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STS UTILIZATION STUDY EXPERIMENT ASSESSMENTS.(U)
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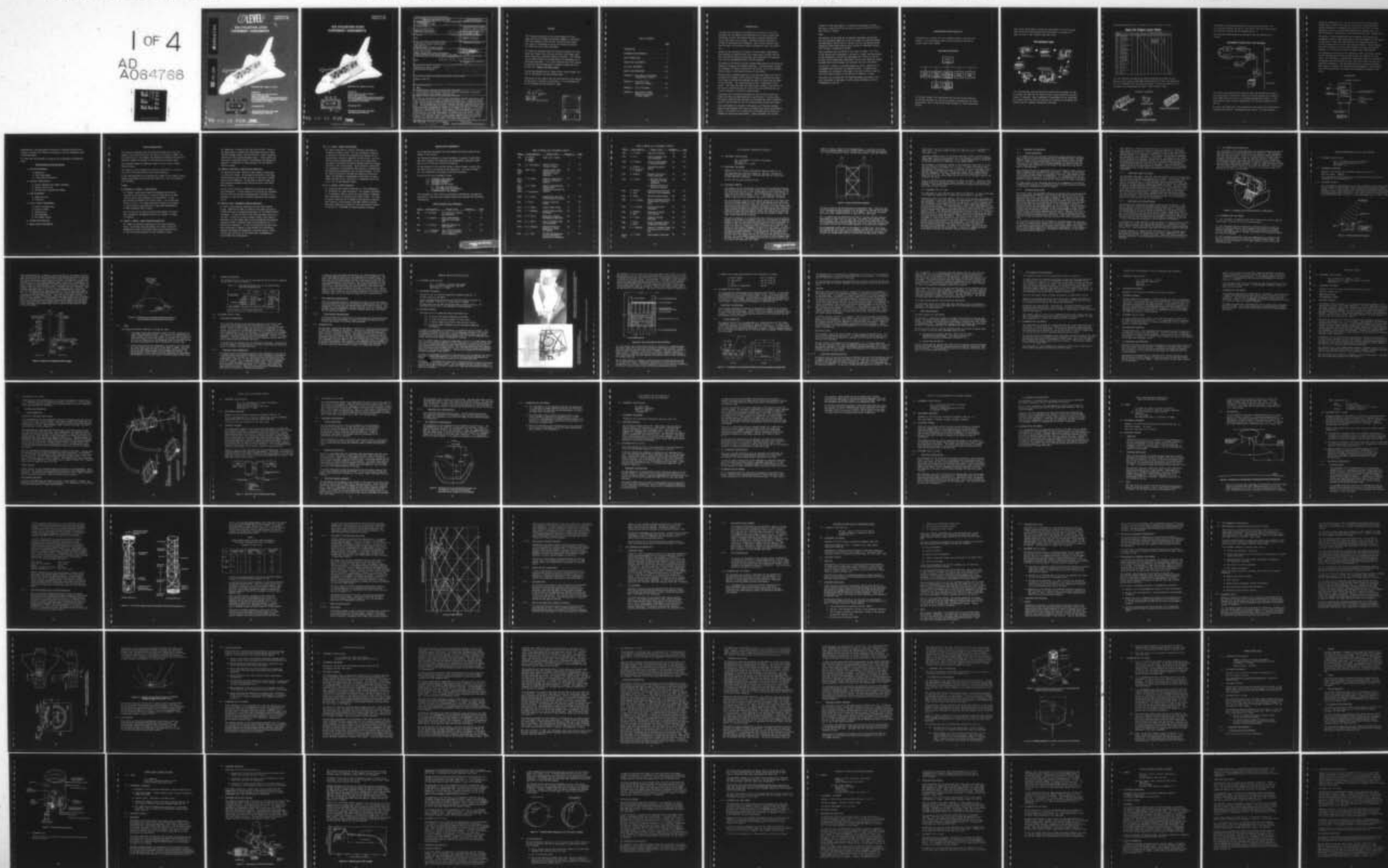

