2Å resolution) or $m_V = 9.5$ (0.5Å resolution). The limiting magnitude for direct imagery is 8 magnitudes fainter than reached by the Celescope experiment on OAO-2. Thus, the proposed experiment will serve to lay the ground work for observations with larger instruments such as the Large Space Telescope and the 1-meter Spacelab Optical/UV telescope.

Instrumentation

The proposed Schmidt camera/spectrograph unit is shown in Fig. 1. It is a 15-cm aperture, f/2 system using electrographic recording and is similar to, but somewhat larger than, devices flown on NRL sounding rockets, on Apollo 16, and Skylab 4. The 15-cm camera is also very similar to a 10-cm aperture, f/1.5 camera/spectrograph unit constructed in 1967 for the Marshall-developed ST-100 platform, intended for a possible second Skylab. An important advantage of electrographic recording is the high quantum efficiency and long-

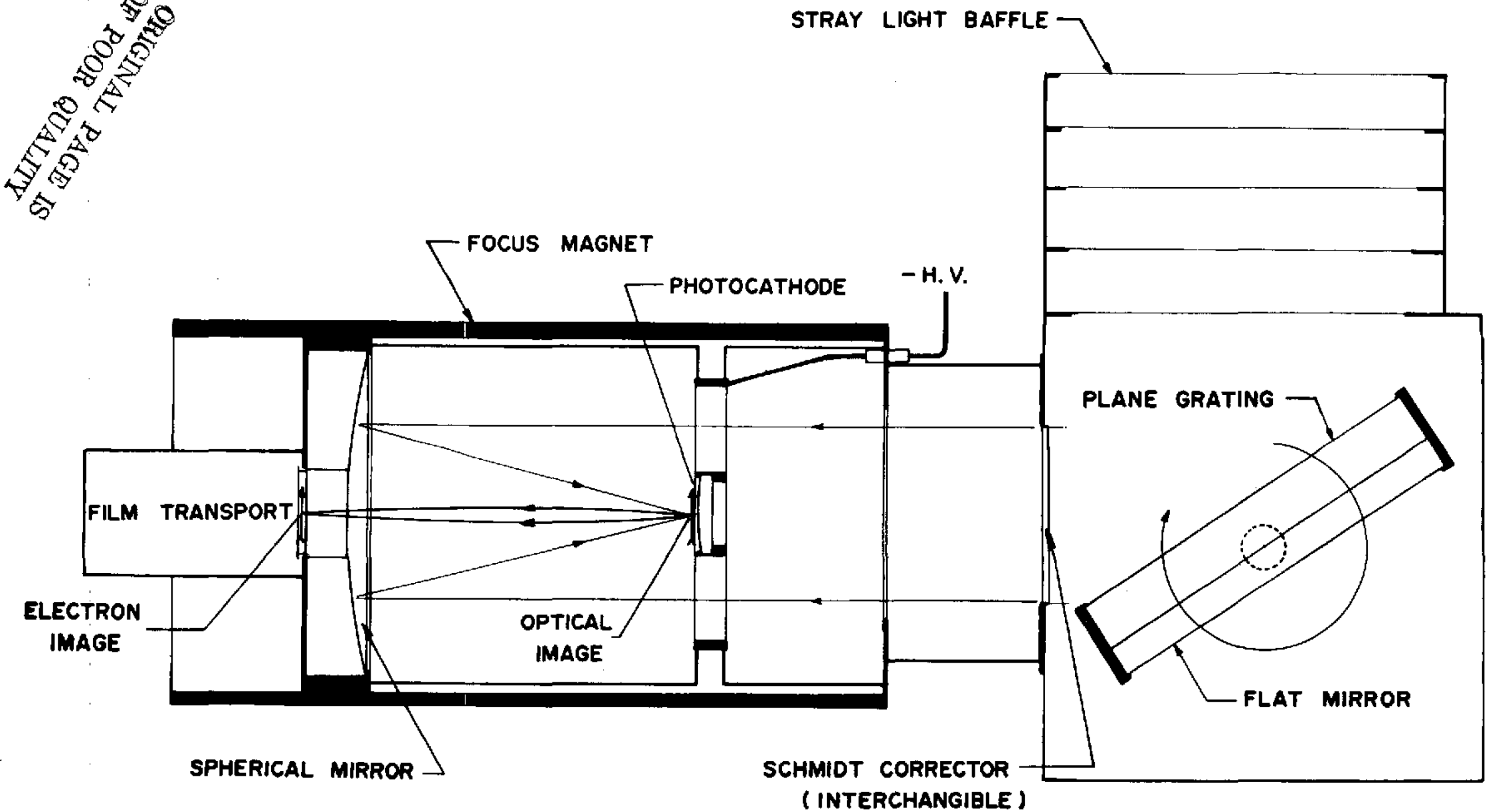


Figure 1. Diagram of the Schmidt camera/spectrograph.

wavelength rejection achieved by the use of front-surface (opaque) alkali-halide photocathodes. Thus, despite the relatively small aperture, the electrographic Schmidt camera has a high overall detection efficiency, plus linearity of response and wide dynamic range.

It is proposed to fly two camera/spectrograph units covering, respectively, the wavelength ranges 1050-1600 Å and 1250-2000 Å (see Table).

	Camera 1	Camera 2
Photocathode	CsI	KBr
Correctors	CaF_2, BaF_2, Al_2O_3	LiF, CaF_2
Imagery	1250-2000 Å	1050-1600 Å
	1350-2000 Å	1250-1600 Å
	1450-2000 Å	
Spectra	1250-2000 Å	1050-1600 Å

Fields of View	11° Circular
Resolution (spatial)	20 arc sec
(spectral)	2 Å (300 lines/mm)
	0.5 Å (1200 lines/mm)

These units would operate simultaneously, while viewing the same region of the sky. Each unit would have an 11° diameter field of view, 20 arc sec resolution (0.5 to 2 Å spectral resolution, depending on choice of grating), and would record images on 70 mm electron-sensitive film (a 150-ft. roll in each unit would last a 7-day mission). For sky mapping, the effective field is a 7° square, and 842 different pointings are needed for complete sky coverage. Therefore, with 30-min. exposure sequences and all modes of operation for each starfield, complete coverage would require the night portion of 4200 orbits. Since the instrument is currently under construction, there would be no difficulty in being ready to fly on the early Shuttle flights. However, the proposed instrument has significantly greater capability than similar ones presently in use in sounding rocket flights and would not be obsolete by 1979.

Pointing and Other Spacelab Requirements

The pointing accuracy required is ± 1° (desired ± 0.5°), which is within the capabilities of the basic Shuttle RCS. However, the pointing stability required is ± 10 arc sec over a 20-minute exposure time, which requires an additional fine-pointing system. Since the fine-pointing requirement is not so severe as for several other proposed instruments, and the sky-survey type of observing program is generally incompatible with the use of startrackers for fine pointing, we propose a special-purpose platform using rate-integrating gyros for fine pointing (see Fig. 2).

Coarse pointing is achieved using the shuttle RCS, with the guidance of the shuttle IMU and the closed-circuit TV starfield camera. During these maneuvers, the platform gimbals and RIG's are caged, with the instruments pointing vertically out of the payload bay. Then,the platform gimbals and RIG's are uncaged, so as to hold the pointing to the required high stability. The platform gimbals and RIG's are then recaged before moving to the next target. Alternately, if successive pointings are close together in direction (as for sky mapping), the coarse slew can be done with torque motors on the platform gimbals, with the RIG's only being caged for the slew.

The ambient gas pressure in the payload bay must not exceed 10^{-5} torr during operations. Thus, RCS jets and overboard venting must be inhibited during exposures. The payload must be kept in dry nitrogen at all times after shipment from NRL. The experiment should be sealed and kept dry during re-entry. Tentative control and monitoring functions evisaged for the Payload Specialist control panel are the following:

High Voltage On/Off

Exposure Sequence Initiate (predetermined automatic sequence)

Manual Film Advance

Manual Selects: Corrector Plate, Mirror/Grating

Film Advance Monitor (flashing light)

High Voltage Monitor (meter)

Closed Circuit TV (pointing monitor)

Gimbal Cage/Uncage

Platform Cage (for launch and reentry)

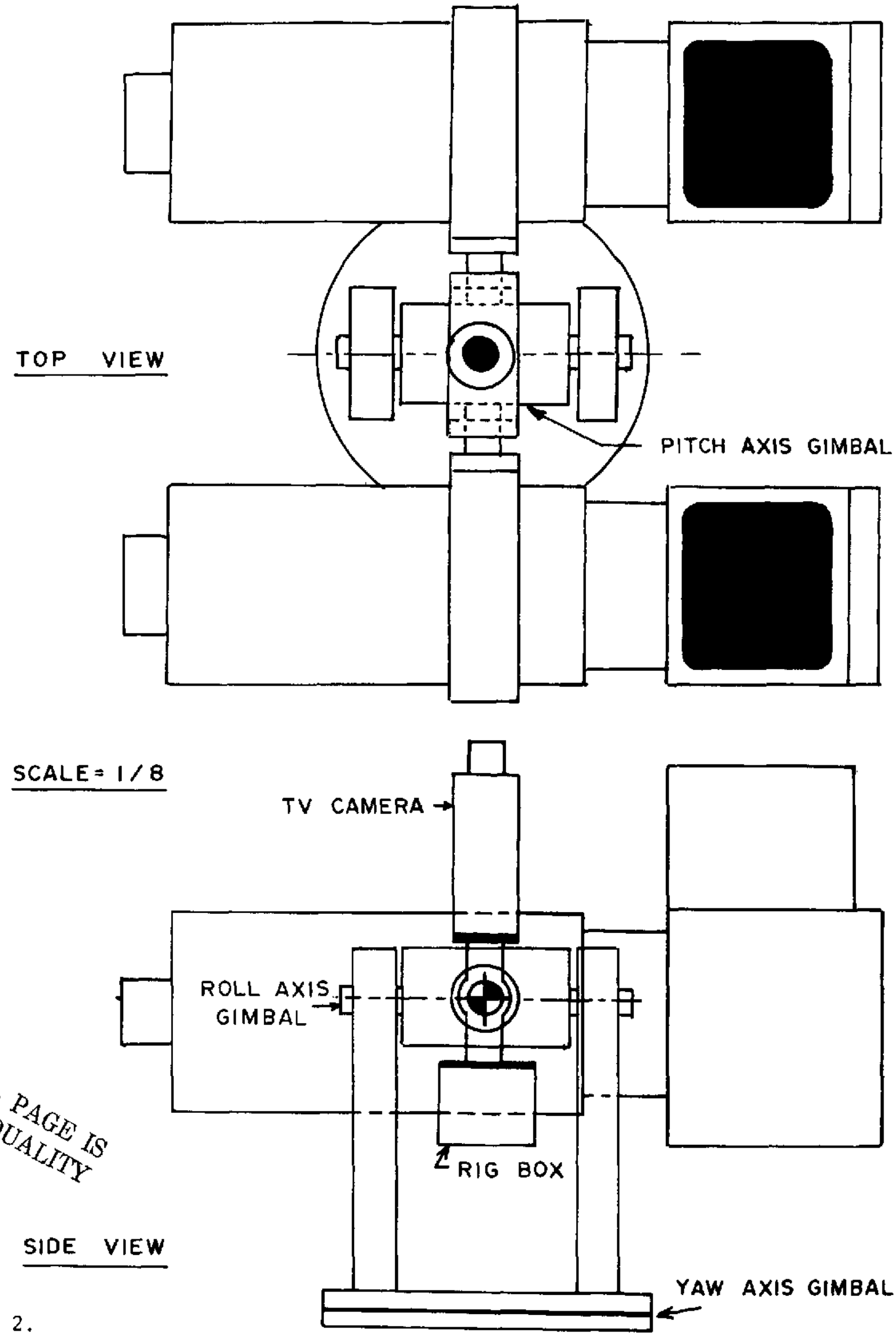


Figure 2.

Conceptual view of an instrument package consisting of two Schmidt camera/spectrograph units mounted on a fine stabilized pointing platform. This package is mounted in the Shuttle payload bay and is controlled from the shuttle cabin or pressurized Spacelab cabin.

3. UV TELESCOPE WITH ECHELLE SPECTROMETER

Y. Kondo and C. Wells, JSC

Objectives

The primary scientific objectives of this experiment are investigations of stellar chromospheres, dynamics of extended atmospheres of supergiants and WR stars, mass transfer in close binaries including x-ray binaries, chemical abundance in stellar atmospheres, and chemical abundance and electron temperature of the interstellar medium. We are currently conducting a multi-year program of spectrophotometry of astronomical objects in the mid-ultraviolet through use of JSC's balloon-borne Ultraviolet Stellar Spectrometer (BUSS). This project of payload development includes the flight-tested JSC BUSS payload and the JSC/SRL BUSS payload (SRL stands for Space Research Laboratory at Utrecht, The Netherlands). The JSC/SRL BUSS payload with adaptations constitutes the JSC/SRL Telescope Spectrometer for Spacelab and is scheduled for a balloon flight in October 1975.

Instrumentation

The proposed system consists of the BUSS telescope and star tracking system, supplemented with a high-resolution echelle spectrograph and SEC vidicon detector supplied by SRL. The instrument is shown schematically in figure 1. Total weight is less than 400 kg, including star trackers and a gimbaled mounting platform. The telescope is an f/7.5 tilted-aplanatic design, which has been used successfully in previous BUSS mission The telescope focal length is 3 meter, its aperture 40 cm. The star tracker shown in the figure allows coarse pointing of the entire telescope to one arc minute towards the target star, while a further refinement of the pointing is accomplished by an image motion compensation system with one arc sec stability even if the shuttle attitude changes at 1°/sec.

The spectrograph is of the echelle type, allowing the entire spectral region of 2000 - 3400 Å to be observed simultaneously by means of the SEC vidicon detector. This is the fundamental

JSC SRL-BUSS SYSTEM CUTAWAY

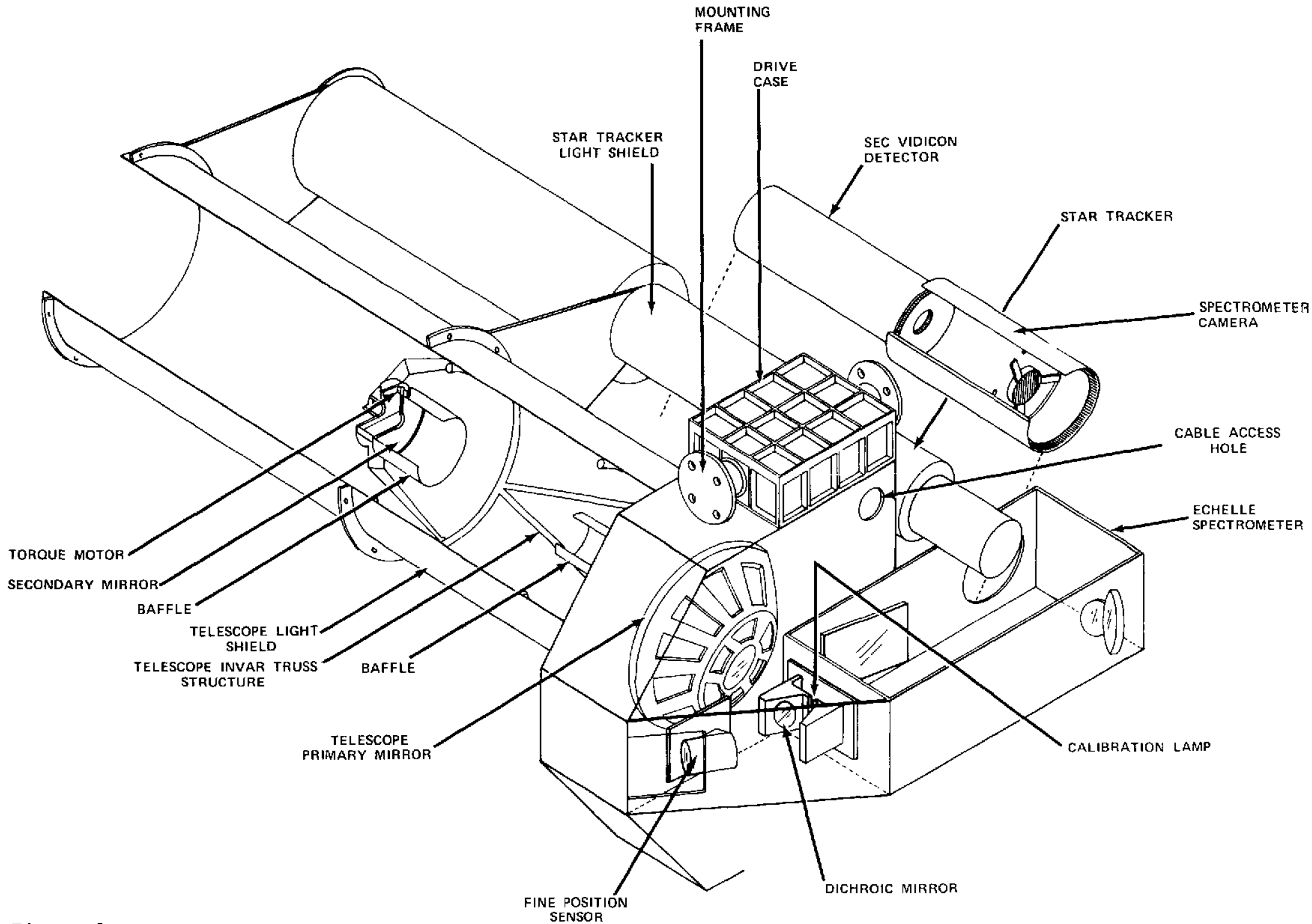


Figure 1.

difference in this instrument as compared with, for instance, S59, BUSS, or OAO-3, where the spectrum is scanned step-by-step. The UV light from the telescope is reflected by means of a dichroic multilayer mirror into the spectrograph, while the transmitted visible light of the star image is used for the image position sensor. The main dispersing element of the spectrograph is an echelle, with a blaze angle of 63°.5 and a groove density of 79 lines/mm. The ruled area of the echelle is 102 x 206 mm, which is illuminated by means of a 500 mm focal length collimator. This design allows a spectral resolution at 2800 Å of better than 0.1 Å even if the convolution of the telescope blur circle and fine pointing errors of the telescope amounts to 3 arc seconds FWHM. The limiting magnitude is about $V=8^m$. The spectral range of 2000 - 3400 Å is displayed in the spectrogram from the 112th order at 2000 Å up to the 66th order at 3400 Å. Reciprocal dispersions range from 1.21 Å/mm at 2000 Å up to 2.05 Å/mm at 3400 Å. The orders are separated spatially from each other by means of a quartz predispersing wedge in such a way that the whole spectrogram is fitted optimally to the 25 x 25 mm target of the SEC vidicon tube. The spectrograph will be equipped with a wavelength reference source in order to allow in-flight wavelength calibrations. The photometric response of the instrument will be determined by means of pre-flight and post-flight calibrations in the laboratory. Later improvements of the instrument include upgrading the spectral resolution to 0.03 Å, extension of the wavelength coverage to the 1150 to 3400 Å, range and using the echelle spectrometer with a one meter telescope.

Pointing and Other Spacelab Requirements

The pointing requirements are compatable with the requirements for SIPS, but the complete pointing system of the BUSS makes it an attractive candidate particularly for early shuttle flights, when SIPS may not be fully operational.

The scientific data of the instrument will be stored on board on magnetic tape. Housekeeping data analysis should be done preferably on board, but could also take place on ground. Both houskeeping and scientific data can be handled by the existing computer facilities in Spacelab, or by a separate minicomputer with 16 K of 16 bit words. Every orbit an average of ten television frames of 8 Mbit each plus 1 Mbit of housekeeping data have to be stored on magnetic tape. Housekeeping data will, together with quick-look scientific data, be transmitted to the ground in parallel at a bit rate of 48 Kilobits/sec in lieu of a specialist. Tasks of the payload specialist would be:

a. To start automatic star acquisition software program (once per orbit).
b. To start the measurement sequences software (once per orbit).
c. To ensure proper data storage (changing tapes, etc. regularly).
d. To take action in case of anomalies.

As a back up all commands can be generated also from the ground.

Additional Payloads

Two other instruments from JSC were discussed at the workshop that exceed the guidelines for small payloads in weight, size, or pointing requirements. The first was a 30-inch Schmidt telescope with a package size of 1.2 x 2.2 x 2.0 m and a mass of 700 kg. The absolute pointing accuracy needed was only 6 arc min and internal stability is provided to 0.1 arc sec but a roll stability of 2.5 arc sec is required. The second payload was an Echelle Nebular Spectrograph with 1 x 1 x 2 m exterior dimensions and a 300 kg mass. Pointing accurate to 1 arc sec is needed with 0.3 arc sec stability.

4. SMALL INFRARED CRYOGENIC TELESCOPE

R. Walker, AFCRL

Objectives

The objective of this work is to obtain observational data characterizing the spectral energy distribution of celestial objects in the intermediate infrared, 4 to 30 microns. Specifically two classes of observations would be performed.

A. Measurements of diffuse sources of large angular extent:

a) Thermal emission from interplanetary particles (zodiacal emission) - A low resolution spectral and spatial map of zodiacal emission would permit identification of compositional classes (silicate, iron, etc.) of the emitting particles and compositional variations with distance from the sun.

b) Cosmic background radiation due to the aggregate of unresolved galaxies - Definition of the spectrum of the cosmic background in the middle infrared will provide much selectivity in choices between steady state and evolutionary models of the universe, and provide needed data on the mean density of matter in the universe.

c) Survey of galactic plane for extended regions of non-thermal emission - A great variety of atomic and molecular emission lines have been predicted for regions where dust and gas are interacting, for example:

H_2 at 4.4, 5.0, 6.1, 8.0, 12 and 28 microns; Ne at 12.8, 15.4, and 14.3 microns; Fe^+ at 26 microns.

A survey defining positions and intensity of these regions would serve as a basis for a great many detailed ground observations, and provide integrated fluxes for the larger objects difficult to observe from the ground.

B. Measurement of sources of small angular extent:

a) Selected Areas Survey - The present point-source IR survey of AFCRL is complete to M(4) = 1.5, M(11) = 1, M(20) = -3 magnitudes for 80% of the sky and will add significantly to our understanding of galactic structure. The longer integration times available on orbit permits observation of small regions, such as the Kapteyn Areas, to a statistical limit 3 magnitudes fainter.

b) Extragalactic objects - Forty-four galaxies were observed in the AFCRL sky survey. These observations indicate that with the longer integration time available on orbit, it will be possible to perform a detailed survey of the Virgo cluster.

Instrumentation

The telescope (less gimbals) will fit within a cylinder 51 cm diameter by 137 cm long. The telescope should be free to view in all directions, except that the optical axis of the telescope must not approach closer than 30° to any spacecraft structure, the sun, the moon or the Earth limb. The telescope will have a vacuum cover which must be removed when in space. This will be by remote command (operator), and the cover will be retained on the telescope or pallet for re-installation at the completion of the mission.

The basic HI STAR rocket telescope would be modified by the addition of an extended "sun shade" and by increasing the capacity of the LHe dewar. The resulting cryogenic telescope would be mounted in a fine-pointing two-axis gimbal to the spacelab pallet. Two modes of operation are envisaged. In the first, the telescope would be pointed to pre-selected celestial coordinates and remain at that position for a predetermined length of time. In this mode the internal chopper of the system would perform total modulation to permit measurement of the absolute sky radiance. Spectral data would be obtained by a multi-element detector array with a "wedge" filter providing narrow wavelength band isolation. In the second mode, the telescope would be pointed at preselected coordinates and a reciprocating scan would be generated by the gimbals. Point objects would be detected as they transit the detector elements. Multi-band interference filters would isolate selected spectral regions. In this mode surveys of the objects in selected areas would be accomplished. Both modes of operation could be employed on a single orbital mission, if desired.

Data from the multi-element array would be conditioned and preprocessed by the "on-gimbal" telescope electronics. Data would thus be transmitted to spacelab for further processing, recording and transmission to the ground.

Pointing and Other Spacelab Requirements

A special gimbal mount is required to point the telescope to within 1 arc minute of the desired celestial coordinate and maintain that line of sight with a stability of 20 arc seconds, peak to peak. In addition, the gimbal should be able to scan at rates on the order of several degrees/sec with a constancy of 1% of the scan rate. Scan amplitudes should be adjustable in the range 1 to 30°. Positional readout during scan should be accurate to $\pm$ 20 arc seconds.

Scan mode will require a special purpose memory unit with 16 bit word size capable of co-adding 30 input channels at the rate of 2000 words per second per input channel. Input words would be 14 bit length, (60K, 16 bit memory). Computer memory would be dumped at completion of area scan, and stored information further processed by on board computer to produce coordinates and amplitudes of sources detected. This can be easily accomplished with a computation rate of 2000 per second and a memory of 10 K. Total data to be "dumped" to ground in one day is determined by number of sources detected. Total is estimated at 10^5, 10 bit words/day = 10^6 bits/day (max.).

For all the observations desired, the orbit should be above 400 km altitude. A variety of orbital inclinations and launch times is desired, depending upon the main objectives of the flight. For example: an inclination of 28° would optimize observation of the regions near the galactic poles, while a sun-synchronous polar orbit would provide the best environment for scanning selected areas.

The telescope would consume 18 kgs. of stored liquid helium during a seven day mission. The LHe would be stored in the telescope dewar at a pressure of 3 atm. The boil-off gases could be exhausted into the local environment if this would not compromise other payloads on the mission.

Manned support would be required to operate the telescope and gimbals. It is assumed that pointing would be through interface with the spacelab computer and aspect reference system.

Of special concern to the infrared experiment is the cleanliness of the local environment. Class 5000 should be maintained in the unpressurized section. Effects of reaction jets is not known at this time; however, it is estimated that emission rates for particles 10-25 microns in diameter should be kept below 15/minute, if possible, and the H_2O vapor column density should not exceed about $10^{14}/cm^2$.

Space chamber tests of the first system would be highly desirable. The chamber should have an internal cold limer at $T \leq 20°K$.

5. TWO EUV EXPERIMENTS

S. Bowyer, University of California at Berkeley

A. EUV Imaging Telescope

A number of classes of galactic objects have been predicted to emit the bulk of their radiation in the EUV band between 100 and 1000 Å. This instrument will be capable of detecting such sources and locating their positions to within 10 arc minutes. If any extended EUV sources are discovered, this experiment can map them by simple pointing maneuvers. In addition, the spectral bandpass may be changed by placing different filters in front of the detector.

The great strength of this experiment lies in its imaging ability. In the EUV, the largest source of background is the resonant fluorescence of solar photons with the gases of the Earth's atmosphere. Thus, this radiation is diffuse, and appears distributed over the image plane. A point source, however, remains confined to one resolution element on the image plane. The net result is that the signal to noise ratio rises by the number of resolution elements, which is typically 1000.

The experiment shown in Fig. 1 consists of a grazing incidence imaging telescope which looks out the nose of the rocket payload and focusses the incoming rays onto a RANICON detector. The RANICON is composed of a microchannel array plate in front of a square resistive anode with signal outputs at each corner. When a photon strikes the plate, it emits a pulse of electrons which then strikes the anode. By weighting the relative strengths of the signals in the four pickups, one can tell where the photon struck the plate. Mounted directly in front of the RANICON is a thin filter designed to restrict the photon bandpass to a desired range of energies. Through the center of the mirror runs a baffle which eliminates rays that can strike the detector without being imaged. At the front of the mirror is a magnetic collimater which rejects electrons of energy up to 25 keV.

The telescope must be pointed and held to $\pm 1^\circ$. Each target must be observed over a total time ranging from 1 minute to 5 hours, though the observation need not be uninterrupted. The experiment should not be pointed closer than 30° to the Sun.

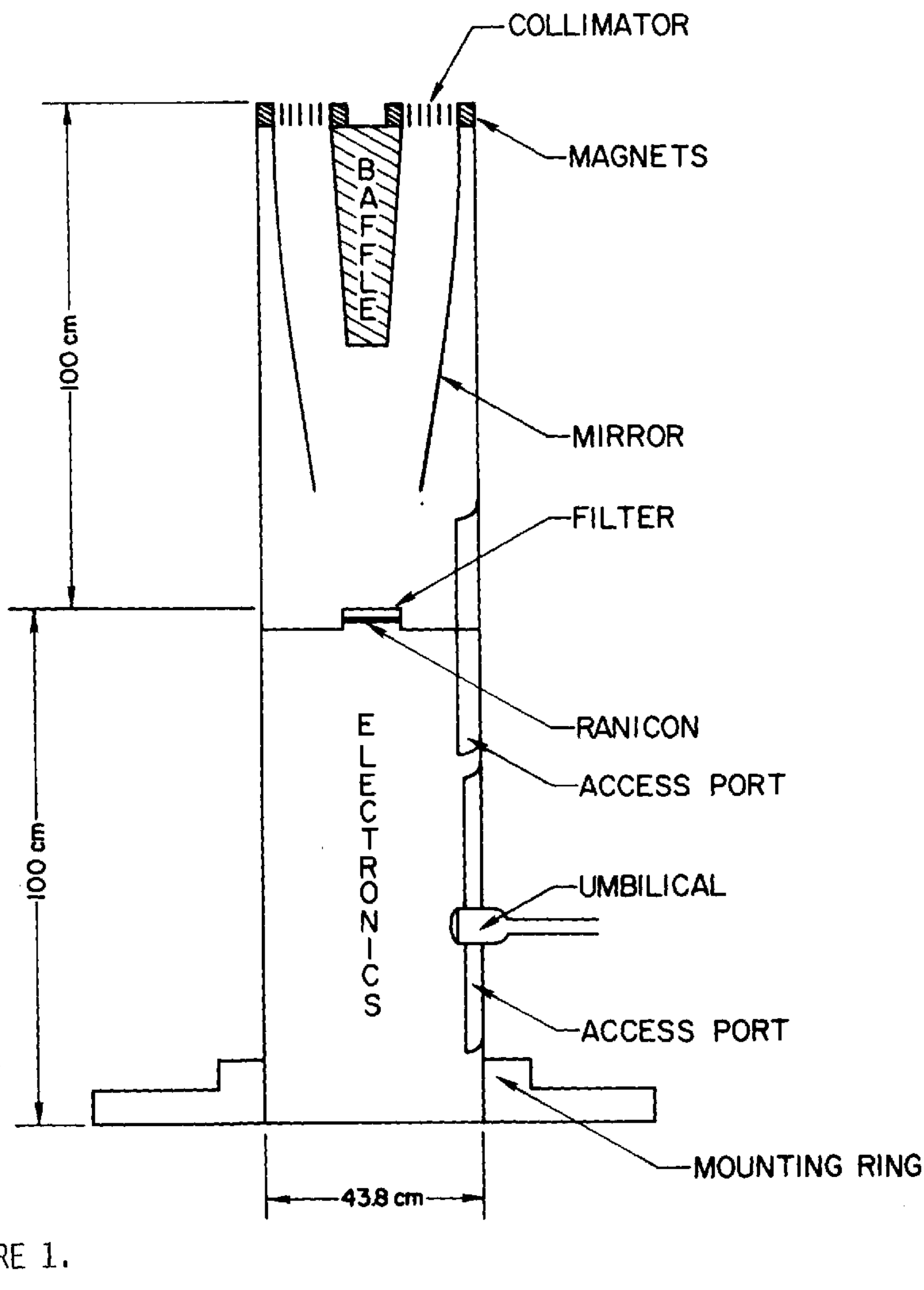


FIGURE 1.

SCHEMATIC OF EUV TELESCOPE

The experiment needs a bit rate of 20 Kbps when operating. Either direct telemetry or on board storage is acceptable. A record of the spacecraft aspect is required; 30' accuracy is required, 5' accuracy is desirable. Note that this is only a recording requirement and is not a pointing requirement. There will be a door on the side of the shell to allow access to electronics. This will be shut and not used during flight.

Four analogue outputs should be monitored intermittently either on board or on the ground. These outputs are:

i) Total Counting Rate

ii) RANICON voltage

iii) Pressure

iv) Current

B. EUV Spectrometer

The primary scientific goals of the EUV Spectrometer are summarized in the following four areas.

a) Geocoronal Airglow

The total existing data on both the atmospheric dayglow and nightglow in the range from 300 to 1050 Å is limited to a small number of measurements made with broadband photometers ($\Delta\lambda \sim 300$ Å) made with sounding rockets. The interpretation of these data is by necessity restricted, as it is based on assumptions as to the wavelengths of the radiation being observed. No moderate or high resolution studies have been made at these wavelengths and no spatial or temporal studies have been carried out. Extreme ultraviolet airglow measurements which should be carried out with the instrumentation include an exploratory search of the EUV band of the spectrum (300 to 1050 Å) to detect with high sensitivity all resonantly scattered and collisionally excited radiation and a search for locally enhanced regions produced as a result of specific sources of collisional excitation.

b) Aurora

The need of remote sensing of auroral phenomena becomes evident when one considers the vast scale, in both time and space, of the necessary measurements. Without considering details, it is obvious that adequate coverage of the aurora using only *in situ* observations is nearly impossible even with a relatively large number of satellites and rockets. Fortunately, the aurora by its very nature is amenable to study by remote sensing techniques. This characteristic contributes to the fact that the aurora is probably the most useful phenomenon for use in efforts to experimentally explore both the magnetosphere and the ionosphere. Currently no auroral EUV spectrum exists.

c) Plasmasphere

The HeII 304 Å line is optically thin at Shuttle altitudes and plays a unique role as a tracer for the plasmasphere. A study of this radiation will facilitate our understanding of the nature of this region and its interaction with the magnetosphere.

Observations of this line will permit detailed evaluations of competing models of the plasmasphere as was carried out by Paresce, Bowyer and Kumar (J.G.R., 79, 174, 1974). Number densities of ionized helium derived from this data may be more reliable than number densities derived from mass spectrometer data because of various experimental difficulties inherent in measurements with *in situ* detectors.

d) Local interstellar medium

It is now well established that the study of resonantly scattered 584 Å radiation from neutral helium will be central in our developing knowledge of the interaction of the local interstellar medium with the solar system. By the time of the Shuttle these studies should have delineated many of the parameters of this interaction, but it is likely that some effects such as changes with solar cycle and trace element measurements will not be fully explored. Studies of 584 Å He I and 1025 Å H I radiation will delineate these interactions and studies of other EUV lines such as predicted by Blum, Fahr, Axford, and others will define the trace element interactions.

Brief Description of Instrument

An optical layout of a possible EUV spectrometer configuration is shown schematically in Fig. 2. The incident light first passes through a baffle to eliminate off-axis radiation. After passing through the entrance slit the light then impinges on a platinum coated concave diffraction grating at an angle of incidence of

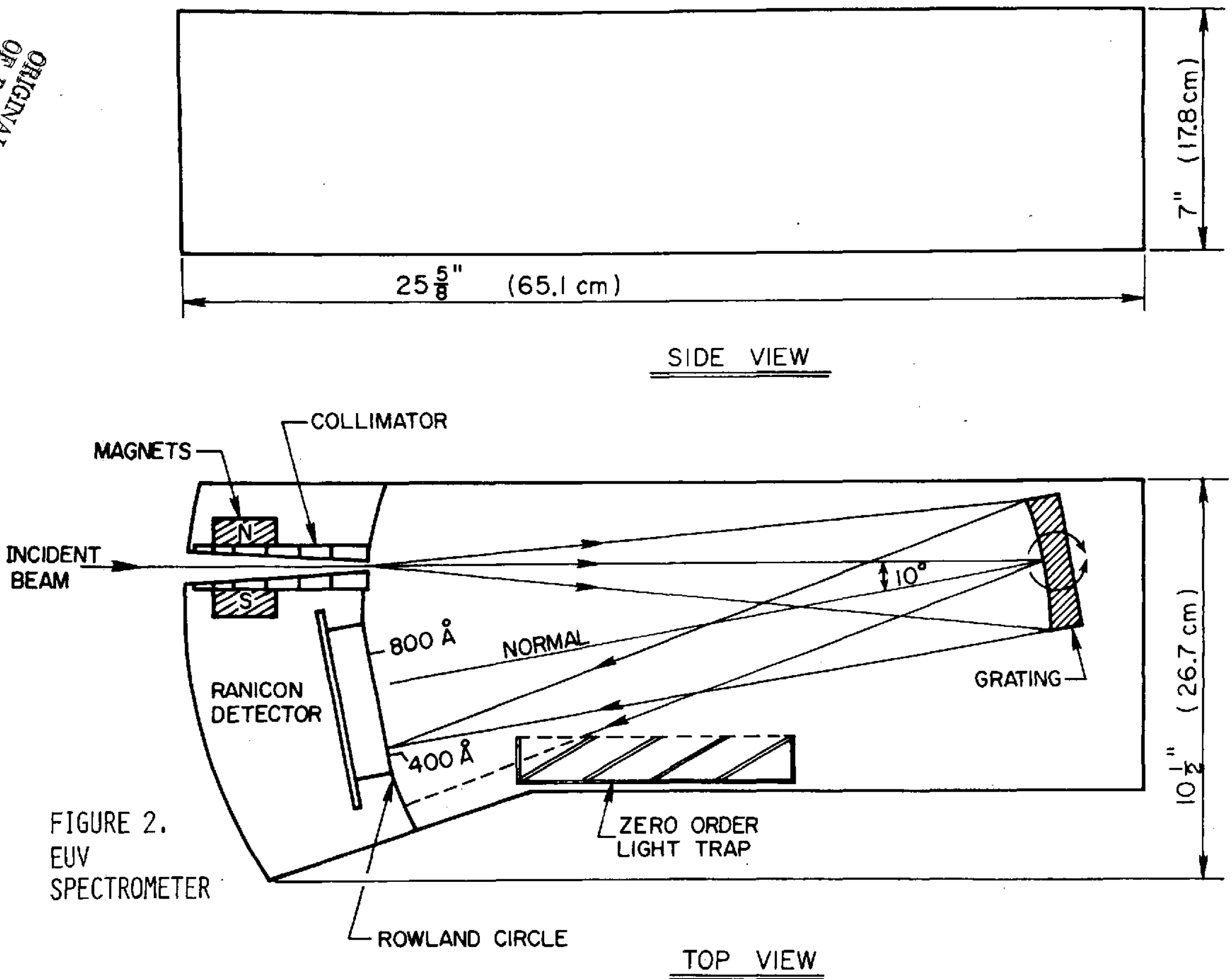


FIGURE 2.
EUV
SPECTROMETER

$\sim 10°$ The grating is an off the shelf Bausch and Lomb replica ruled at 2400 lines/mm, blazed at 1000 Å and having a radius of 400.7 mm.

The diffracted radiation is focused by the grating onto a RANICON situated on the Rowland circle. The inside order spectrum is used for packaging convenience. The RANICON serves as an efficient position sensitive EUV photon counter and consists of a 75 mm diameter channel electron multiplier array followed by a resistive anode. The front face of the CEM array is the photocathode, where photoelectrons are generated; an individual electron is multiplied about 10^7 times in traveling the length of a channel. The close spacing of adjacent channels permits good spatial resolution of an EUV spectral image. Each electron pulse produced by the CEM array is proximity focused onto the resistive anode. This anode is connected to low noise charge sensitive amplifiers, whose relative output pulse amplitudes give the location of the detected photon. The image is accumulated in a small random access memory for periodic readout.

The pointing requirements depend on the scientific objective.

a) Geocoronal airglow: random or programmed sweeps of overhead sky (1 to 5°/second) to accuracy of ± 10°.

b) Aurora: programmed sweeps of auroral arcs (1° to 5°/second); pointing at one geographical point (± °2) for duration of overhead pass.

c) Plasmasphere: programmed scans (1 to 5°/second) to accuracy of ± 5°.

d) Interstellar medium: random pointing or programmed sweeps of sky within ± 40° of zenith.

A record of the aspect is required with 1° accuracy. A maximum data rate of 40 K bps for intervals of 5 minutes is required for auroral observations. At other times, a maximum of 10 K bps is needed. The experiment must be purged with dry nitrogen until shortly before launch (typical flow rate: 1 cubic foot per hour).

C. X-Ray Payload

A high-time resolution x-ray experiment was also discussed. The Spacelab requirements for pointing and power were similar to those of the EUV payloads. A data rate of 200 K bps, 3 deploying doors, and gas supply bottles are included as special needs.

6. ULTRAVIOLET PHOTOMETER

A. Code and R. Bless, University of Wisconsin

Objectives

The purpose of this experiment is to establish the absolute energy calibration for a net of about 40 early-type stars in the spectral interval 925 to 3400 Å. Any member of this group of carefully measured stars would serve as a secondary standard of absolute flux for other UV telescopes in orbit.

Instrumentation

This payload is essentially identical to that flown on Aerobee rockets. It includes a spectrograph feeding 7 detectors sensitive between 900 Å and 1700 Å, each with about 50 Å bandwidths, along with four individual filter photometers sensitive to radiation from about 1900 to 3400 Å with bandpasses ranging from 50 Å to about 200 Å (see figure 1.)

The spectrograph consists of an 8-inch spherical mirror (whose field of view is limited to about 2 by 30 arc minutes), which illuminates a 600 line/mm plane diffraction grating blazed at 1200 Å. The resulting spectrum, with a dispersion of about 17 Å/mm, is focussed on Bendix windowless channeltrons fixed in the focal plane. These detectors are operated in a pulse counting mode. The payload is evacuated before flight to minimize out-gassing problems.

The second group of four photometers mentioned above are of a type we have flown many times before - - two-inch quartz refractors with six-layer MgF_2-Al interference filters to shape the ultraviolet pass bands - - and EMI 6256b photomultipliers operating in a pulse counting mode. The zero-order alignment detector used on the Aerobee will be permanently mounted on the shuttle payload.