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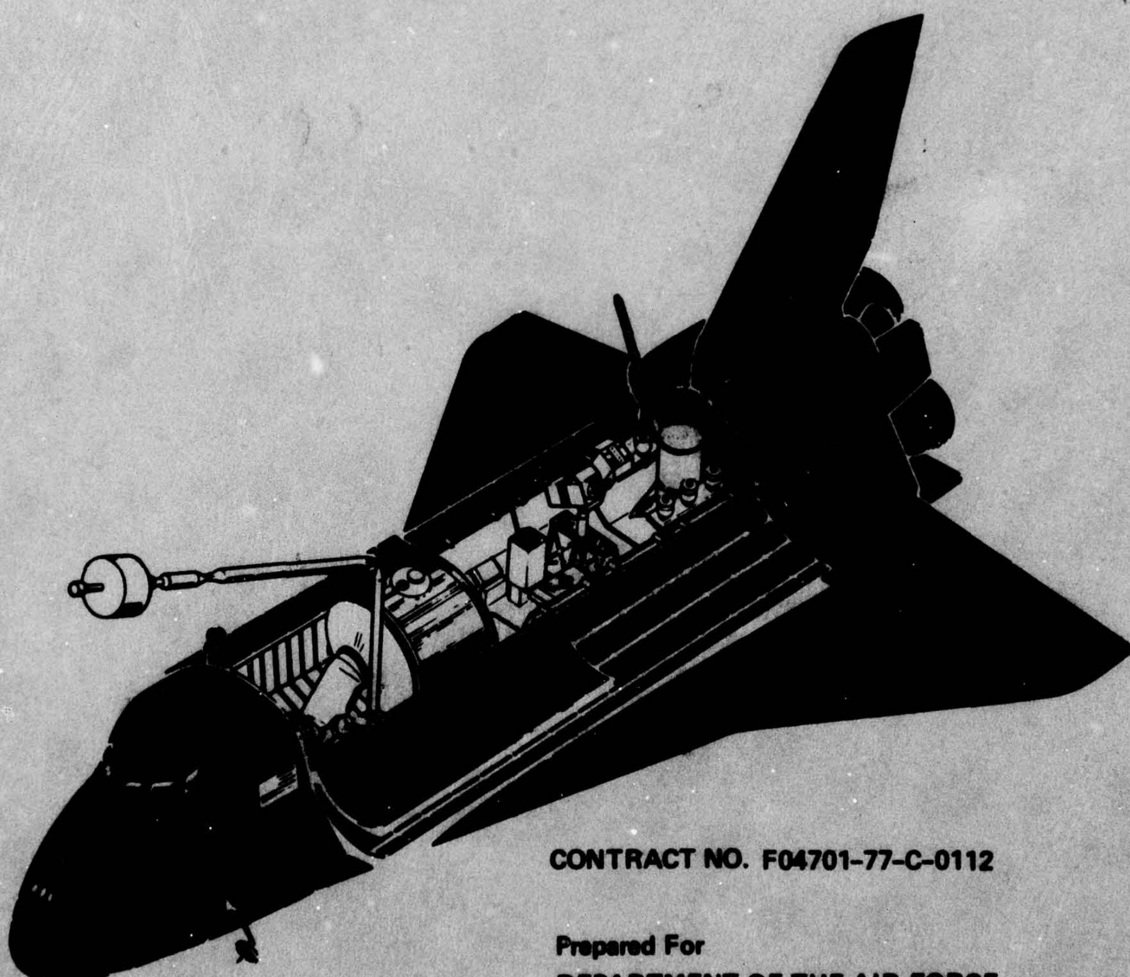
STS UTILIZATION STUDY
EXPERIMENT ASSESSMENTS



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SPACE AND MISSILES SYSTEM ORGANIZATION
P.O. BOX 92960 WORLDWAY POSTAL CENTER
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30 December 1977