Pointing and Other Spacelab Requirements

The instrument requires an absolute pointing accuracy of 2 arc min and a stability of 5 arc sec during an observation of 20 minutes. After orbital insertion the mission specialist will command small slew steps of about 10 arc sec and read the output from a zero-order detector in order to measure the absolute pointing offset between the telescope and SIPS mount. After on-orbit calibration of the pointing platform errors, the absolute pointing errors should be only ± 15 arc sec. In zero gravity, only thermal changes should affect the ability to maintain a 15 arc sec absolute pointing. Over

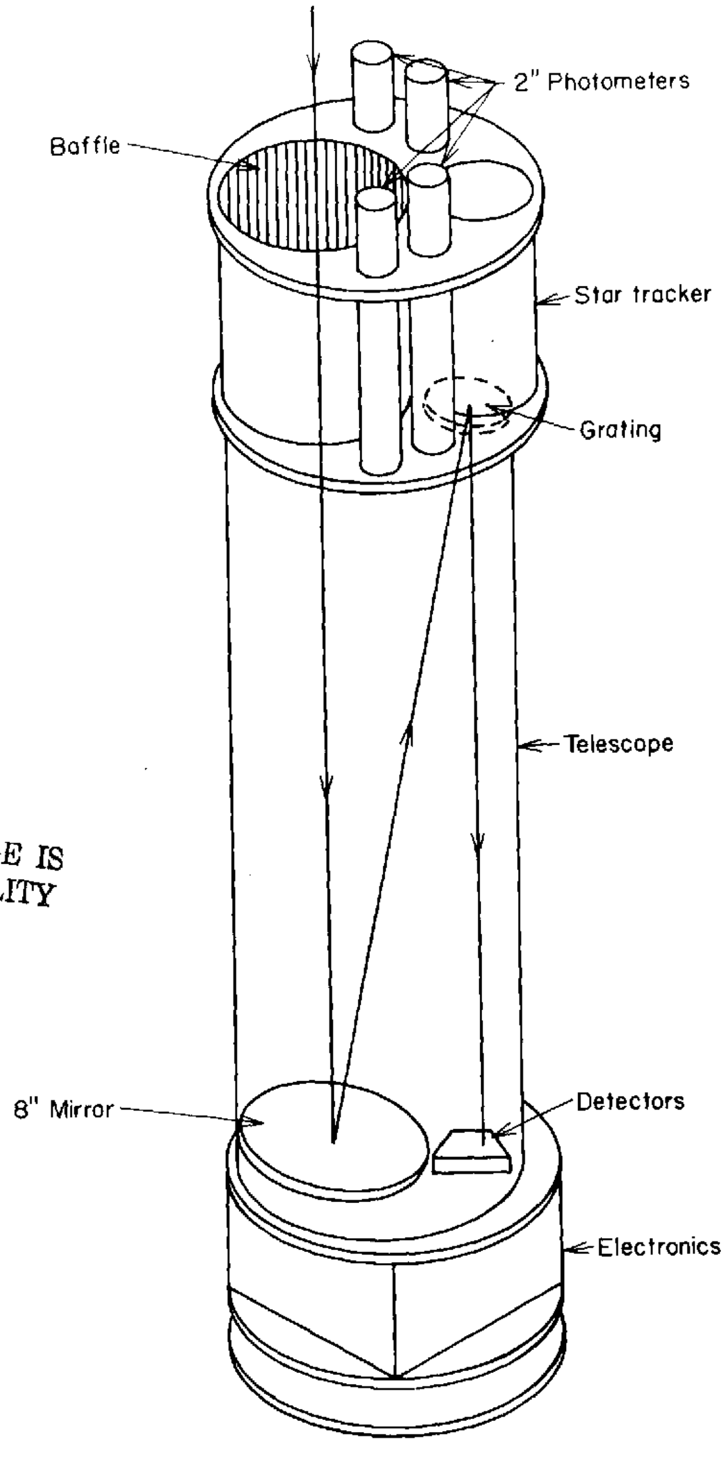


Figure 1. Wisconsin far UV spectrometer payload including four broadband photometers.

1 week mission we want to observe bright stars spaced over 1 hemisphere of the sky twice. Do not observe in sunlight; close shutter when near sun. Strict cleanliness precautions are necessary for calibration payloads and dry nitrogen will probably be required for purging during launch and re-entry.

Data is recorded by an on-board computer and transmitted via TDRS whenever possible. Check-out phase (1-3 orbital nights): payload commanded by mission specialist must have voice contact during this period; otherwise, we must have real-time data link. During the first day we should have several data dumps to control center. After the first day, one dump per day is sufficient.

After check-out the payload can be operated automatically from pre-programmed commands. These should be capable of quick revision. Since there are no movable mechanisms in this particular spacelab payload, control of the experiment can be relatively simple namely: turn on/off experiment low voltage, turn on/off experiment high voltage, turn on/off calibration lamp. Total lines needed: 3.

However, to take advantage of the power of spacelab's command ability a more flexible and safer (in the event of a payload subsystem failure) command sequence can be used with only a small increase in hardware. Each of the 12 detectors, counting the zero order detector, can be individually enabled or disabled through redundant payload hardware. Each detector would require 2 command lines, i.e., enable/disable detector LV and enable/disable detector HV. Additional command lines would be needed to provide LV to housekeeping circuitry, calibration lamp power supply and shutter open/close, zero-order detector field stop, and nitrogen purge on/off. Total command lines required: 30.

We would like about one month as close to flight as possible to recalibrate payload.

The following table summarizes our thoughts on some of the important parameters of a Spacelab flight. In order to maintain the basic philosophy of the sounding rocket program which has been quite successful over the years, the Spacelab support systems should be designed to satisfy the goals listed in the final column.

Comparison of Rocket, Satellite and Spacelab Missions for Optical Astronomy

	ROCKET	SATELLITE	SHUTTLE SORTIE
Scientific Objective	Specific measurement	Variety of invest. possible	Variety of invest.
Observing Time	Minutes	Months	Days
Lead Time	6 Months	2 - 3 Years	1 Year
Integration Time	Month	6 Months	Months (?)
Turn-Around Time	6 Months - 1 Year	Years	6 Months - 1 Year
Payload Weight Volume Aperture Qual. testing	~ 100 - 200 lbs $1 - 2 \times 10^5$ cm^3 up to 10"-12" telescope fairly extensive	100 lb - 2000 lb $4 \times 10^5 - 5 \times 10^6$ cm^3 up to 1 m very extensive	large like satellite like satellite relaxed (off shelf?)
Cost (Experiment)	\$.3 M	~ \$5 - 10 M	~ \$.5 M
Experiment \$/ Observing Time	~ \$1000/sec	~ \$10/sec	~ \$5/sec
Maximum Opportunity	2/year	1/5 years	up to 2/year
Data Analysis	Moderate effort	Large effort	Moderate effort
Interface Requirement	Minimal	Extensive	?
Man Interface	None	Unlikely	Possible (intended)
Training and Simulation	Little required	Extensive	Perhaps 3 months (?)
Pre-flight Calibration	Relatively simple, within weeks or days	Year lead time	Could be same as rocket
In-flight Calibration	Possible	Possible	Possible
Post-flight Calibration	Possible	Not Possible	Possible
In-house Organization Required	Small	Large	Small
Quick Reaction to new research or targets of opportunity	Possible	Only accidently	Possible

7. SCHWARZSCHILD CAMERA

A. Smith, GSFC

Objectives

The experiment is designed to measure faint surface brightness such as that associated with supernova remnants, planetary nebulae, emission and reflection nebulae, and galaxies. Most of the high excitation forbidden lines of O II, O III, Ne III, Ne IV, and Ne V from which temperatures and densities can be derived will be observable. In order to record ultraviolet surface brightness of other galaxies equivalent to 19th visual magnitudes per square arcsecond, exposure times will be on the order of 20 minutes.

Instrumentation

The camera has a low focal ratio and utilizes only two reflecting surfaces to achieve diffraction limited performance. Some of its characteristics are listed in the following table.

SCHWARZSCHILD CAMERA CHARACTERISTICS

	Camera 1	Camera 2
Aperture	141 mm	141 mm
Focal length	200 mm	176 mm
Effective Focal Ratio	f/1.7	f/1.4
Field of view	0.2 radians	0.2 radians
Focal plane diameter	40 mm	40 mm
Resolution	37 arc sec	12 arc sec (diffraction limited)
Vignetting	50% at edge of field	60% at edge of field

In the column labeled "Camera 1" are listed values which can be attributed to an existing Aerobee rocket payload. The characteristics of "Camera 2", an improved version of camera 1, are based on ray trace designs and diffraction analysis.

An optical schematic is shown in Figure 1. The secondary mirror is larger than the primary. The reflected light is imaged through a central hole in the primary mirror to a nearly flat focal plane which permits the use of different kinds of detectors. A circular baffle must be placed between the secondary and the primary mirrors to prevent direct illumination of the focal plane. When diffraction limited performance at 2 arc sec or worse is desired the Schwarzschild design possesses some obvious advantages. There are only two axially symmetric surfaces to manufacture, albeit they are general aspheres, and the focal plane is both flat and accessible. The major drawback is the large vignetting as indicated in the table. However, at the edge of the central 3 degrees of the field of view the vignetting is approximately 16% for camera 2.

The camera can be used by itself to obtain images of various kinds of nebulae and galaxies; in which case, broad band filters can be inserted in the light path preferably before the entrance aperture. Alternatively, an objective grating can be used to diffract the light from, say, well defined supernova filaments, before the light enters the camera. Figure 2 is an isometric drawing of the existing rocket payload which operates in the objective grating mode. As an indication of the system's sensitivity when using a microchannel plate (MCP) image intensifier and 2537 Å light, a suitable image is recorded in 8 seconds if the surface brightness of the source is 2.3×10^{8} photons cm^{-2} s^{-1} $ster^{-1}$ or 2.9×10^{3} Rayleighs. The resolution of MCP intensifiers cannot approach the resolution of the optics so that the more conventional magnetically focused image intensifiers or electrographic detectors would be a better selection for Shuttle use. These detectors may not provide the luminous gain of the MCP detectors, but the increase in observing time will much more than compensate for this minor deficiency.

Pointing and Other Spacelab Requirements

The attitude control system is of crucial importance to most optical astronomical experiments. While the SIPS, as presently conceived, adequately points instruments with optical axes parallel to the symmetry axes, it cannot handle "side lookers". For this reason, modifications to the Ball Brothers SIPS or an altogether new design should be undertaken.

Sometimes the signal for controlling the stability can be supplied

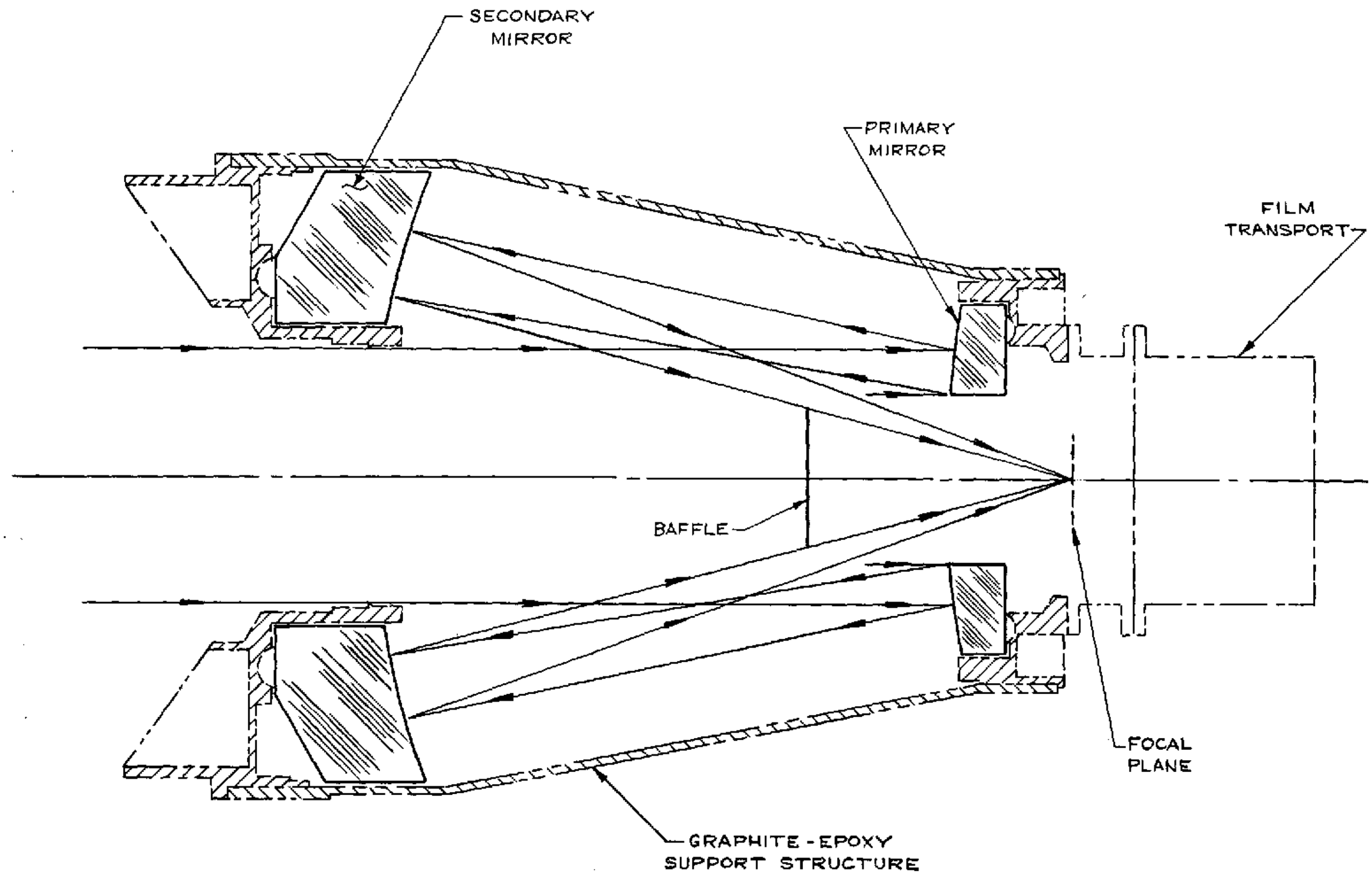


FIGURE 1 SCHWARZSCHILD CAMERA

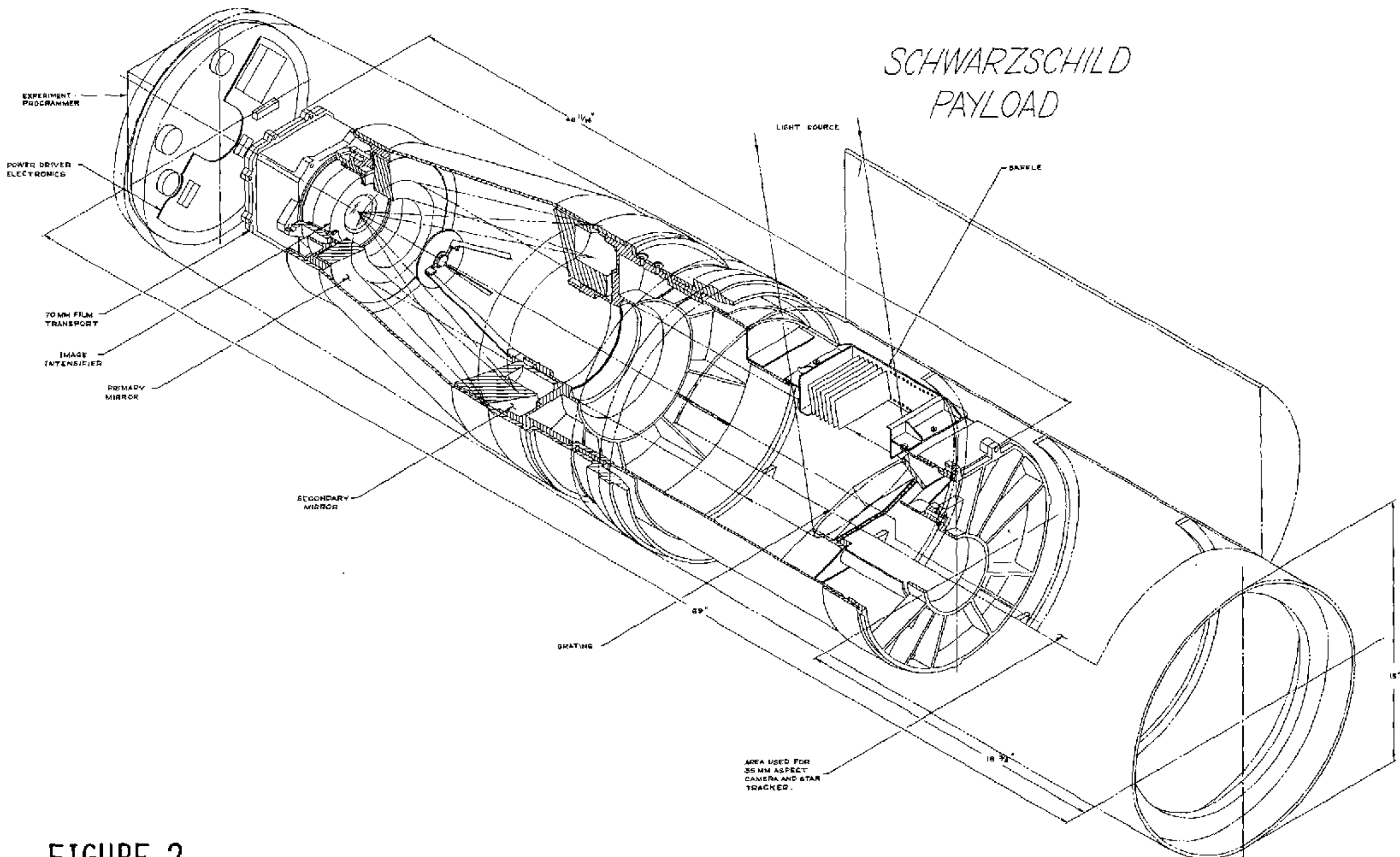

SCHWARZSCHILD
PAYLOAD
EXPERIMENT PROGRAMMER
POWER DRIVER ELECTRONICS
70 MM FILM TRANSPORT
IMAGE INTENSIFIER
PRIMARY MIRROR
SECONDARY MIRROR
LIGHT SOURCE
BAFFLE
GRATING
69"
AREA USED FOR 35 MM ASPECT CAMERA AND STAR TRACKER.

FIGURE 2

by the user, but this is not always the case. Startrackers used for this purpose on sounding rocket payloads are now routinely provided by the Sounding Rocket Division of GSFC and this service should be provided to the user of Spacelab. Often, as in the case of the Schwarzschild Camera observations, there is no star in the field bright enough to provide adequate signal for guidance purposes. In these cases two possible guidance methods come to mind. The first utilizes a single star tracker and exceedingly good low drift gyros, i.e. with drift rates ~ 0.001 degree/hour. In this case, the startracker is used to update the gyros while the gyros themselves provide error signals which are used to correct for short term pointing fluctuations. The second method would use two gimbaled star trackers, which when programmed to point at two acceptably bright stars would point the experiment optical axis in the desired direction. Ideally, the star trackers would prevent any significant drift and provide on the order of $\pm$ 15$''$ stability about all three orthogonal pointing axis. To narrow the limit cycle to $\pm$ 1 arc sec, rate integrating gyros could be used with periodic updates from the star trackers to minimize drift.

Since we are attempting to detect faint surface brightnesses we will want to observe in orbital night. Thus, we want to be pointed before entering orbital night and remain pointed throughout the duration of orbital night.

The detector used in the initial flights will very likely be an image intensifier plus film. An on-board computer could control the film advance, shutter and high voltage functions using inputs made by a payload specialist. However, the input could be made from the ground if the on-board computer had enough memory capacity to control the experiment during the times when there is no contact with a ground station.

If real time contact can be maintained, then in the case where an image tube such as a SEC vidicon is used, a quick look data reduction program and CRT display should be available at Goddard. If real time contact cannot be maintained, then a CRT display plus a minimized data reduction capacity should be available at a payload specialist station. We can envisage situations, particularly when orbit to orbit ground contact is not possible, when a payload specialist will be necessary to maintain the most efficient use of observing time. Thus, modifications to the observing program and evaluation of data provided in the quick look mode can be handled best on an orbit to orbit basis by a payload specialist.

In the case where film is used as the recording device we need to prevent "backheating" of the film after re-entry, or the capacity to bring the film into the Shuttle cabin before re-entry. We need the opportunity to evacuate and backfill our payload when it is mounted in the Spacelab. The maximum temperature gradients permitted in the optics section are about 0.2 °C/cm, which implies a temperature differential across the diameter of less than 8° C.

8. THREE ROCKET-CLASS PAYLOADS FOR SPACELAB

C. Lillie, University of Colorado

A. Microchannel Spectrometer

The Microchannel Spectrometer, shown in Figure 1, has been described by Lawrence and Stone (1975 in Rev. Sci. Instr.). It was flown on Aerobee 26.024 in January 1974 to observe Comet Kohoutek. The next flight of the payload is scheduled in October 1975 to observe Venus, Mars, and (perhaps) Capella. The instrument consists of an exponential baffling system which provides an 8' x 8' field of view; a concave grating with a one meter radius of curvature, and two Varian, Model 8964 microchannel plate (MCP) detectors in a chevron configuration with two resistive strip anodes. The MCP's are 3 cm diameter with 50μ channel spacing, and have a CsI cathode coated onto the input side of the detector. A trap door is provided to seal the instrument when not in use, and an ion pump maintains an internal pressure of 10^{-5} torr. The location (or wavelength) of each photoelectron pulse on the anode is determined by a charge division method. The electron pulse at the output of the MCP's forms charge pulses A and B at the input of two DC coupled, charge sensitive amplifiers. In the second stage we form two pulses of amplitude A and A + B. The divider then forms the signal 10A/(A + B) which is proportional to the distance along the resistor where the original pulse occured.