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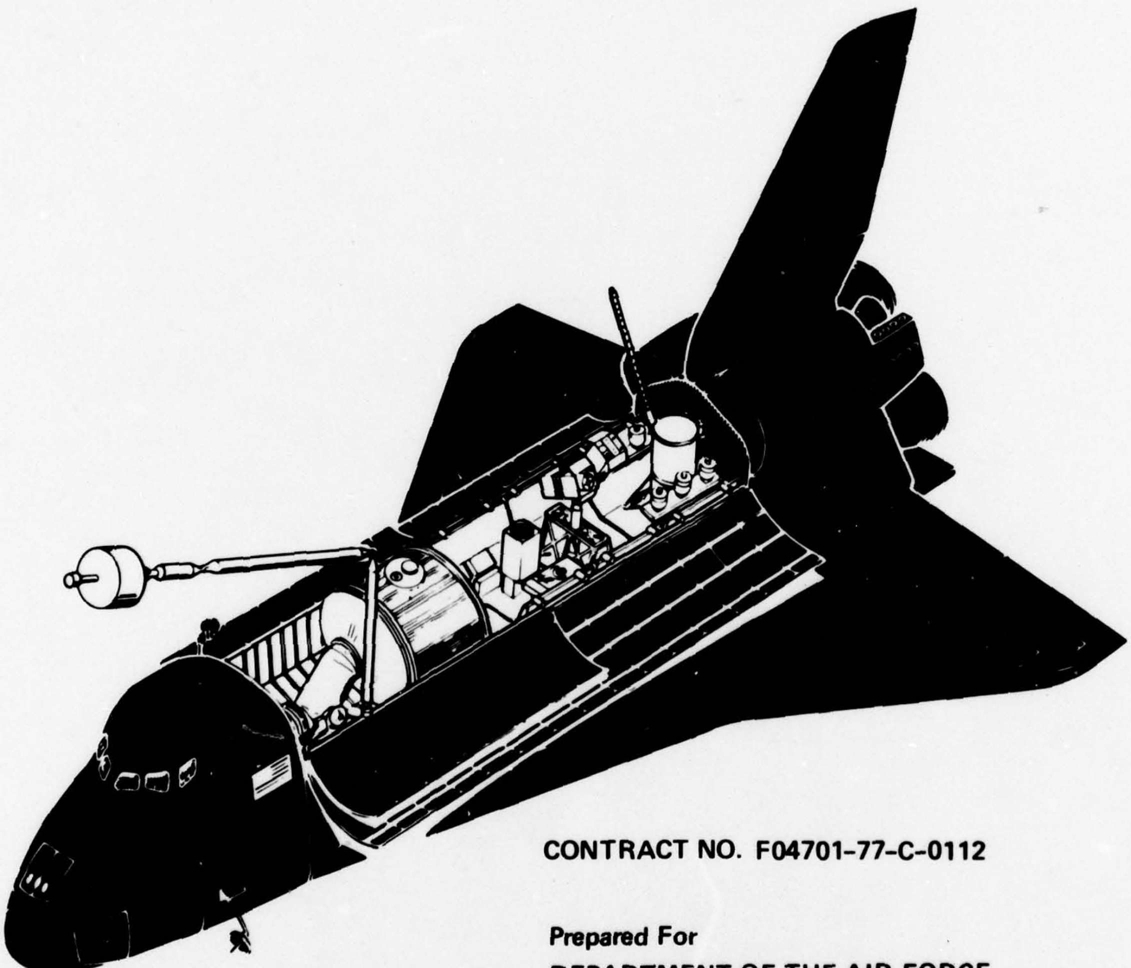

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A study was performed to identify those experiments from DoD laboratories that will be able to use the Space Transportation System (STS). Applicable experiments were assessed to determine the most effective carrier system within the STS. Design suggestions were made to improve experiment compatibility with the STS. The report describes the study, includes the experiment assessments and data on payload accommodation capability of elements of the STS. It is concluded that a considerable amount of DoD space flight experimentation can be projected for the STS flight era. Most experiments will require one of the payload carriers, now under development, to interface with the Orbiter. Many will require the use of special flight support equipment such as a pointing system. In a specific area, it was found that there is basic materials research within DoD that might benefit from space experimentation.