The flight instrument covers two spectral ranges: 500-950Å, and 1210-1660Å with a resolution of ∿2.5Å for point sources, and an effective aperture of 2 cm^2 out of a geometric area of 50 cm^2. For use on Spacelab the spectral range of the instrument would probably be ∿900 to 1800Å. For additional wavelength coverage a second instrument could be flown to cover the 1750 to 3100Å region. A third resistive strip anode could

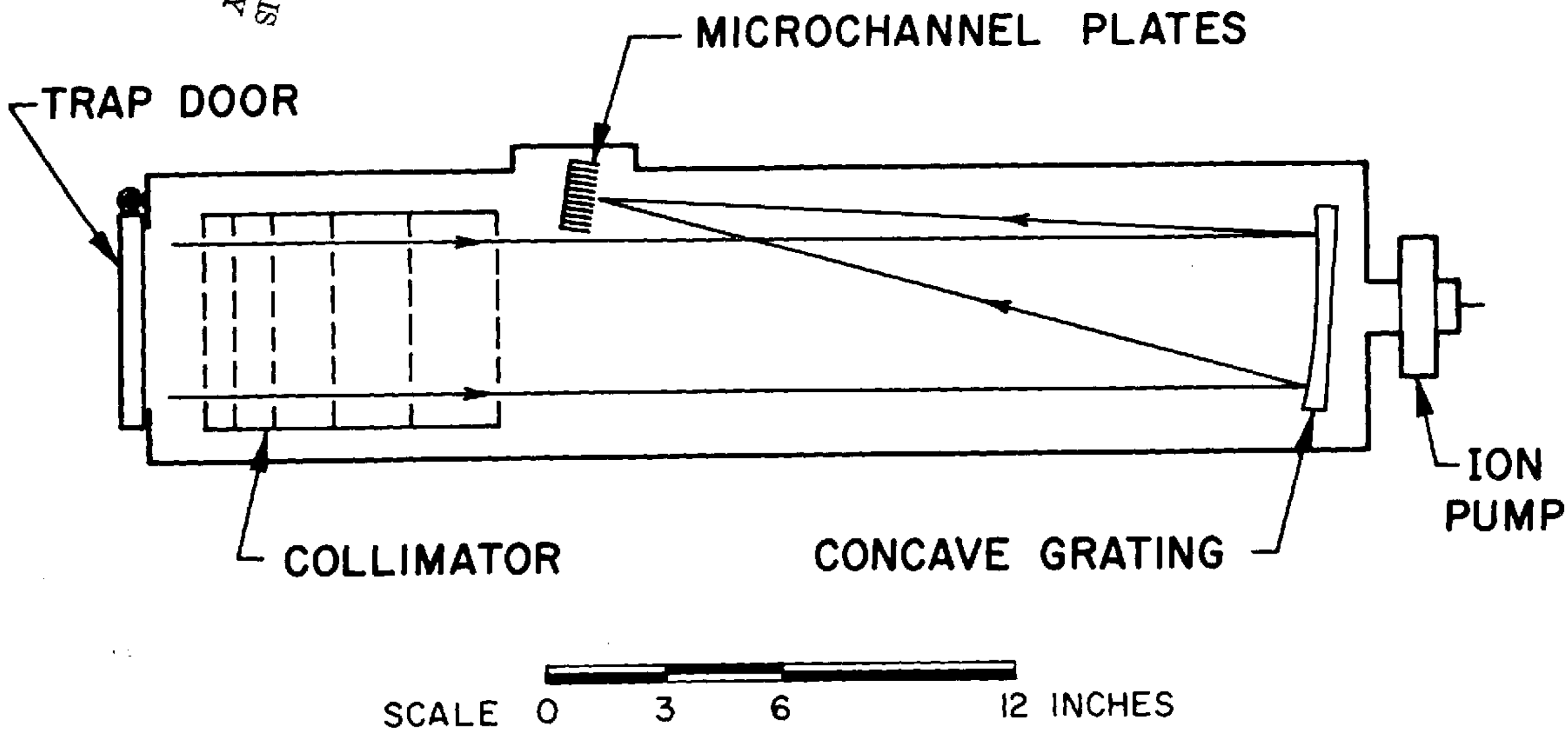


FIGURE 1. A SCHEMATIC DRAWING OF THE MICROCHANNEL SPECTROMETER.

be included to cover the 450 to 900Å region to observe nearby white dwarf stars and chromospheric and coronal emission features of late type stars.

In its present configuration, with a 1 hour integration the microchannel spectrometer can observe unreddened OB stars of V $\sim 13^m$ with 3% photometric accuracy, and with 2.5Å resolution. This sensitivity will permit the observation of nearby white dwarf stars, planetary nebulae, the brighter galaxies, late type stars, heavily reddened OB stars, OB stars in other spiral arms and the Large Magellanic Cloud, the planets, and the emission from comets as faint as $m_1 \sim 8^m$. An improved version of this instrument with $\sim$ 5 to 10x more sensitivity, and $\sim$ 1Å resolution is planned for future rocket flights. The major disadvantage of this instrument is its limited dynamic range: with the present (commercially available) electronics, pulse pile-up begins at $\sim 10^4$ counts/sec, making the pulse location less precise. Thus the current instrument will saturate on an unreddened $6^m.9$ O star. This limitation can be overcome somewhat with improved electronics. Another solution would be a motor driven iris to vary the aperature size. This maximum allowable count rate means the minimum integration time for 10^3 counts/channel will be $\sim$22 seconds.

The pointing requirements of this experiment are ± 1 arc min absolute, ± 30arc sec jitter. It can observe during the day if no sunlight is incident on the instrument and no illuminated surface is within $\sim 30°$ of its optical axis.

B. Ultraviolet Polarimeter

This payload which is scheduled for flight in the summer of 1975 consists of seven ultraviolet polarimeters which will be flown to measure the brightness and polarization of the zodiacal light, stars, airglow, and the Milky Way in the 1500 to 4100Å region. The instrument (Figure 2) consists of a 15 cm, f/1.4 cassegrain telescope, aperture, rotating analyzer,

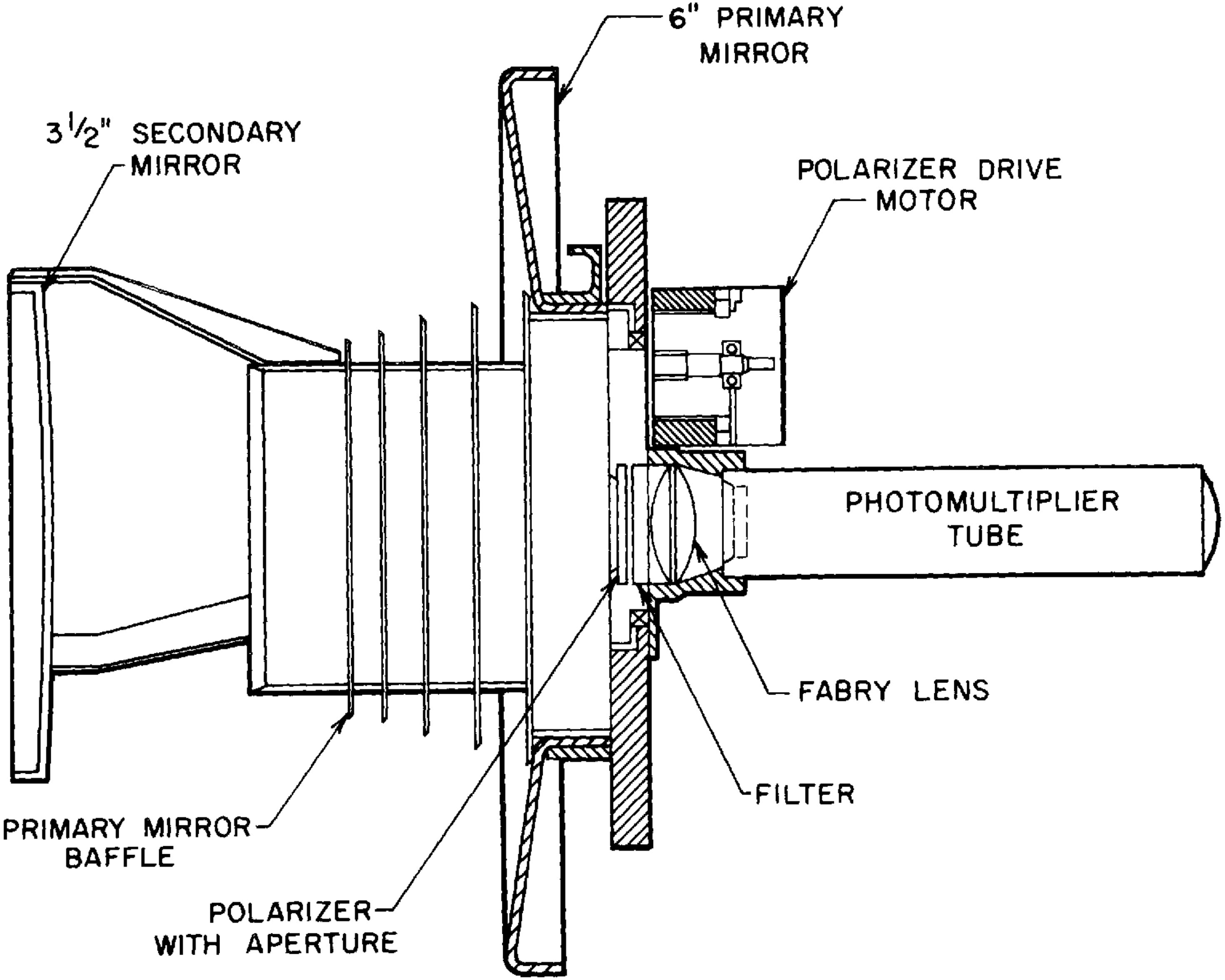


FIGURE 2. **ZODIACAL LIGHT PHOTOPOLARIMETER**

SCALE 0 1/4 1/2 1 INCH

a filter, Fabrey lens, and photomultiplier tube. A motor rotates the analyzer at 10 rps. A shutter provides a dark signal. High and low voltage power supplies, a pulse-amplifier/discriminator unit and a logic unit complete the instrument. This rocket polarimeter is a derivative of our Mariner Jupiter/Saturn 1977 Photopolarimeter Experiment shown in Figure 3. This instrument has an eight position filter wheel and an analyzer wheel with four discrete positions per measurement cycle: no analyzer (open), and analyzers with 0°, 60°, and 120° orientations. A four position aperture plate provides fields of view with diameters of 4°, 1°, 1/4°, and 1/16°. The sensitivity of the instrument is such that a $V = 10^m$ AOV star can be observed with $\sim$ 1% photometric accuracy in $\sim$ 100 seconds integration time. For sky background observations with the 4° field of view we receive $\sim$ 5000 counts per second per Rayleigh in the most sensitive bandpass. This means a surface brightness of $\sim 25^m$ per square second of arc can be measured with a signal-to-noise ratio of 10:1 and with long integration times, the threshold for detectability is about 5^m fainter. A modified version of this instrument on an early shuttle flight would observe stellar sources and the sky background and could determine the sky brightness due to outgassing from the spacelab and the shuttle. The overall dimensions of the MJS photopolarimeter experiment are 20 cm diameter by 34 cm long, plus a 71 cm shadow caster extension which permits observations to within 20° of the sun. In the sky brightness mode and for bright stars a pointing accuracy of ± 0.5°K would be sufficient; for faint stars ± 1' pointing is necessary.

C. High Resolution Spectrograph

The third payload shown in Figure 4, is a high resolution echelle spectrograph with a resolving power of $\sim 2 \times 10^4$ at Lyman-alpha. It has been proposed for flight on an Aerobee rocket in FY '77 with a 36 cm telescope with a servo-controlled secondary similar to one developed at Johns Hopkins University.

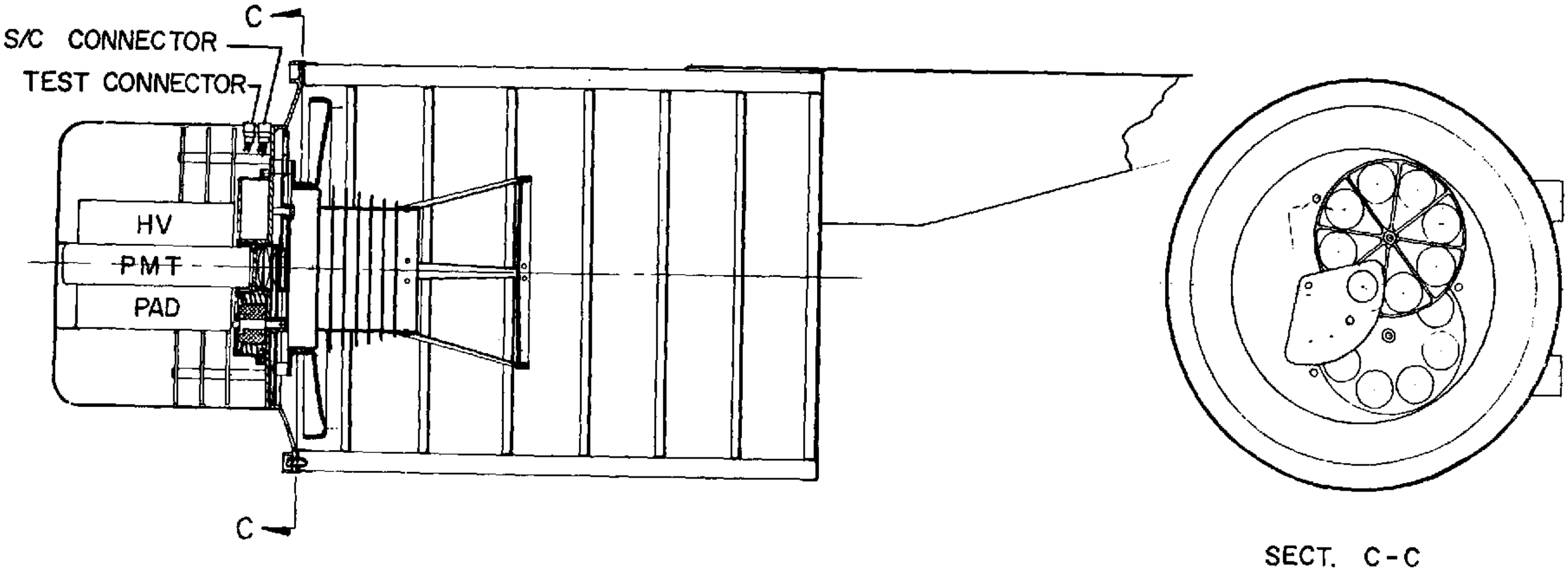
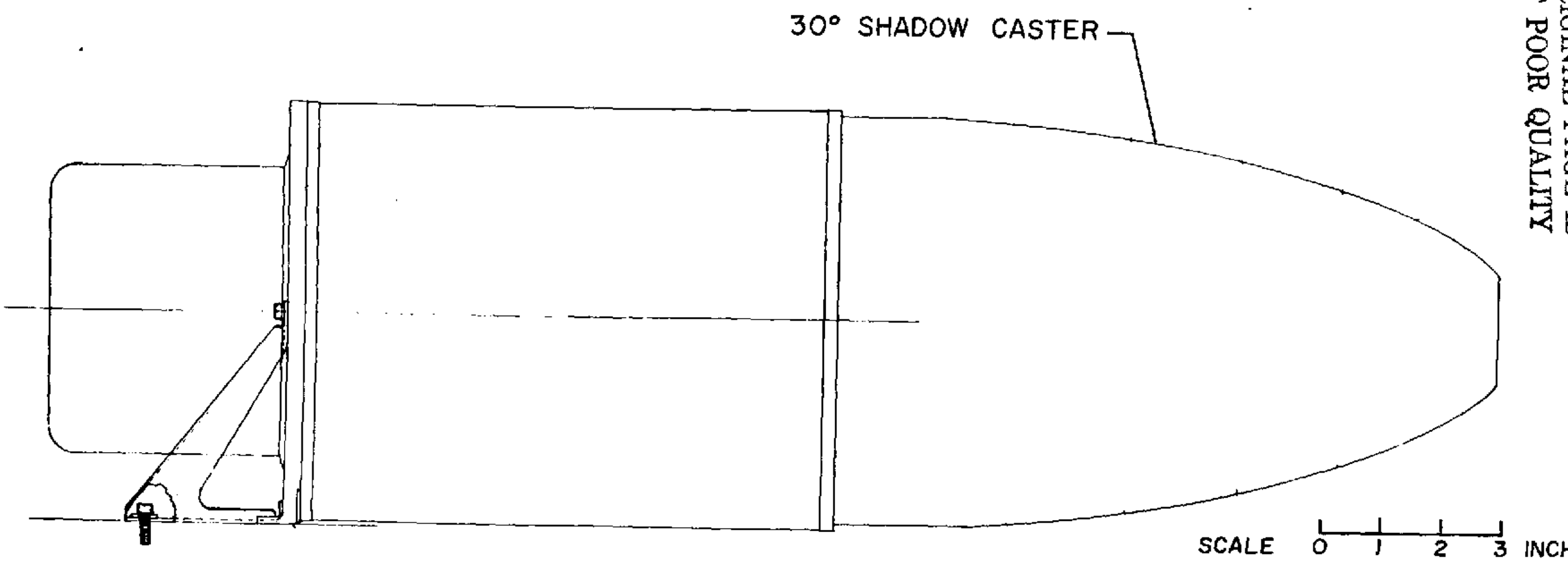


FIGURE 3. THE MARINER JUPITER/SATURN 1977 PHOTOPOLARIMETER EXPERIMENT.



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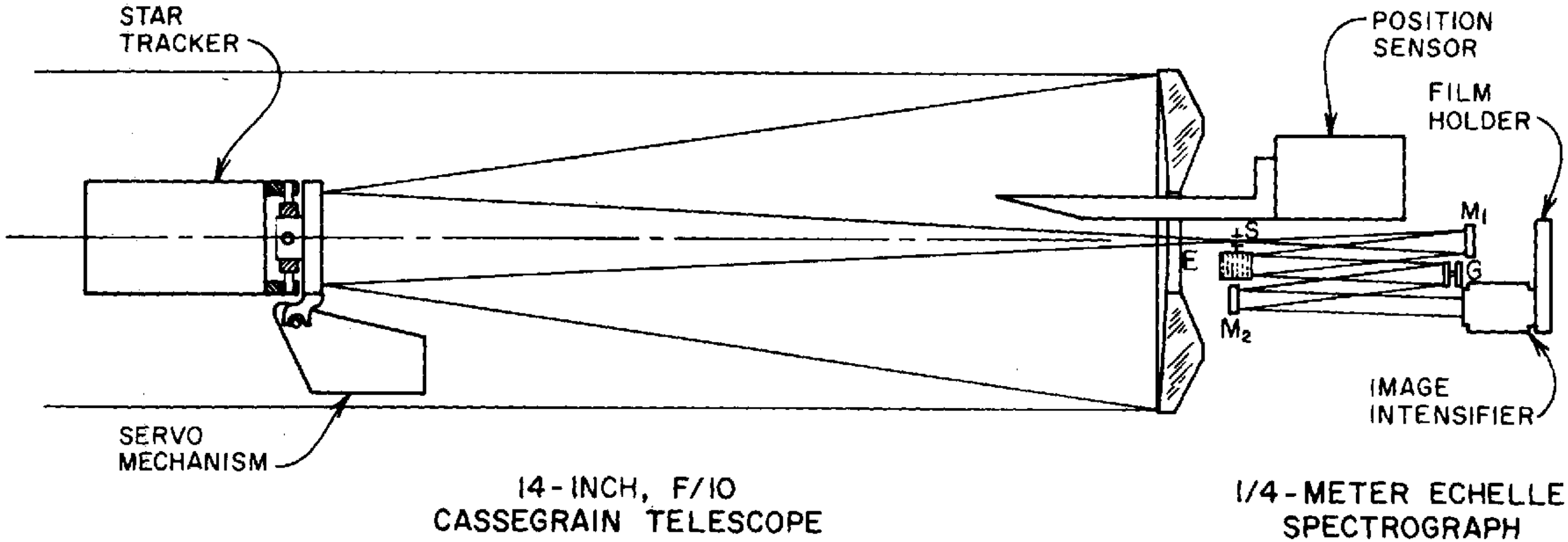


FIGURE 4. HIGH RESOLUTION SPECTROGRAPH EXPERIMENT

SCALE 0 3 6 INCHES

The most desirable detector system would seem to be the intensified film camera developed by Carruthers at NRL. On a sortie mission with a one hour exposure it should be possible to observe unreddened OB stars as faint as $V \sim 11^m$ with resonable accuracy.

General Spacelab Requirements

The preferred mode of operation of these experiments would be with manned support by a payload specialist from our investigation team. We would provide an instrument with a control unit mounted in the spacelab. If possible, we would provide a dedicated mini-computer with A/D inputs, oscillosope, and mass storage device to automate the instrument operation, collect and store data, and to provide a quick-look data analysis capability in orbit. This system would permit development of the hardware and software interfaces at the users institution, and result in considerable savings in overall cost of operations.

We anticipate operating these experiments $\sim$ 12 hr/day or $\sim$ 7 orbits/day. The number of objects observed per orbit would vary from 1 or 2 during routine operations to 5 or 6 during peak periods. During a seven day mission 50 to 100 targets could be observed. The orbital operations would be supported by personnel on the ground, at both mission control and the user's home institution. Quick look analysis of the data (payloads A and B) between operating shifts would permit modifications in the observing sequence to optimize the data collection. The detailed data analysis would be performed after the flight.

9. ADDITIONAL PAYLOADS

Several astronomers have proposed experiments for Spacelab but did not make an oral presentation at the Workshop. The documentation for these payloads is generally less complete than for the first eight groups. A brief description of each instrument follows.

A. Cryogenically Cooled IR Telescope

P. Dyal, Ames Research Center

The telescope is a folded Gregorian cooled with supercritical LHe and operating at 20°K. The detector is cooled to 4°K. A combination of flexible lines and rotary cryogen transfer joints may permit locating the LHe in a tank separated from the telescope. The forward end of the telescope tube is covered with a vacuum tight door that is remotely removed in flight for conduct of observations and avoidance of contamination. To avoid contamination inhibition of main thrusters will most likely be required during the observing program. Attitude control by vernier thrusters with wide (±20°) deadbands is acceptable. Controlled, programmed dumping of excess H_2O and venting will be required. The telescope, exclusive of the LHe and its tank, is 0.5m in diameter by 2m in length and weighs 75kg. The pointing requirements are 5 arc sec in pitch and yaw and 1° over ±90° range in roll (absolute) with a stability of 1 arc sec pitch and yaw and 1° roll. Two dimensional raster scan capability is desired. The data rate is 10kbps.

B. Mariner Jupiter/Saturn Ultraviolet Spectrometer

A. L. Broadfoot, Kitt Peak National Observatory

The spectrometer is 12.5 x 14.5 x 43 cm and has a 2° x 10° field of view. The instrument would look at earth airglow with a stability requirement of 1 arc sec. Spectral coverage is from 400 to 1800Å using a micro-channel plate anode array for a detector. The mass is 3.5kg and power needs are 2 watts.

C. IUE Spectrograph

A. Boggess, Goddard Space Flight Center

The telescope with echelle spectrometer is currently scheduled for launch on the IUE Spacecraft in 1977. With minor modifications to the optics and an updated detector system, a copy of the instrument would be a good experiment to fly on Spacelab. High resolution spectra would be obtained in the 1150 to 3000Å region. Operation on Spacelab would be from the IUE Mission Control Center located at GSFC. The package is 0.6m in diameter and 3m long with a mass of 107kg. Pointing requirements are 1 arc min pitch and yaw and 1° roll (absolute) with a stability requirement of 0.25 arc sec in pitch and yaw from internally produced error signals. The experiment uses 185W of power and has an SEC vidicon detecor read out at 40kbps.

D. Ultraviolet Telescope-Spectrometer

H. W. Moos, R. C. Henry, and W. G. Fastie; Johns Hopkins University

The experiment consists of an Aerobee payload of 38cm diameter by 178cm long. The prime targets would be the weak ultraviolet emissions from planets and cool stars. The detector is a micro-channel plate overcoated with CsI and is readout electronically. Pointing accuracy and stability needed is 3 arc sec with additional image stabilization provided internally by moving the secondary mirror, while tracking bright stars or planets. Total mass is 91kg. The data rate is 200kbps, but could be greatly compressed by onboard processing.

E. Narrow-Field Objective Spectrograph

R. C. Bohlin and T. P. Stecher, Goddard Space Flight Center

The payload is an Aerobee rocket experiment with a mass of 70kg and dimensions of 38cm in diameter by 150cm long. Targets include nebulae and faint stellar objects where there are no bright guide stars in the field. The detector is a micro-channel plate with a 35mm film transport for recording the spectra between 1150 and 2900Å. The main modification for Spacelab would be to increase the film supply from the current 25 frames to around 250 frames. The field of view is 17 x 24 arc min requiring an absolute pointing accuracy of about 2 arc min to center the target on the detector. A stability of 2 arc sec during a 30 min exposure would be compatible with the resolution of the detector and optics. Ideally, the film temperature should not rise much above 20°C at any time.

F. Far-UV Wide-Field Telescope (Wynne Camera)

S. Bowyer and co-workers, Berkeley

The instrument consists of three parts: a Wynne camera which may be used for direct photography or with an objective prism, a micro-channel plate detector with a cesium iodide photocathode, and a film magazine and drive mechanism. The useful field diameter is 4.5 degrees. This instrument was designed in France and has flown on a French Veronique rocket. The complete package weighs 60kg and is 57cm in diameter by 146cm in length. Pointing accuracy required is 3° with 6 arc min stability. If the detector is converted to electronic readout, a bit rate of 256kbps would be needed. The power requirements go from 30W for film to 100W after conversion.

ASTRONOMY MISSION STUDIES

W. Scull, GSFC

An an initial effort in looking at system interfaces and the potential problems of flying a variety of instruments on Spacelab, GSFC conducted a quick missions/system study of several instruments from the disciplines of UV/optical astronomy, solar physics, and high energy astrophysics. These initial efforts were started with the possibility in mind that indeed the early missions might include payloads from a variety of disciplines as opposed to a dedicated discipline mission. The preliminary studies were aimed at determining the feasibility of flying mixed discipline payloads and at planning the mission operations. Clearly, if the observational requirements of a particular discipline required a major share of the observational time, it might be better to consider missions dedicated to that discipline.

Three missions were studied:

1. Combined Solar, UV and High Energy Astrophysics Missions
2. Facility Class Mission
3. Free-Flyer Delivery Mission with additional attached instruments.

Instruments

Instruments selected as candidates for these studies are listed in Table 1 together with their equipment characteristics and requirements. The Mission 1 instruments were selected to exclude facility class instruments. The Orbiter would be used for pointing and orientation in conjunction with a Small Instrument Pointing System (SIPS) being studied by GSFC. However, Mission 1 would not require use of the ESRO-studied Instrument Pointing System (IPS). Thus, for the first study mission, the Solar Physics instruments consisted of an Externally Occulted Coronograph, (SO-1), a Solar X-Ray Telescope (SO-2), and a Solid State Flare Detector (SO-3), mounted on a single pallet. High Energy Astrophysics instruments included in a Large Area X-Ray Detector (HE-1), mounted on a single pallet, and a large Cosmic Ray Detector (HE-3), mounted directly to the Orbiter. A general purpose IUE-class UV Telescope (UV-2) and a Schwartzschild Camera (UV-1) for astronomy, mounted on a single pallet, completed the payload shown in Figure 1.

TABLE 1

EQUIPMENT CHARACTERISTICS AND REQUIREMENTS

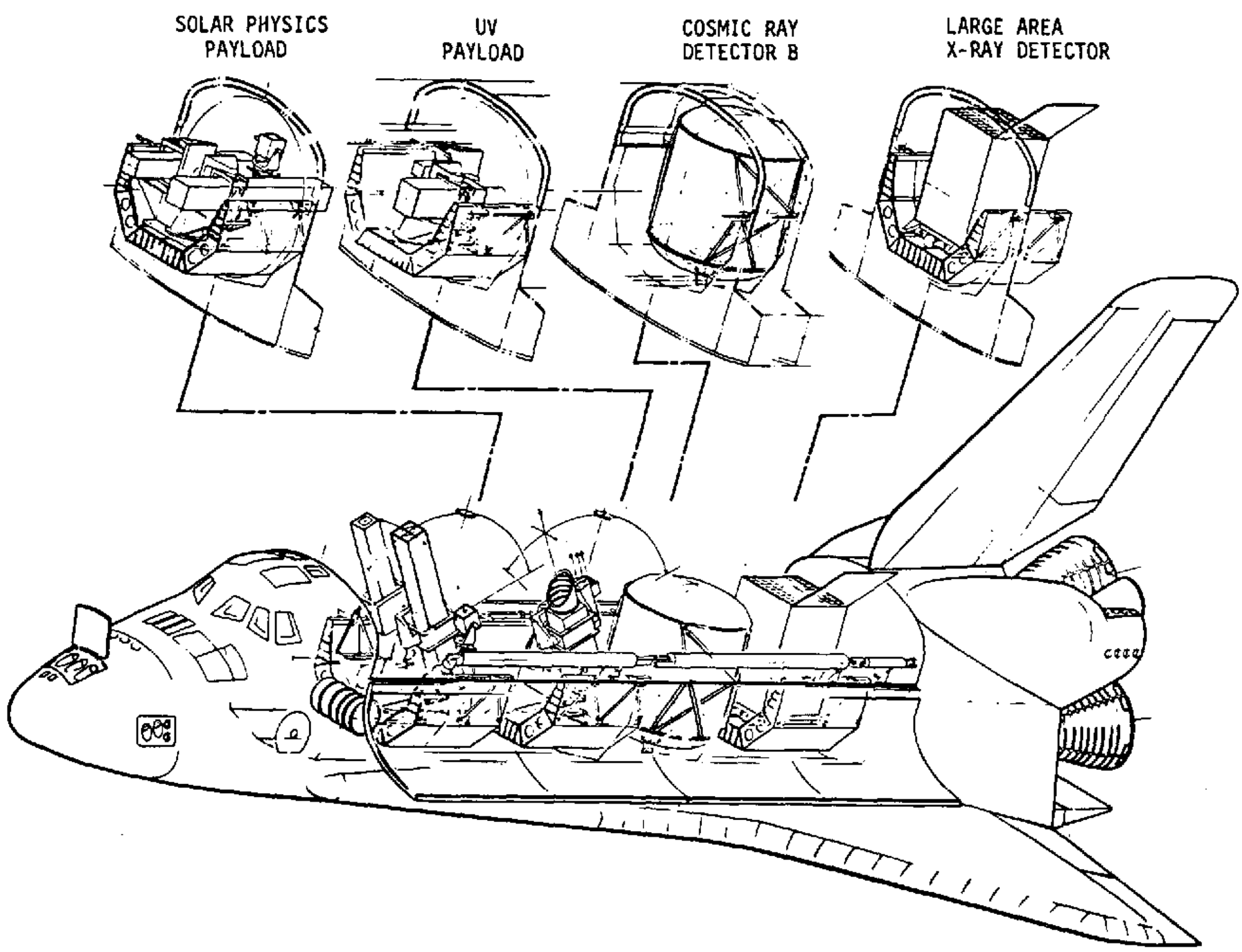
INSTRUMENT	UNIT SIZE (M) W OR D	H	L	UNIT DRY WT (KG)	POWER (W) OPER	PEAK	PK DUR (HR)	AC OR DC	TEMP LIMITS (°K) OPER MIN	OPER MAX	NON-OPER MIN	NON-OPER MAX	REMARKS
CORONA-GRAPH	0.60	0.60	4.60	204	40	100	0.0111	AC	291	298	275	325	295 ± 10°K INTERNAL
SOLAR X-RAY	0.50	0.50	4.00	250	50	110	0.1	AC	288	300	277	305	295 ± 10°K INTERNAL
FLARE DETECTOR	0.50	0.50	0.50	90	20	20	N/A	AC	292	296	277	305	
LARGE AREA X-RAY	2	3	2	2000	150	150	N/A	DC	273	308	243	308	<5°C GRADIENT ACROSS GLASS GRID
COSMIC RAY DETECTOR B	2.20	2.20	3.00	3000	90	90	N/A	DC	273	308	253	338	
SCHWARZ-CHILD CAMERA	0.38	1.90	-	129.5	80	100	0.1	DC	280	310	250	310	MINIMIZE TRANSIENT ΔT
GENERAL PURPOSE UV TELESCOPE	0.76	1.27	-	45.4	30	50	.00007	DC	273	313	273	313	

For the second study mission, the UV pallet and its payload was replaced by a single pallet carrying the UV facility-class (1-meter) telescope mounted on the ESRO Instrument Pointing System (IPS). As a result of the volume occupied by this instrument, it was necessary to reduce the High Energy Astrophysics payload to a single instrument, HE-1, while still including the Solar Physics payload.

The third study mission, for studying the combination of a deployable free flyer and a pallet payload, included the UV Astronomy payload of two instruments plus a typical free flyer. The Solar Maximum Mission (SMM) was chosen as a representative deployable free flyer.

For pointing the smaller instruments that required more accurate pointing than that provided by the Orbiter, a Small Instrument Pointing System (SIPS) was included in the study. This device contains two individually controlled sets of gimbals mounted on a single pedestal as shown in the following presentation on SIPS.

FIGURE 1
COMBINED SOLAR, UV, & HIGH ENERGY MISSION



Scientific Observational Targets

For operational flexibility of missions carrying instruments from different disciplines to exist, it was apparent that, while the Orbiter could be used for coarse pointing, simultaneous and independent observations with the various instruments would be necessary. Accordingly, a series of targets considered scientifically desirable for observations was developed by scientists in the three disciplines. These targets are shown in Figure 2, using an ecliptic coordinate system. Solar Physics requires solar viewing orientations, while the majority of the High Energy Astrophysics targets in this study resulted from requirements of the X-Ray experiment. It was desired to obtain 10^5 seconds of observations of the Andromeda Nebula (M-31), to scan the Vela remnant in a 6 x 6 scan matrix (36 individual matrix element observations of 23 minutes), and to scan the galactic plane in 1° steps, plus other targets as possible. Requirements of the Cosmic Ray Experiment were not as severe, it being desired that the instrument field of view not be occulted by any part of the Earth. UV Astronomy targets included 25 locations distributed over the sky. Of these 25 targets, 5 were first priority, with the remainder as second priority targets.

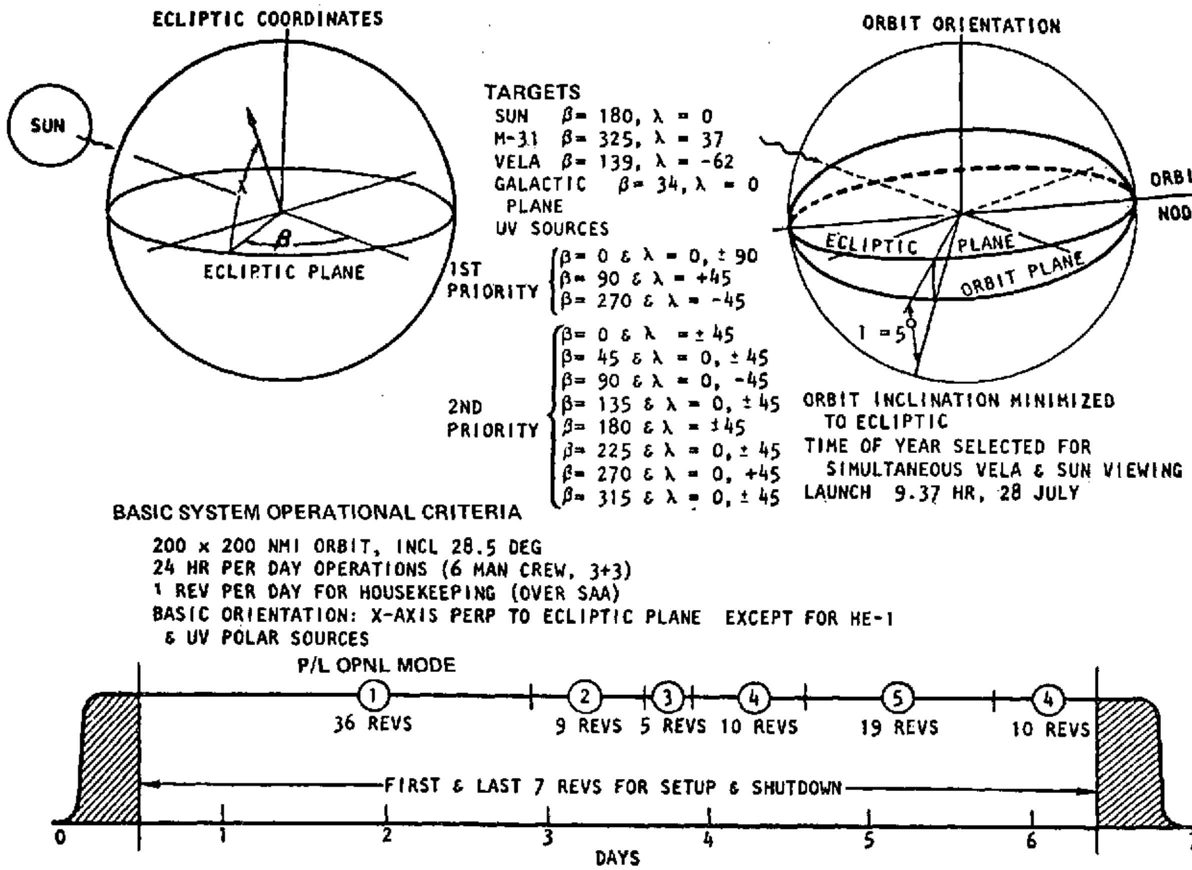
Orbit

A 200 n mi circular orbit at 28.5° inclination, with the launch timed to minimize inclination of the orbit to the ecliptic and allow simultaneous Vela and sun viewing, was considered. To maximize scientific data acquisition, 24 hours/day operation was considered. A six man crew, including 3 Payload Specialists for continuous observations, was included. One revolution per day was set aside for housekeeping purposes. By selecting a basic orientation of the Orbiter X-axis (longitudinal-axis) perpendicular to the ecliptic plane (X-PEP), except when making observations with HE-1, the large X-Ray Detector, and observations of the UV polar sources, it was possible to observe most targets with periodic roll/pitch maneuvers requiring about 6 minutes.

FIGURE 2

MISSION GEOMETRY AND SUMMARY PROFILE

Combined UV, Solar, High Energy Sortie Mission

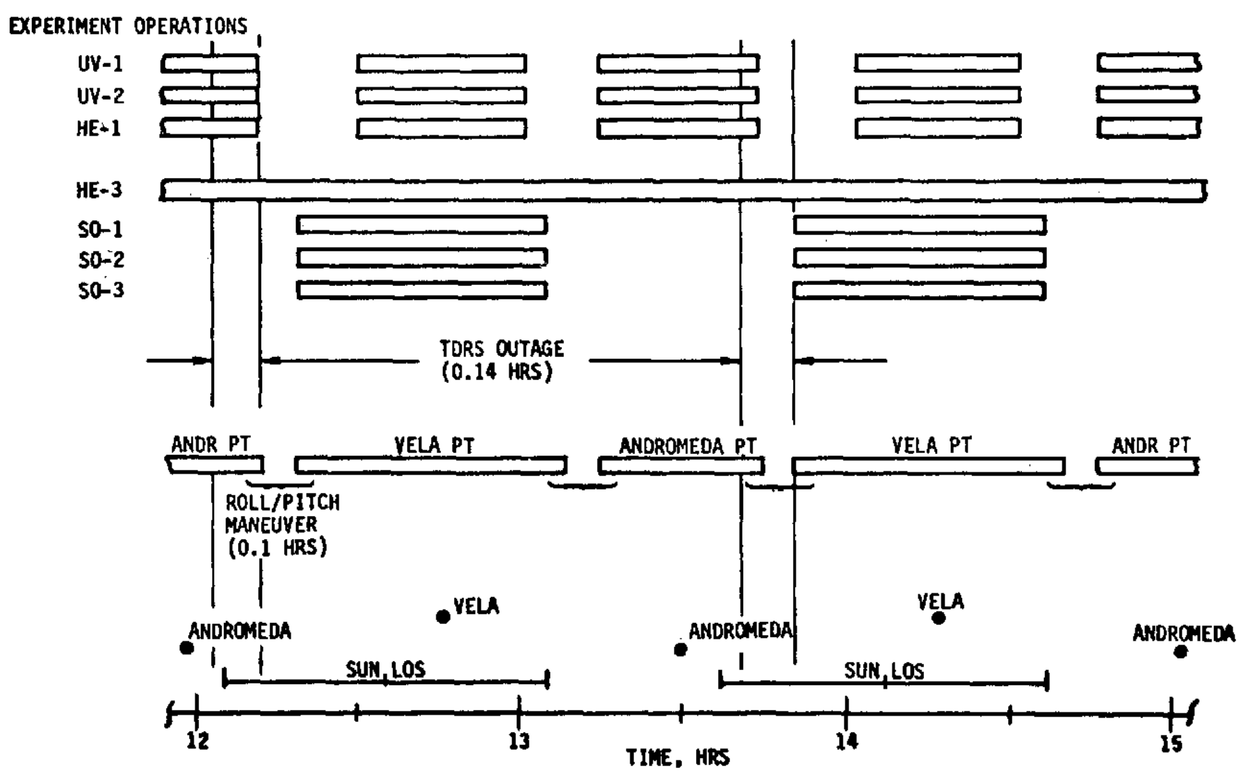


Operational Time Lines

For the 7-day duration of Mission 1, a time line of 5 different modes was established for 6 operational days, the first half and last half mission days being set aside for setup and checkout after launch and stowage and descent preparations prior to return.

Mode 1, shown in Figure 3 for a 2 revolution duration, was performed to prioritize X-ray observations. Periods of Andromeda and Vela pointing and the maneuvering times to change targets are shown. For HE-1, the actual times when the Vela/Andromeda sources would fall in the instrument field of view are shown. Also shown are the times when the Orbiter-Z axis would coincide with the sun line-of-sight (LOS). Around these times are then shown the times within which Solar Physics instruments, SIPS-mounted, could track the sun or UV Astronomy instruments also SIPS-mounted, could observe. Significant periods of solar and UV astronomy observations occur. In addition, since the Orbiter Z-axis is pointed away from the Earth during this time, cosmic ray observations are practically continuous. An estimated 10 percent outage of Tracking and Data Relay Satellite (TDRS) coverage per orbit is also shown.

FIGURE 3
X-RAY OBSERVATIONS (MODE 1)



Mode 2 was prioritized for 9 revolutions of observations of combined Solar Physics and first priority UV Astronomy targets. No observations with the X-ray detector are programmed, while again the cosmic ray experiment has continuous observations. Mode 3 prioritized cosmic ray observations for 5 revolutions, with a continuous 3.9°/minute roll rate about X-PEP. With the Z-axis maintained continuously away from the Earth, continuous cosmic ray observations are possible. Mode 4, prioritized for 20 revolutions of X-ray (galactic plane scanning) and solar pointing, includes significant coverage for all of the instruments. Mode 5, prioritized for 19 revolutions of X-ray and UV Astronomy observations, also includes significant coverage for all of the instruments.