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FOREWORD

This study was conducted for the United States Air Force, Space and Missiles Systems Organization (SAMSO), in accordance with the Statement of Work for the "STS Utilization Study." This report is submitted as partial fulfillment of Contract No. F04701-77-C-0112, CDRL Item 004A2.

The study was conducted under the direction of Major Carl Jund, Space Test Program Plans Division, with Mr. Larry Weeks, Aerospace Corporation, providing technical direction. The findings of this report should not be construed as STP acceptance of an individual experiment. It is still required that final approval be obtained from the Department of Defense through the use of DD Form 1721, Request for Space Flight.

The TRW Study Manager was Mr. Robert Elkins, Space Systems Division of TRW Defense and Space Systems Group.

Publication of this report does not constitute Air Force approval of the report's findings or conclusions. It is published only for the exchange and stimulation of ideas.

Carl S. Jund
Major, USAF
Chief, STP Plans Division

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INTRODUCTION

The Space Test Program has performed very successfully in the past, providing the DoD community with opportunities to prove concepts and technology in space, and provide the means to qualify hardware for use on operational space systems. Dedicated boosters, containing a variety of experiments and "piggyback" rides on other boosters, have been the means for providing a space environment up until the Shuttle era.

Because of the limitations of weight, size and budget, experimenters have competed for opportunities, and in many cases, valuable time has been lost because there have been many more experiments and equipment then could be accommodated within the STP framework.

The advent of the Shuttle provides expanded opportunities to evaluate performance of hardware and develop technology. Increased weight, volume and frequency of flights, coupled with the ability to retrieve hardware, open up areas of investigation previously unavailable.

Recognizing this expanded capability to perform experiments with the STS, the Space Test Program Office directed a study to reevaluate the technical needs of the DoD and determine the means for exploiting the added utility provided. The study conducted by TRW is summarized in this report. The report describes the study scope, includes a description of the services provided by the STS, and outlines the conditions and constraints involved in utilizing a manned system.

Experiments were evaluated to determine if the STS would provide the proper test bed for experimentation or qualification. In cases where the proper conditions were provided, an assessment was made to determine which of the available STS carriers would provide the best environment.

These assessments are included in the report and illustrate the many modes and environments which can be provided by the Space Transportation System. The versatility is graphically illustrated by the various accommodation techniques demonstrated. These assessments will provide insight for other experimenters, scientists and engineers to determine ways in which the STS may be used to provide a proper test bed for their field of interest.

SCOPE

The objective of the study was to identify experiments and concepts that use the added capability of the STS, assess the interface between the experiments, the STS and its various carriers, and develop design suggestions and/or modifications which provide an integrated approach with the Space Transportation System.

These assessments were performed at two levels of depth. The first, called "medium level assessments," provides an insight into the purpose of the experiments, outlines the assessments for "flying" on the STS, provides design suggestions, operational restrictions, describes support equipment which may be needed, and considers the cost implications.