A similar time line study was performed for Mission 2. The entire 6 day observational period was similar to Mode 1 of Mission 1 except that more accent was placed on priority observations with the UV Astronomy facility class telescope.

For Mission 3 the selected orbit was 332 n. mi. circular, with 30° inclination. Approximately 13 revolutions are used to check out and deploy the free flying SMM satellite and set up for UV observations, and 7 revolutions are used as in other analyses to prepare for descent. Once the free flyer is deployed, the entire observational time is available for UV observations, since no retrieval of a spacecraft is programmed.

Mission Performance Study

In summarizing mission performance for the available 6 day observational time, Mission 1 resulted in 168 UV Astronomy observations of at least 30 minutes duration each with all targets covered at least once. X-ray observations cover all requirements except for scanning only about half the galactic plane. Cosmic ray observations were possible more than 90% of the observational time with more than 50% of the observational time without any Earth occultations. Solar observations were possible about 60% of the available observational time. Mission 2, optimized for UV facility-telescope operations, include 178 observations, each of at least 30 minutes duration, with 15 observations of each of the five first priority targets. X-ray observations included a complete scan of the galactic plane and about 70% coverage of the Vela and

Andromeda targets. Solar observing totalled about 65% of the available observational time. Mission 3, once SMM was deployed, of course resulted in excellent UV Astronomy coverage.

Reaction Control System (RCS) Operation

From the operational aspects of the Orbiter, propellant usage does not appear to be a problem for the missions studied. Mission 1 required approximately 4400 pounds of the available 6040 pounds of Shuttle propellant. For all these missions of 7 days, it should be noted that approximately 50% of the propellant was used for payload operations, the remainder for ascent/descent and setup/shutdown/housekeeping (See Table 2).

TABLE 2 RCS PROPELLANT UTILIZATION

	RCS PROPELLANT, LBS		
	Mission No. 1 (UV-HE-Solar Sortie)	Mission No. 2 (UV Facil, Solar/X-Ray Sortie)	Mission No. 3 SMM Del, UV Sortie
Shuttle Ascent & Descent	1320	1320	1330
Payload Operations	2430	2080	1810
Setup House-keeping Shut-down	660	660	560
TOTAL	4410	4050	3700

RCS TANK CAPACITY = 6040 LB

Mission Weight

Mission 1 weight (Table 3) included about 31,500 pounds, of which approximately 27,600 would be payload chargeable landing weight. With a 65,000 pounds up-weight capability, the mission does not appear weight

TABLE 3 MISSION 1 WEIGHT SUMMARY

	Provided By		
Equipment	Orbiter (P/L Chargeable)	Spacelab	ASP
Experiments			12132 • UV Array • HE Array • Solar Array
Structural	1302 • Bridge Fittings • Keel Fittings	4983 • Basic Pallets • Igloo	200 • HE Supports
Elec. Power Syst.	1450 • EPS Tankage • EPS Reactant	122 • Exper Inverter • Subsyst. Inverter	--
Command & Data Handling Syst.	--	382 • Recorders • Computer	140 • Formatters • C&D Panel
Pointing & Stabilization	3090 • RCS Propellant	--	6920 • SIPS(3) • Flare Det Mtg
Communications	263 • TDRS Wide Band ANT	--	--
Crew & Provisions	481 • Personnel (2) • 14 M-D Provisions	--	--
TOTALS	6586	5487	19392

Mission Grand Total = 31465

limited. Mission 2 with an up-weight of about 24,700 pounds and approximately 21,100 pounds on landing appears volume limited rather than weight limited. For Mission 3 the total down-weight after deploying the SMM spacecraft and returning to Earth without recovering any free flyers would be about 12,600 pounds. This number results from the release of the SMM (3824 pounds) and the use of approximately 10,000 pounds of RCS propellant during the mission after a total lift-off payload weight of approximately 27,250 pounds.

Longitudinal Center of Gravity (CG)

In studying the placement of instruments in the cargo bay, the location of the center of gravity had to be considered in addition to instrument fields of view, pointing system coverage capabilities, etc. The CG's of the payloads for the three missions studied all fell within the longitudinal allowable CG envelope both wet and dry (propellant expended) and below the limits of take off and landing weights (Figure 4). Similar considerations of CG envelopes in the other two axes also indicated no problems. The significant shift forward (wet to dry) of the CG for Mission 3 is due to deployment of the SMM.

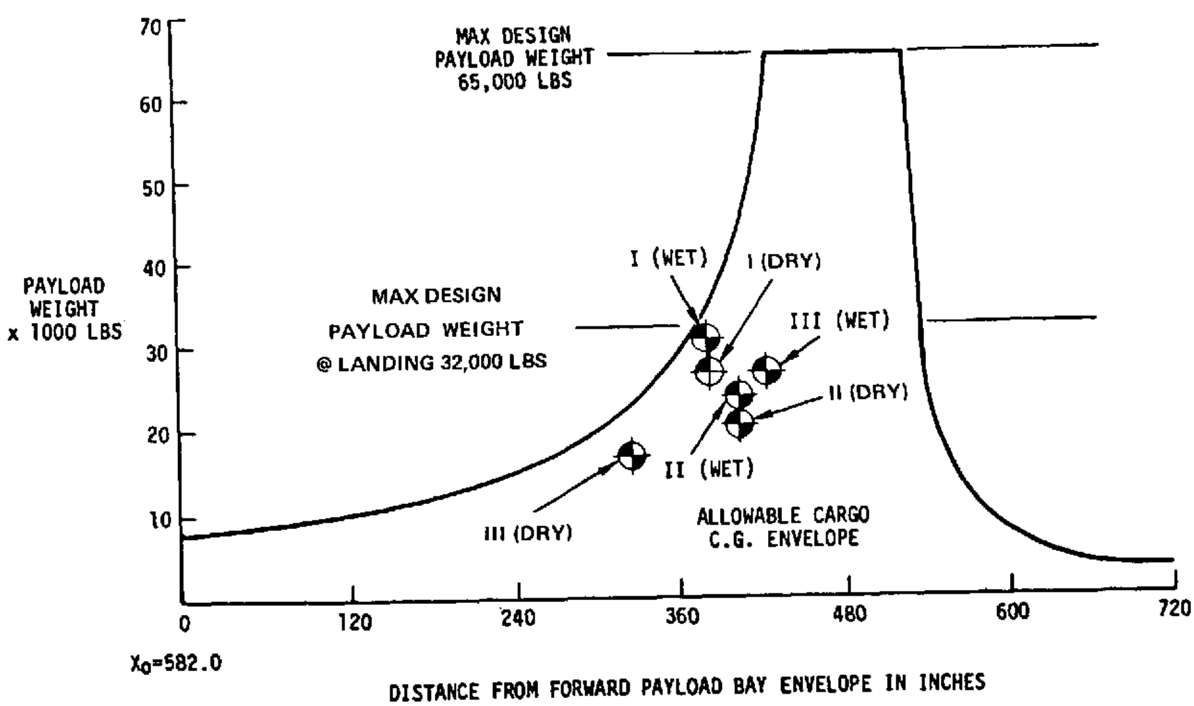
Orbiter Attitude Control

For controlling the attitude of the Orbiter, the primary reference is a navigation base located in the crew area. Part of this nav-base, an Inertial Measurement Unit (IMU) determines the attitude reference. Attitude control is by coarse (950 lbs. thrust) and fine (25 lbs. thrust) bipropellant (monomethylhydrazine and nitrogen tetroxide) jets. Several modes of control - free drift, inertial hold, and source tracking of a fixed reference are available.

Line of sight (LOS) attitude control may be from experiment mounted sensors in the cargo bay, with nominal LOS determination with respect to the nav-base. However, if nav-base references are used, a bias of 2-4 degrees between the nav-base location and the cargo bay due to structure deformations must be considered. Anticipated errors in the inertial mode can be $\pm 0.5^{\circ}$, based upon $\pm 0.1^{\circ}$ misalignment of the IMU, $\pm 0.25^{\circ}$ error in the control system, and gyro drift of $\pm 0.15^{\circ}$ for one orbit. Realignment of the IMU approximately each 1.5 orbits would be required to compensate for gyro drift. In a source tracking mode, attitude errors could be $\pm 0.35^{\circ}$ since no gyros are required. This type of control could also be maintained for longer periods of time dependent upon RCS consumption, thermal requirements, etc.

The RCS thrusters located fore and aft on the fuselage, can maintain variable rates of 0.001 to 0.033°/sec or 0.25 to 1°/sec and accelerations of 0.025 to $0.035^{\circ}/sec^2$ or 1 to $2.5^{\circ}/sec^2$ for the fine and coarse thrusters, respectively. Rates can be maintained to 0.1°/sec with any one vernier jet non-operational.

FIGURE 4
CARGO CG LIMITS (ALONG X AXIS)



Command and Data Handling

For the Command and Data Handling Subsystem (CDH), the general rules and assumptions (Table 4), used in the study are basically that the payloads will use existing Spacelab hardware where possible. Data requirements were calculated for a 90 minute orbit and assuming only 60 percent TDRS coverage. This assumption includes the previously discussed 10 percent TDRS outage, plus coverage unavailabilities due to number or locations of Ku-band transmitting antennas on the Orbiter, availability of the Ku-band link, etc.

TABLE 4

COMMAND AND DATA HANDLING GROUND RULES AND ASSUMPTIONS

- Caution & Warning Must Have Redundant Sensors & Transmission - Preferably Using Different Techniques
- Payload Power On/Off Dedicated Control Panel
- Command & Control Of Experiment & Pallet Subsystem Using Data Bus (Multiplexing) Techniques
- The CDHS For Experiments Is Basically Independent Of Orbiter & Located In Igloos. Orbiter Computer Used As Backup
- Utilize Space Lab Equipment Whenever Possible Including Computer Software
- Data Rates Based On 90 Minute Orbit, 60% TDRS Coverage, 1K bps Per Instrument For Housekeeping, 10 K bps For Space Lab Housekeeping
- Data Transmitted In Real Time Requires Use Of K Band Capabilities Of TDRS
- Scientific Observations Are Directed From The Ground Via Orbiter RF System. On-Board Operator Function Is Control Of Instrument Operations

Data capacity per orbit totals 3400 megabits; and with an estimated 40% outage of TDRS, storage of approximately 1400 megabits per orbit would be required. A comparison of the experiment requirements in terms of housekeeping telemetry rates, peak experiment data rates (real time), data storage and playback rates, and total storage capability indicates that the current Spacelab CDH system can handle the requirements of the payloads studied. A sample data rate profile for Mission 1 in Figure 5 is matched against the data handling capabilities in Table 5. Possible command/control functions and typical tasks that would be required to be actuated from/through the Orbiter are listed in Table 6. Although these functions and tasks are shown as actuated by the PSS or MSS (Mission Specialist), it is possible that many of these functions or tasks could be accomplished from the ground. These considerations are still under study.

TABLE 5

C&DH REQUIREMENT CAPABILITY COMPARISON

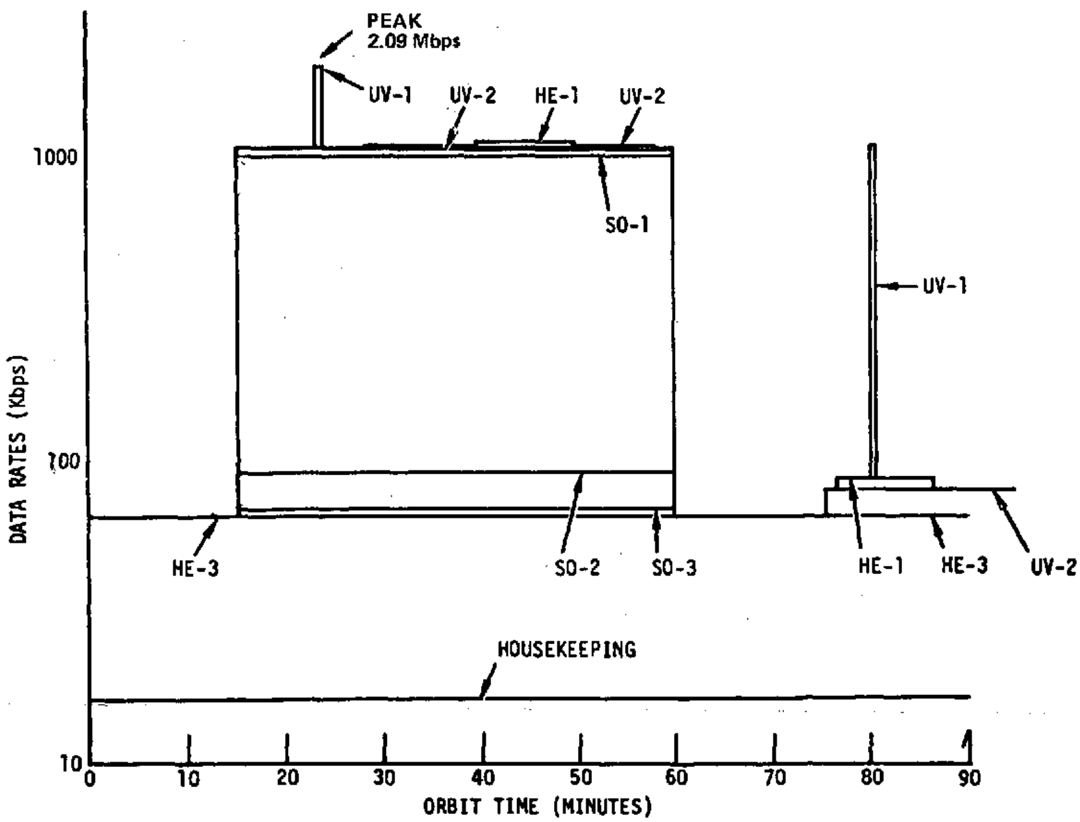
	EXPERIMENT REQUIREMENT	CAPABILITY	
		SPACELAB	ORBITER
HOUSEKEEPING DATA ACQUISITION RATE	17 K BPS	1 M BPS (DATA BUS)	PERFORMANCE MONITOR SYSTEM
PEAK EXPERIMENT DATA ACQUISITION RATE	2.09 M BPS	50 M BPS (DEDICATED COAX)	N/A
COMMAND/CONTROL DATA RATE	TBD	1 M BPS (DATA BUS)	N/A
RECORDING RATE/TIME	2.09 M BPS PEAK 1.09 M BPS (NORMAL) FOR 36 MIN (MAX)	7.5/15/30 M BPS 80/40/20 MIN	VOICE REC., LOOP MAINT REC.
RECORDER STORAGE ORBITER TO GROUND TRANSMISSION RATE	1720 M BITS 1.08 M BPS	360 M BITS N/A	1 CHANNEL (K BAND) AT 50 M BPS 1 CHANNEL K BAND AT 1 M BPS 2 CHANNELS S BAND AT 64 K BPS
GROUND TO ORBITER TRANSMISSION RATE	TBD	N/A	1 M BPS K BAND 2.4 K BPS S BAND
COMMAND/CONTROL DISPLAYS	KEYBOARD/CRT EXP DEDICATED C & W	KEYBOARD/CRT	~3400 IN^2 OF PANEL AREA
CAUTION/WARNING		DISPLAY/AUDIO	DISPLAY/AUDIO

TABLE 6 COMMAND/CONTROL FUNCTIONS FROM ORBITER (PSS/MSS STATIONS)

FUNCTION	TYPICAL TASKS
• Experiment Management	• Control of Instruments • Data-Record/Dump • Data Selection • Type Changes
• Performance Monitoring	• On/Off • Temperatures/Pressures • Data Stream Stability
• Instrument Pointing For Target Acquisition	• Control Gimbals To Point Instrument (3 Axis) • Axis Transformation
• Caution & Warning	• Instruments/Gimbals Locked/Free • Instrument Position • Critical Temperatures, Pressures
• On-Board Checkout	• Functional Test Of Major Assembly As Indicated By Performance Monitoring On Ground

FIGURE 5

TYPICAL DATA RATE PROFILE



Electrical Power and Energy

Estimated average and peak power and total energy requirements for the missions studied are compared to the Spacelab/Orbiter capability in Table 7. As with the CDH system, the Electrical Power System (EPS), using one 840 KWH reactant kit, plus the 50 KWH furnished by the Orbiter, can readily handle the power/energy requirements for a 7-day mission.

TABLE 7

EPS REQUIREMENTS/CAPABILITIES COMPARISON

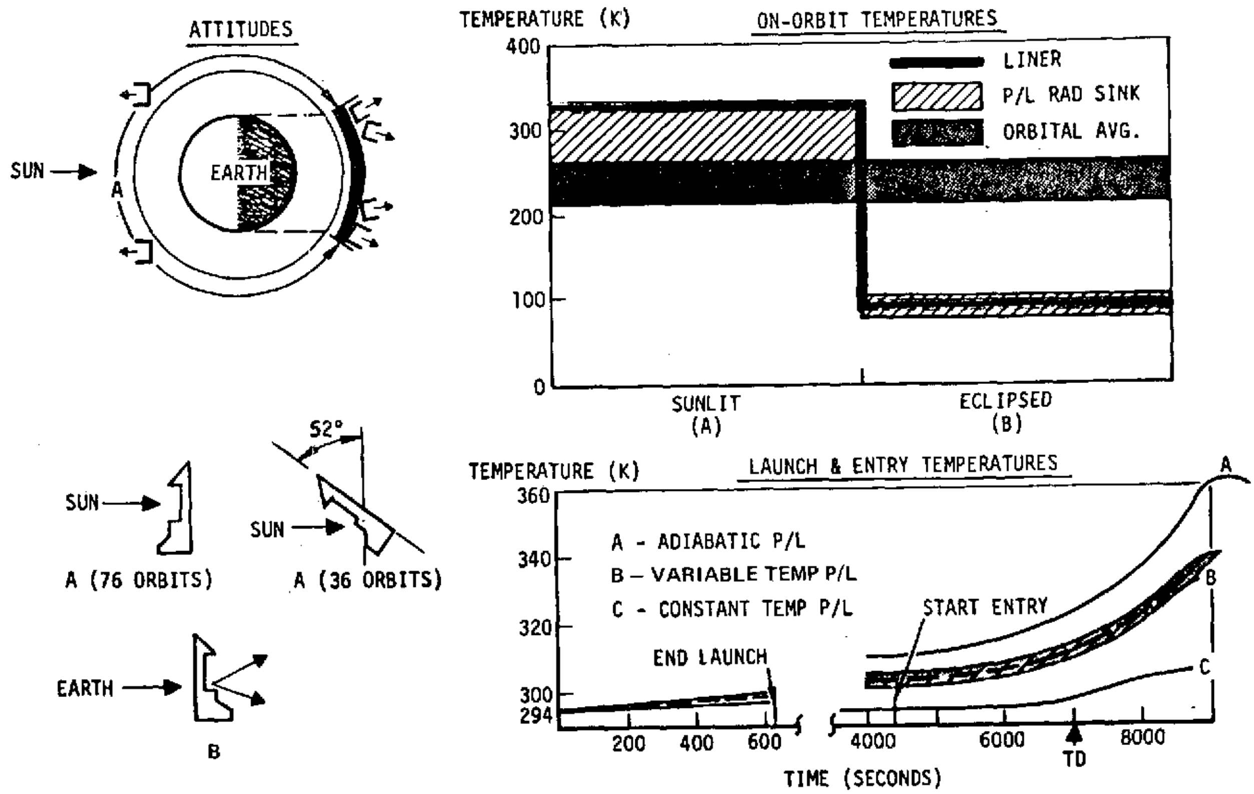
Function	Req't	Spacelab/Orbiter Capability
Mission Energy	521 KWH	890 KWH
28 VDC Unreg Power - Sustained	4474 W	7,000 W
115 VAC, 400 HZ Power	384 W	1,000 W
Peak Power	4854	12,000 W

Thermal Environment

For estimating the temperatures of the thermal environment, the orientations of the Orbiter were X-PEP (Position A) for 76 orbits or 52^{O} from X-PEP (also Position A) for 36 orbits during the sunlit portions of the orbits with the sunlight illuminating the cargo bay. During eclipses, operational orientation was X-PEP with the cargo bay facing outward to space (Position B) as shown in Figure 6. Calculated payload bay liner temperatures vary from about 330^{O}K during the sunlit portion of the orbit to approximately 100^{O}K during eclipses. Orbital average temperatures are 210^{O}K to 260^{O}K.

FIGURE 6

SHUTTLE THERMAL ENVIRONMENT



Payload bay temperatures, estimated for three different thermal configurations of payloads, are shown in Figure 6 for both the launch and return (descent) phases. The probable bounds on the bay temperatures during the ascent phase would be near nominal room temperature, i.e., 295° - 300°K. For reentry, bay temperatures for an adiabatic payload might reach 365°K about 2000 sec (approximately 1/2 hour) after touchdown. At the other extreme of a constant temperature payload, i.e., one with an infinite sink, the bay temperature would approach only about 305°K. Thus, for some configurations, the payload bay thermal environment will exceed payload temperature limits; however, payload temperatures are expected to remain within limits due to thermal capacitance and use of insulations. Certain elements, e.g., film canisters, may require localized thermal protection.

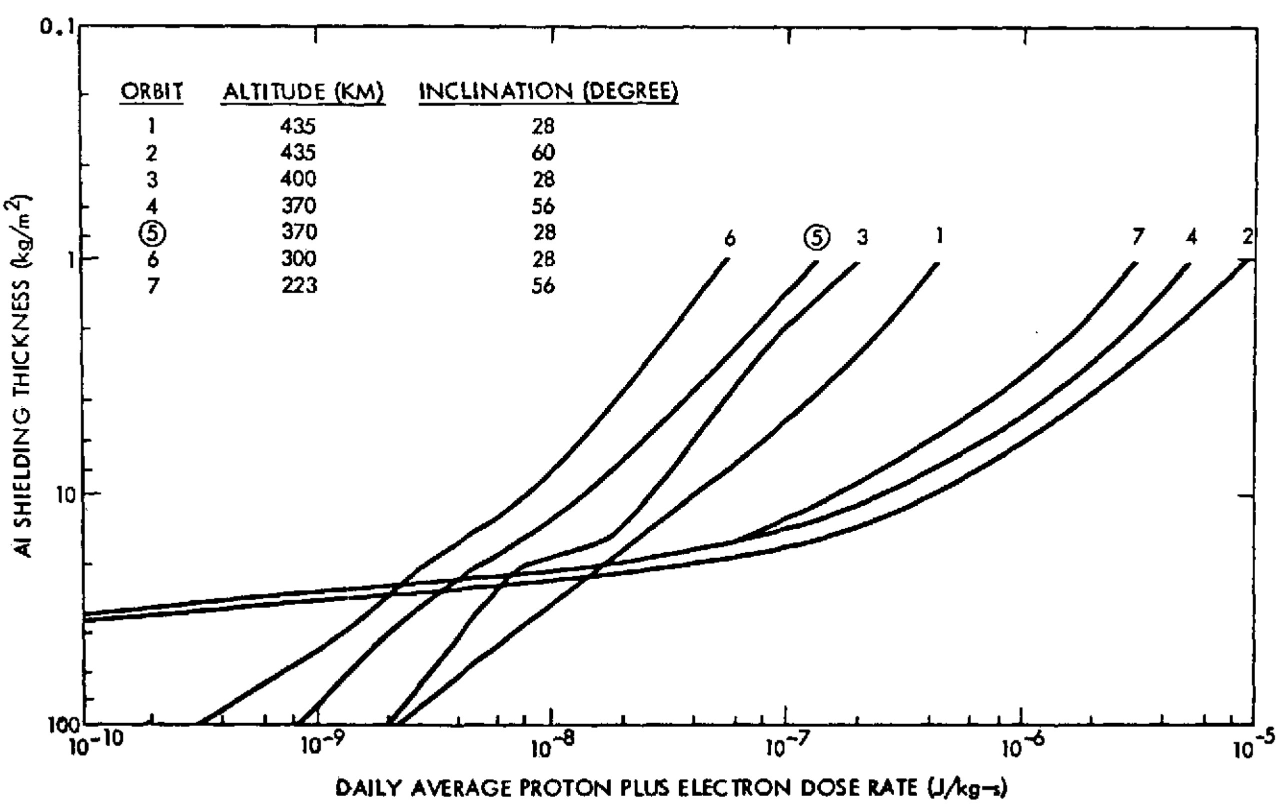
A candidate thermal control system for some of the astronomy class instruments is a thermal canister embodying a combination of active and passive control. High performance insulation and radiating areas, plus heaters and circumferential and longitudinal variable-conductance heat pipes could be used to maintain temperatures and temperature gradients within appropriate limits.

Orbit Selection

Several orbits were considered as candidates for Missions 1 and 2. The low altitude orbit, number 5 in Figure 7 at 370 Km (200 n. mi.) circular and 28° inclination, not only minimized the angle between the ecliptic and orbital planes for the time of launch, but also minimized usage of the Orbital Maneuvering System (OMS) compared to the other orbits except the 6 and 7 orbits. However, with the exceptions of orbit 6, the average dose rate for trapped radiation environment is less for a specific amount of shielding than for other orbits. The No. 6 orbit is poorer than the No. 5 orbit with respect to the duration of contact pass time.

FIGURE 7

TRAPPED RADIATION ENVIRONMENT



SMALL INSTRUMENT POINTING SYSTEM (SIPS)

C. Henrickson, Ball Brothers Research Corp.

The Small Instrument Pointing System (SIPS) will allow presently developed small instruments up to the size of ATM instruments to fly on Spacelab without extensive modifications.

The SIPS is conceptually envisioned as an adaptation of the mounting and pointing hardware which has been developed for the OSO program, with modifications enabling maximum benefit to be derived from the shuttle operational environment. Figure 1 shows SIPS with two thermal canisters.

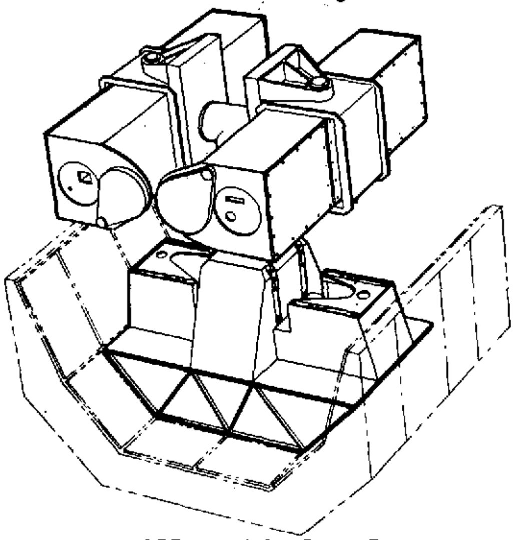


Figure 1. SIPS with Two Instruments

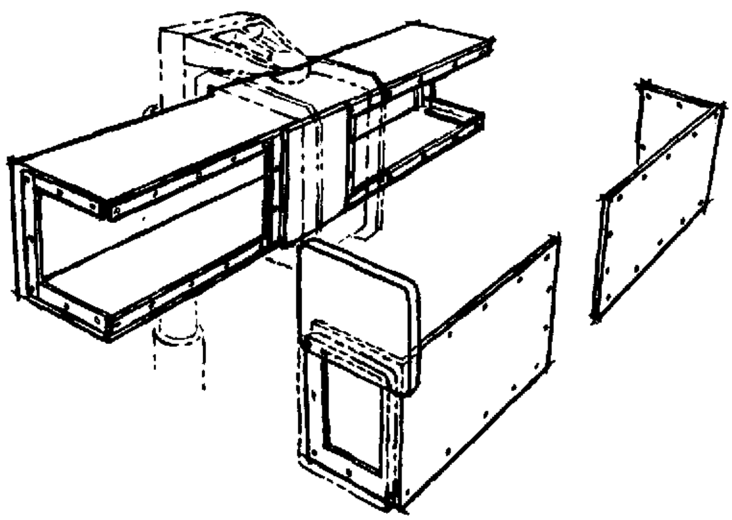


Figure 2. Cannister Concept

The most likely major characteristics of this pointing system are presented in the following paragraphs.

An instrument canister is held in a rectangular frame that is similar to the "elevation frame" that held the pointed instruments on OSO-H. This frame is supported top and bottom by trunnions that allow each about 10 degrees of right-left freedom for fine pointing. The trunnions can rotate through 90 degrees elevation to give both independent coarse elevation control and independent fine up-down pointing. The elevation drive is located at the top of a deployable pedestal. The pedestal itself can rotate to provide coarse azimuth control. By means of this pedestal, the instrument is retracted into a "cradle" during launch and landing. The system will be capable of deployment and operation in a 1g environment.

No separate roll gimbal is provided. However, when the shuttle is oriented such that observation is at the zenith, the azimuth drive becomes a coarse outer roll gimbal and full 3-axis gimbal capability is achieved. An inner roll gimbal that could provide fine roll stabilization, as well as rotation through $\pm 90^{\circ}$ for slit orientation, is under study as optional hardware for those experiments, that need roll control, such as "side lookers", polarimeters, imaging devices, and slit spectrographs.

It should be noted that this mount would usually accommodate two separate fine-pointing instruments, supported on opposite sides of the pedestal pedestal, as shown in Figure 1. The two systems would share deployment and coarse azimuth control but could pursue observations of either different or identical points lying within a strip of about 10 x 90 degrees.

The SIPS canister is sized to accommodate instruments with dimensions up to 91.4 x 91.4 x 315 cm (36 x 36 x 124 in). These dimensions will allow any instrument up to ATM size to be enclosed. The upper limit on weight handling is expected to be about 340 Kg (750 lbs). The inner roll gimbal should hold standard Aerobee payloads (38 cm in diameter) and, hopefully, Aerobee 350 payloads (56 cm in diameter).

Pointing Capability

The SIPS can operate in either of two modes. The first is an "open loop" mode in which the shuttle orbiter serves as a reference. The second is a "closed loop" mode in which sensors on or in the instrument serve as the pointing reference.

In the open loop the SIPS is pointed using information from the Orbiter's navigation systems. The gimbals are then locked with respect to the Orbiter and the pointing is done by the Orbiter. Accuracy is dependent on the inherent pointing capability of Orbiter and the distortion of the Orbiter due to the thermal variations and gradients. Accuracy will probably be limited to several degrees.

In the closed loop mode, accuracy and stability can be extremely good depending principally on the type of reference sensor(s) used. With Spacelab provided low-noise sensors (sun sensors, rate integration gyro's, or star trackers using bright stars), stability of 1 arc-sec should be possible. The absolute accuracy will depend directly on the sensor complement for a particular instrument, but should be on the order of an arc minute with a package that includes a star tracker and good rate integrating gyros.

Environmental Canister

The canister will provide protection from shuttle-borne contamination and will also facilitate instrument temperature control. Ideally, it will accommodate existing instruments without modification of their tie-down fixtures. The canister's basic structure can be in the form of a channel, as shown in Figure 2. ATM instruments (and others of that size) can be tied down to the thicker base wall (bottom of the "U") using the original non-redundant fixtures. Alternatively, if the instrument is sufficiently stiff, it may be hard attached to the base wall with small dimensions between the attachment points.

With the canister and thermal controls, the SIPS weight is 703 Kg (1550 lbs). To carry a second canister, add 290 Kg (640 lbs). The canister can be separated from the SIPS and sent to the experimenter. He can then mount his instrument to the canister, test his system and only after it is fully ready for flight will it be coupled to a SIPS. This allows for the maximum use of the SIPS while providing much flexibility to the experimenter.

Thermal Control

Thermal controls range from simple passive systems to complex active systems depending on instrument needs and environment. Simplified thermal modeling has been performed on two representative instruments. The models include the instrument and SIPS but not the thermal canister. These models added to a Shuttle bay model in development will be used to evaluate several possible canister thermal control concepts.

Note: This is only an interim report. Those areas undergoing further analysis and definition include thermal control, launch and landing restraint mechanism, engineering and operational interfaces with Shuttle, and pointing control. A final report on this work is due in June 1975.

POINTING

W. Nagel, GSFC

In addition to the SIPS, there are two other pointing systems planned. As a result of the Small Payloads Workshop, a third system is being studied.

Orbiter

The orbiter can be used as a pointing system. The inputs to the control system can be either from the navigation system or from sensors mounted on the pallet or the instrument. If the inputs of the navigation system are used for pointing, there will be the errors inherent in the navigation system plus errors due to distortions of the Shuttle from temperature gradients and mechanical distortion at the pallet. This latter source of error may lead to accuracies of no better than 2° - 4°. Using a pallet mounted star tracker and observing celestial targets, pointing accuracy improves to $\pm$ $0^{\circ}\!.35$, with 0.1° deadband.

Instrument Pointing System (IPS)

The IPS is a system being studied capable of pointing large, heavy payloads accurately. Several arrangements have been investigated for the IPS. A conventional gimbal arrangement, an inside out gimbal arrangement and a suspended pallet concept.

Any of these would be automatically controlled by the computer utilizing on-board sensors. Table 1 lists some of the requirements for IPS.

Table 1

Requirements for IPS

- Pitch and Yaw Pointing Accuracy ± 1 $\widehat{\text{sec}}$ 3σ
- Pitch and Yaw Pointing Stability ± 1 $\widehat{\text{sec}}$ 3σ
- Roll Pointing Accuracy ± 30 $\widehat{\text{sec}}$ 3σ
- Roll Pointing Stability ± 10 $\widehat{\text{sec}}$ 3σ
- Slew Rate 30 deg/min
- Gimbal Range $\pm 50^{\circ}$ Pitch and Yaw $\pm 90^{\circ}$ Roll
- Size Payload 2 M Dia x 6 M Long
- Weight 3000 KG

Tiny Instrument Pointing System (TIPS)

The Workshop brought out the need for a less sophisticated pointing system than either IPS or SIPS with weight carrying capabilities considerably reduced from either. As a result of these needs the concept of TIPS has been introduced with accuracy of 1 arc-minute and stability of 10-15 arc-seconds. It will have 3-axes and support about 100 kg.

MECHANICAL

D. Miller, GSFC

Payload Attachment Location in Payload Bay

Thirteen (13) primary payload structural attachment points are provided along the payload bay. With the exception of the aft most position, each attachment consists of three points, one on each longeron and one at the keel. The aft attachment consists of attachment points on the two longerons, but none at the keel. The attachment points in Spacelab are identified in blueprints. The allowable reaction loads which may be reacted in each direction (X,Y,Z) at each primary attachment point are shown in Figure 7-20 of Reference[1] 1.

Pallet Attachment

There are 24 hard points for payload attachment on each pallet. The hard points are ball/socket joints bolted to the pallet structure having load carrying capability of:

X direction	2910 kg
Y direction	1880 kg
Z direction	7650 kg

Figure 1 demonstrates typical use of hard points.

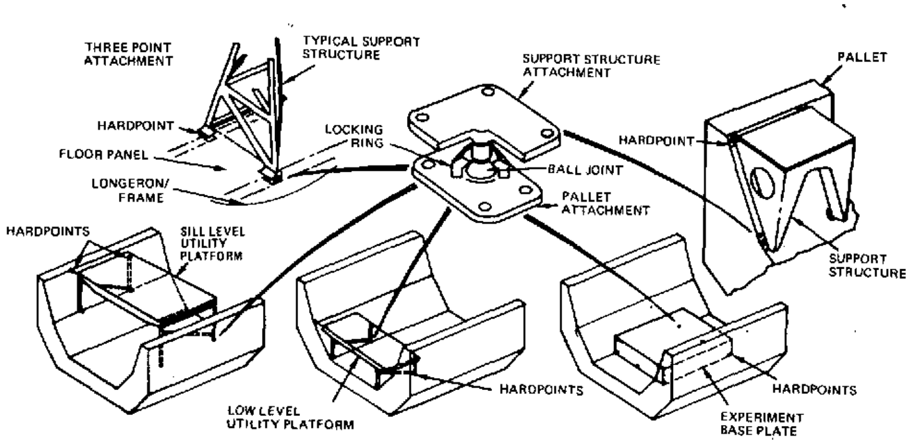


Figure 1. Typical Use Of Hard Points

1/ "Space Shuttle System Payload Accommodations", JSC 0770, Vol. XIV, Rev. C., JSC, July 3, 1974.

Pallet Description

The pallet's cross-section is U-shaped and is made of aeronautical shell-type construction. It provides hard points for mounting heavy experiments and a large panel surface area to accommodate various payload configurations. The pallets are modular (3 M nominal length) and can be flown independently or interconnected. As many as three pallets can be interconnected.

To increase the surface mounting area and particularly the viewing capability of the pallet, additional experiment utility platforms can be provided as shown on Figure 1. Two types of platforms are proposed: one 1.5 meters wide and mounted horizontally at sill level, the other 1.5 meters wide and mounted horizontally at the first frame kink (from the top) of the pallet. In both cases the platforms can be mounted between any two main frames (not end frames) whether or not the pallet segments are rigidly connected or separately suspended. The platforms are flat and consist of a grid of beams covered with honeycomb sandwich panels, in a similiar manner to the pallet. The intersections of the pallet beams provide mounting for hard points to accommodate heavy pieces of equipment while lighter experiments are attached via inserts in the snadwich panels (8 mm diameter honeycomb inserts with metric self-locking thread at any requested hole pattern).

The pallet floor has a limited load capability and precautions will be necessary to avoid damage. Pressures are limited to 50 Kg/M^2. Figure 2 shows a basic two pallet configuration with igloo, forward utility bridge and other pallet features. The igloo is a cylinder with controlled temperature and pressure (N_2 atmosphere) capable of containing the following data management and power distribution equipment:

- 3 computers
- 2 I/O units
- 1 mass memory
- 3 subsystem RAUs
- 3 experiment inverters (50, 60, and 400 Hz)
- 1 subsystem inverter
- 1 emergency inverter
- 1 power battery and bit
- 1 power control box
- 1 secondary power distribution box
- 1 caution and warning logic

The same igloo structure, although designed for subsystem installation, is offered to the user as an option for experiment-peculiar equipment installation (e.g., experiment support container). In this option, the igloo is mounted to the pallet floor.

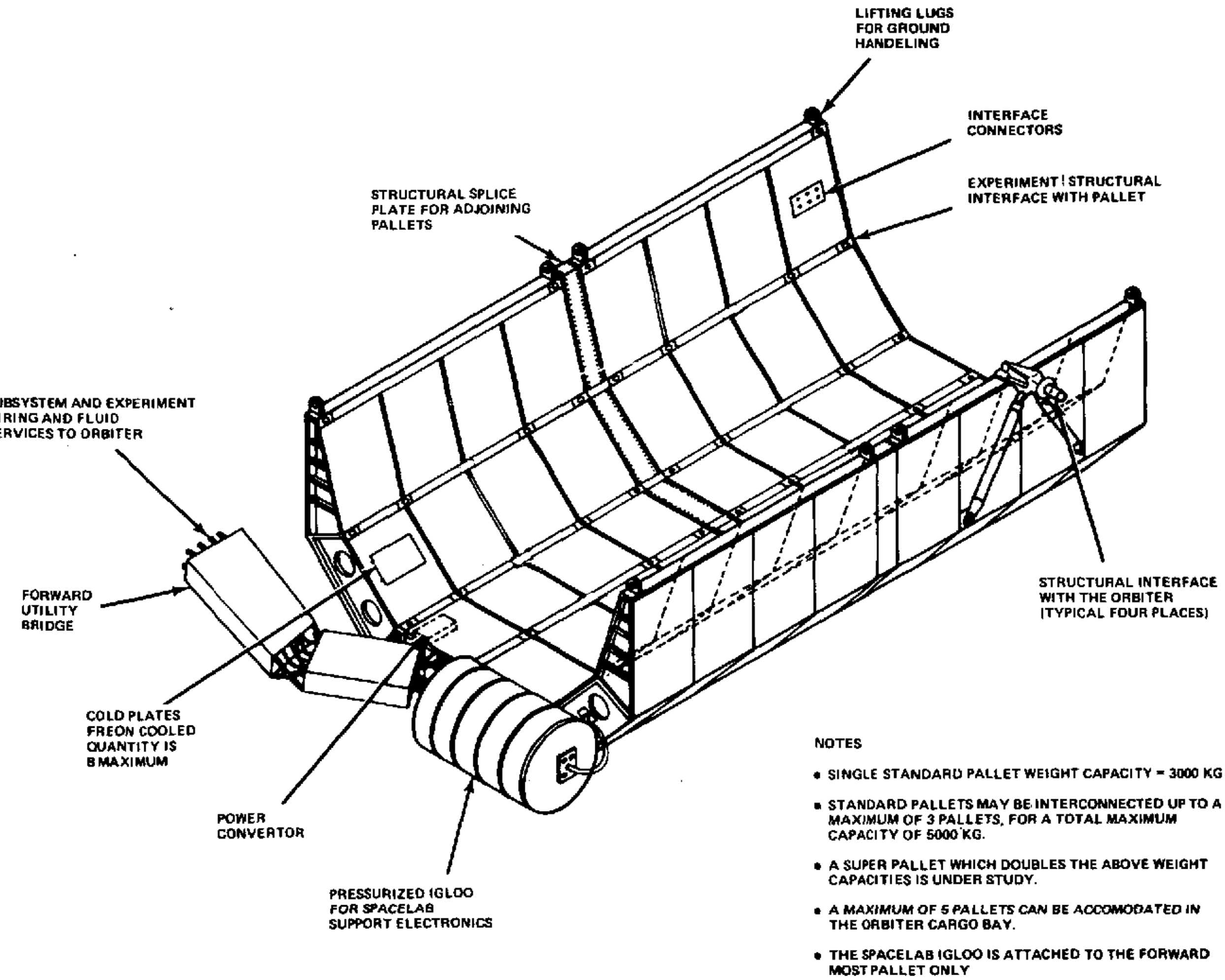


Figure 2. Two Pallet Configuration With Igloo

Pallet Dimensions

Figures 3 and 4 give two views of the pallet and include its dimensions in inches.

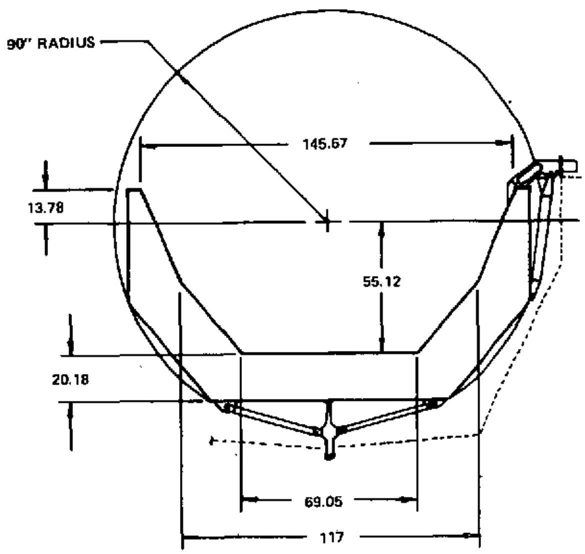


Figure 3. Pallet End View

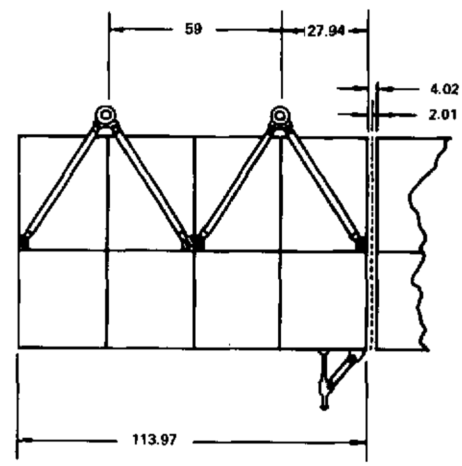


Figure 4. Pallet Side View

Pallet-Only Mode

For the astronomy missions the "Pallet-Only" mode is the only mode presently planned. This mode may consist of from one to five pallets.

THERMAL

S. Ollendorf, GSFC

The thermal problems normally encountered in space are of concern to and are being investigated by GSFC. Some of the areas being studied are listed below.

SIPS

A thermal cannister enclosing the instruments using the SIPS is being designed to allow a favorable, constant operating temperature. As design goals, it will hold instrument bulk temperatures at $20 \pm 10^{\circ}C$ dissipating between 20 and 200 Watts of power.

Pallet Mounted Equipment

A thermal analysis has shown that radiation can be trapped between the pallets and Shuttle giving rise to hot spots. Methods are being investigated to alleviate this problem.

Experiment Thermal Problems

GSFC is investigating thermal problems on specific experiments that appear to have unique thermal requirements. Figure 1 shows a typical instrument model on a pallet with nodal points.

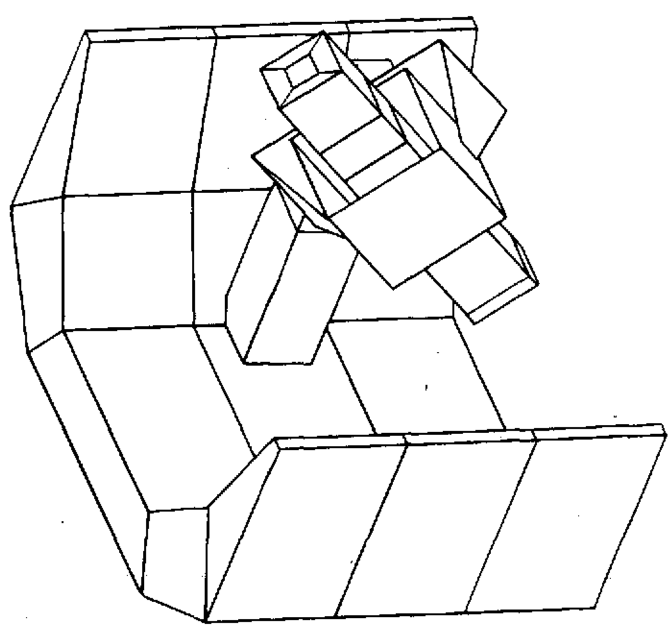


FIGURE 1. Typical Instrument Model On Pallet With Nodal Points

Thermal Model of Spacelab

A thermal model of Spacelab is being prepared by GSFC and results will be available at a later time to experimenters. This information should enable an investigator to determine the effects of the thermal environment on his equipment and properly correct for them with heaters, radiators, insulations, heat pipes, thermal covers, or whatever may be necessary. Figure 1 of Reference 1 is the overall Spacelab thermal model being used for analysis

Shuttle Environment

Figure 2 shows a typical profile of responses of payloads during the reentry and post landing phases. The upper curve shows a case where the payload rejects no heat to the walls (adiabatic). The lower curve shows the response of a payload with fixed thermal mass. Most payloads will fall within these extremes if not thermally protected.

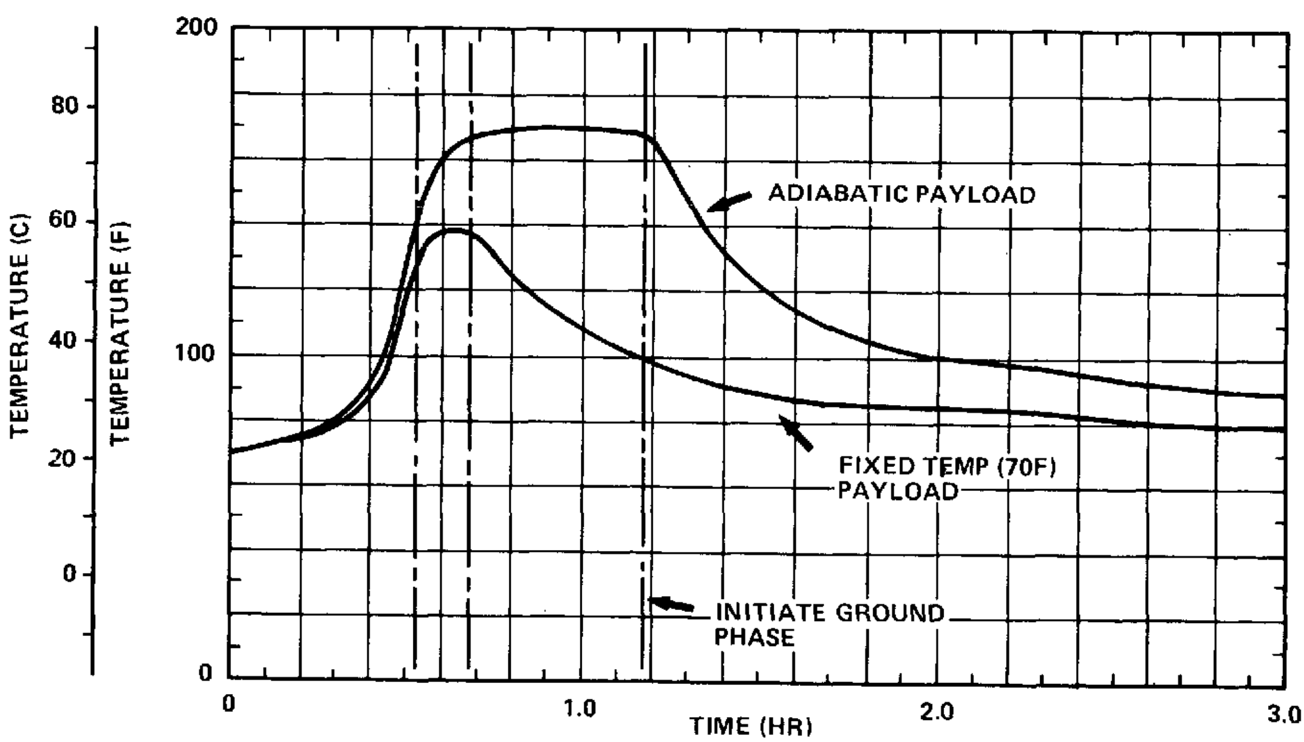


Figure 2. Typical Entry Profile of Temperature Vs. Time

1 Note: Fixed temperature payload is one which has a fixed operating temperature (70^{o}F). The curve is the temperature profile for best temperature control. Ground phase initiation corresponds to the opening of the payload bays.

1/ Thermal Design Support for the Astronomy Shuttle Payloads, Almgren, D.W., and J. T. Bartoszek; Available through GSFC-ASP Study Office.

TEST, EVALUATION AND INTEGRATION

R. Heuser, GSFC

The test and evaluation facilities for the astronomy payloads will be available at GSFC. An experimenter should be able to coordinate his tests with GSFC's Test and Evaluation (T&E) Division personnel to assure that insofar as possible, the optimum series of tests are defined to assure reliable and productive operation of the payload in-orbit.

Information Required From Experimenters

Generally, experimenters must provide adequate information to form a baseline or criterion against which the results of functional and environmental tests can be compared. The purpose of the test may be either to measure a characteristic or to evaluate performance. The detailed information required of the experimenter will vary depending on the purpose of the test and the nature of the experiment. The actual tests to be performed will be decided on a case by case basis. However, a more detailed philosophy/plan will be available in mid-1975.

Tests

Listed below are tests that may be performed at GSFC.

- Initial Magnetic Field
- Leak Detection
- Electrical Performance
- Pyrotecnic Performance
- Physical Measurements (Weight, Center of gravity, Moments of Inertia)
- Temperature and Humidity
- Vibration
- Acoustic Noise
- Shock
- Structural Loads
- Thermal Vacuum
- Antenna Pattern
- EMI

The user will supply payload-peculiar or unique hardware which may include bench test equipment and the personnel for its operation.

Not all of the tests listed may be required. However, because other experiments and man's safety are involved, stricter requirements will be placed on Shuttle payloads then on sounding rocket payloads. An experimenter may be able to demonstrate by analysis, with tighter requirement restraints, that his equipment does not require certain tests.

Integration

There are four levels or phases of integration. The first two levels (Levels IV, III) are performed at GSFC while the last two (Levels II, I) are performed at the launch site. The four levels are listed in Table I.

Table 1
Integration Levels for a Spacelab Payload

Level	Location	Activity
IV	GSFC	Install Instruments/Support Equipment on Pallet Segments
III	GSFC	Experiment Checkout & Integration
II	Launch Site	Spacelab Integration
I	Launch Site	Orbiter - Cargo Integration

From initiation of Level IV through launch is approximately 22 weeks.

This process is being reviewed from the point of view of the small payloads experimenter. Hopefully, ways will be found to reduce the lead time, minimize the time invested by the experimenter, and to make the payload accessible up to a few days before launch.

Note: The integration levels, the activities and locations are under review and are subject to change.

COMMAND AND DATA MANAGEMENT

H. McCain, GSFC

General

The Command and Data Management System (CDMS) provides a variety of services to the Spacelab payload by means of a dedicated data processor, data bus and interfacing units. These services include data acquisition, monitoring, formatting, processing, displaying, caution and warning, recording and transmission in addition to providing command and control capability for the Spacelab payloads. An additional set of identical equipment provides the same services to the Spacelab subsystems.

Figure 1 illustrates the assemblies comprising the CDMS with respect to experiments. Experiment outputs including status and scientific data are sampled by Remote Acquisition Units (RAU), converted from analog to digital form and transferred to the experiment-dedicated computer by the input/output (I/O) controller.

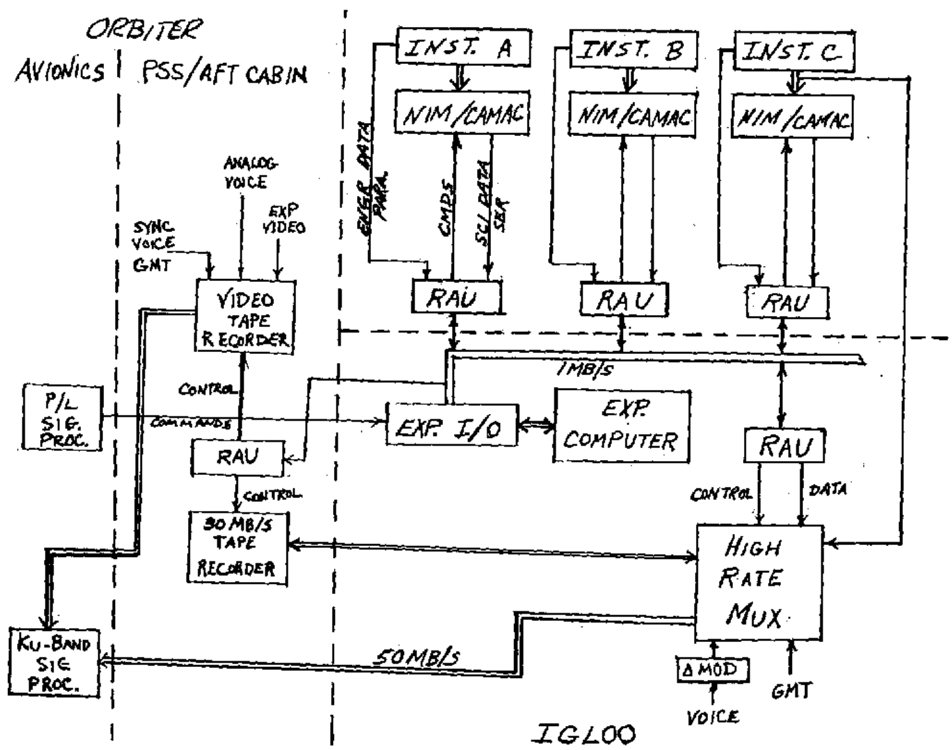


Figure 1. Assemblies Comprising CDMS

Note: GSFC is studying the use of NIM/CAMAC with power requirements and reliability suitable for Spacelab use.

RAU

The RAU can acquire both analog and digital data. The analog portion converts the signals to 8 bit resolution digital. The 32 high level inputs have a range from 0 to 5.12V while the 32 low level inputs range from 0 to $\pm$256mV. Maximum sampling frequency is 100 Hz. The inputs are single-ended with 10MΩ impedence.

The 60 digital inputs have Transistor-Tranisitor Logic (TTL) levels. The average data rate is 100 Kbps with a maximum rate of 1 Mbps for 1 msec.

High Data Rate Inputs

There are both analog and digital high frequency data inputs. Both are 75 ohm impedance and both feed into a high rate multiplexer. The analog input has bandwidth of 6 MHz. This information may go either to a 5 MHz recorder or to the downlink transmission. The digital rate is up to 50 Mbps with biphase level coding. This information can either go directly to the downlink at 50 Mbps or stored on tape at 30 Mbps.

TV Signals

TV signals generated by experiment-supplied cameras can be acquired by the Spacelab closed-circuit TV system. There is one input provided in each rack segment and on each pallet. The signal can be monitored at the Orbiter crew station or the operator console on the TV monitors or it can be transmitted by the Orbiter RF equipment to ground. For non-direct transmissions times the video signal can be recorded.

Data Processing

The CDMS provides a dedicated on-board computer for processing data which has been acquired by the experiment data bus system. The processing outputs are displayed on cathode ray tubes (CRT) and transmitted and/or delivered back to the experiments depending on the mission requirements. The computer facilities allow general processing, such as checkout, sequencing and control of experiments, data reduction, filtering, averaging, histograms, computing, etc. Application software is supplied by the experimenter.

Computer

Table 1 summarizes the characteristics and capabilities of the Spacelab computer.

Table 1. Computer Characteristics and Capabilities

Formats

Operands: 16,32 and 24+8 (floating point) bits

Instructions: 16 bits

Control Unit

Micro-programmed control unit

Control memory capacity:

1st level: 256 40-bit words
2nd level: 32 40-bit words

Number of Instructions

100 instructions including:

- Single-word (16 bits) and double-word (32 bits) call and store
- Fixed-point arithmetical operations on 16 and 32 bits, and floating-point arithmetical operations on 32 bits (24 + 8)
- Logic and comparison operations
- Shift operations
- Fixed-to-floating and floating-to-fixed conversions
- Conditional and unconditional jumps

Addressing Modes

Immediate, direct, indirect, relative to a base, indexed, relative to program counter

Number of Addressable Registers

20 by micro-instructions, of which 12 can also be addressed by instructions

Computing Speed

Single-word length (16 bits):

Add (register-to-register):	1.8 μsec
Add (register-to-memory):	2.4 μsec
Multiply:	7.5 μsec
Divide:	9.0 μsec

Double-word length (32 bits):

Add:	3.6 μsec

Floating point (32 bits = 24 + *)

Add:	9.0 μsec minimum
	17.1 μsec maximum
Multiply:	26.4 μsec minimum
	27.3 μsec maximum
Divide:	27.9 μsec minimum
	28.8 μsec maximum

Digital Input/Output

Data exchange with peripherals may be serial or parallel, depending on either of two modes of operation: programmed (controlled by the program) and channel (independent of the arithmetical unit).

Data exchange takes the following times:

<u>Serial</u>
30.9 μsec in the programmed mode
32.1 μsec in the channel mode, and at a maximum frequency of 31 K words/sec in the locked channel mode

<u>Parallel</u>
4.0 μsec in the programmed mode
1.8 μsec in the channel mode, and a maximum frequency of 555K 16-bit words/sec in the locked channel mode

The maximum number of addressable channels is:

496 on the serial bus

2,048 on the parallel bus

Memory

- Type: 18 mil ferrite cores, 3-D, 3 wire configuration
- Capacity: 39 K 16-bit words for the basic version, extendible to 64 K 16-bit words in 8K word modules
- Cycle time: 1.2 μsec

During ascent, the POCC would be in a monitor and advisory posture. On-orbit, the POCC can provide experiment data for the evaluation and control of the payload. The POCC requires real-time telemetry data and command uplink to effect evaluation and operation of the payload.

During de-orbit, the POCC will serve to assure proper power down and de-activation of the Spacelab equipment. During actual descent the equipment is assumed to be inactive and the POCC should not require any inputs during this time.

Additional On-Orbit Capabilities

The POCC can provide the following functions and capabilities:

- Decommutate, evaluate, and display payload housekeeping data.
- Provide payload operations control via a real-time command link from the POCC.
- Process quick-look experiment sensor data and display for experiment analysis and operations planning.
- Provide computational capability for payload operations planning and experiment operations control.
- Provide payload attitude determination and control by interfaces with external computer systems or with Shuttle Mission Control Center.
- Coordinate with Shuttle MCC and payload specialist for Shuttle and Spacelab payload operations.
- Interface with orbit determination systems to provide orbital data for payload operations.

Payload Operations System Overview

Figure 1 illustrates the information and communication paths between the Shuttle, free flyers, tracking stations, and operations centers, including POCC, for payload operations.

CONCLUSIONS

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The Spacelab Astronomy Small Payloads Workshop provided a useful medium of interchange between a substantial group of potential Spacelab users and the engineers responsible for the development of support subsystems required to accomodate their instruments. A major goal of the Astronomy Spacelab Payloads study is to provide a benign environment with realtively simple interfaces to which sounding rocket and balloon class payloads of the sort that now exist may be adapted at low cost. The theme of interface simplicity pervaded the Workshop discussion. If Spacelab is to provide an acceptable extension of our current sounding rocket capability, an experimenter must be able to easily integrate and de-integrate his payload, have access to it at specified times during the integration process, and operate it (or even have it fail) without interfering with other payloads. He should be able to simulate and verify payload operations at his home institution. The current philosophy of the sounding rocket program for payload accomodation should be followed in the Spacelab program if scientific viability and instrument costs per observing second are to be maintained at an attractive level.

A major subsystem requirement is a 3 - axis pointing platform with star trackers and a rate integrating gyro system available as part of the subsystem. The Small Instrument Pointing System (SIPS) concept, including thermal cannisters, is attractive with respect to its interfacing simplicity and to its possible commonality with solar physics instrumentation. The use of SIPS for UV-Optical Astronomy will require roll stabilization, accomodation of side-looking instruments (impacting both the fineness of roll stabilization and thermal cannister design), raster scanning, capability for payload evacuation or dry N_2 purge, and the accomodation of cryogenic dewars. In addition to venting provisions, the latter will require accessibility for cryogen top-off within eight hours of launch. The currently envisioned SIPS is somewhat over designed for many astronomy payloads and one might consider a smaller pointing system for 100-150 Kg payloads with stability requirements of ~ 10 arcsec. Alternatively one might mount more than one small instrument in a single thermal cannister. The possibility of deploying small payloads with current Aerobee pointing controls and retrieving them after use should be considered.

At present an overall concept for command and data handling has not been firmly established. Problems of concern are the integration and verification of software while maintaining maximum experimenter independence and self-sufficiency. Also, the relative roles of on-board payload specialists and a ground control center need to be defined. Two extreme positions with respect to the payload specialist role were expressed at the Workshop. On the one hand, command and data access to instruments through remote acquisition units (RAU's), coupled with a nearly full-time telemetry capability through TDRS might obviate the need for a payload specialist. On the other hand, observers with relatively simple instrument control and data requirements, who seek maximum interfacing simplicity, should not be required to interact with a very complex ground control center. Many participants envisioned a payload specialist performing simple operational tasks, such as power on/off collimation checks, command sequence initiation, manual film advance, performance monitoring, etc. Since it will usually not be possible to fly one payload specialist for each instrument, one will have to decide if he is willing to have his instrument operated by a payload specialist (an astronomer) who is not intimately familiar with it.

Other problems of concern to Workshop participants include the following:

- difficulties in using long light path instruments because of large scale payload bay thermal flexures and mutual interference with other instruments
- magnetic isolation requirements for electrographs and image intensifiers
- power requirements and vibration sensitivity of standardized electronics modules (NIMS, CAMACS)
- sky brightness and large column densities of light atoms and molecules introduced by orbiter vernier control-system exhaust
- protection of film from thermal "backsoaking" after re-entry
- the potential cost impact of NASA's testing and documentation requirements
- frequency of flight opportunities and choice of observing season to complete finite survey programs; total number of flight "slots" available per year.

Typical lead times for the initial development or adaptation of Spacelab rocket-class payloads range from two to three years. Therefore, NASA should begin to make payload development funds available for the initial orbiter test filghts and early Spacelab missions in 1976. To continue the involvement of the scientific community in support subsystems development, Goddard Space Flight Center will regularly conduct Small Payloads Workshops and will actively encourage dialogues between individual experimenters and the engineering group leaders involved in the ASP study. The illustrative payloads discussed at the first Workshop will be utilized for on-going ASP mission analyses and subsystems design studies.