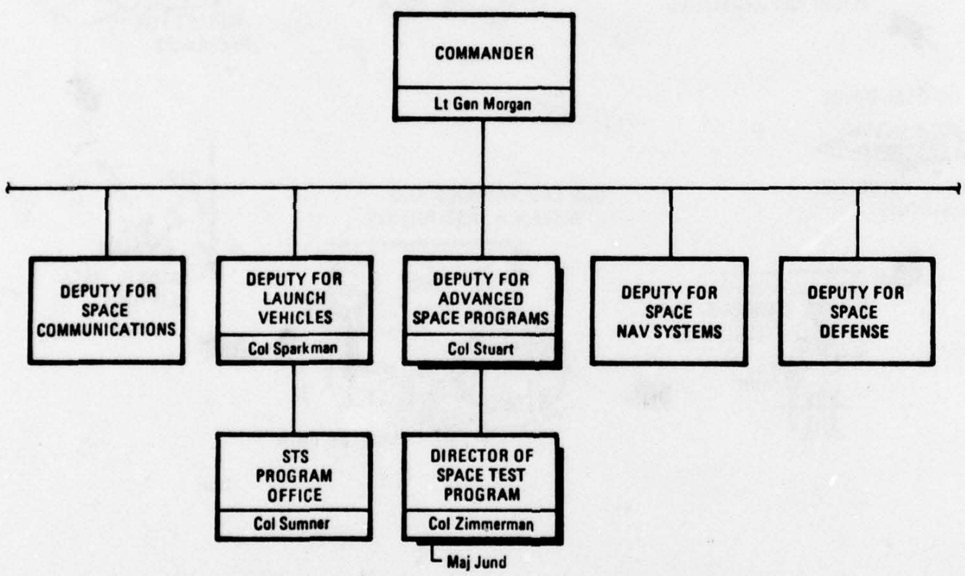
If practical, a sketch is provided showing an artist's concept of an arrangement which satisfies the experiment needs. 32 medium level assessments are contained in the report, and cover a diverse area, covering from particle physics to processing of materials in space.

The report also includes 100 "low level"assessments providing a very brief evaluation of the manner in which these experiments could be accommodated by the STS.

BACKGROUND AND METHODOLOGY

The Directorate of Space Test, more commonly known as the Space Test Program, is an organization within the Air Force Space and Missile Systems Organization (SAMSO).

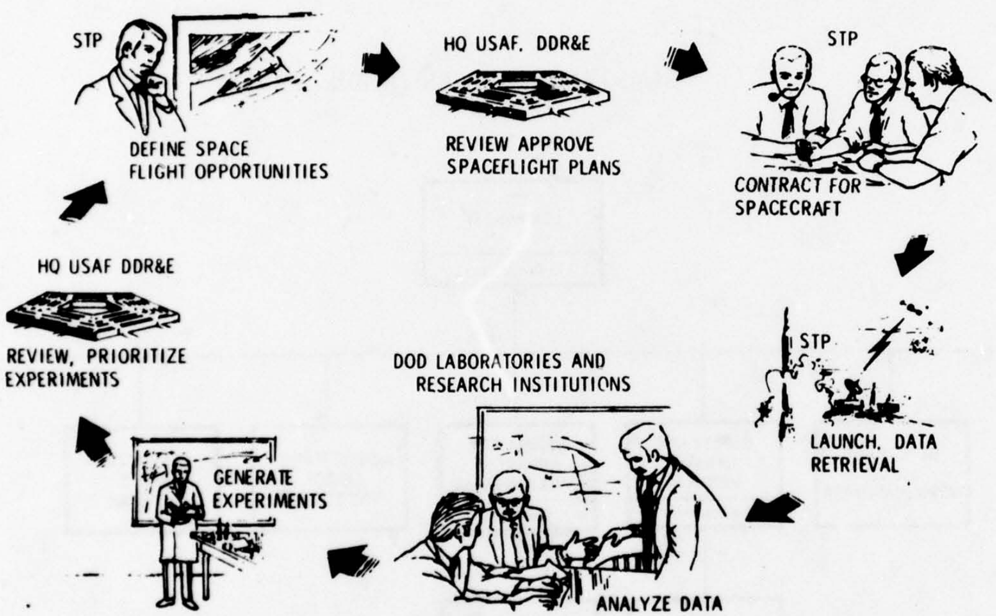
SAMSO AND STP ORGANIZATION



For the past decade, this organization has serviced the DoD community by providing the means for evaluating concepts and techniques for space application as well as qualifying flight hardware for future operational systems.

The role of STP has been to review the requirements for "flying" equipment, define the flight opportunities and proceed to implement the needs as shown in the following Experiment Cycle.

The Experiment Cycle



This flow indicates that the Space Test Program Office arranges for and/or provides the funds for the launch vehicle, launch operations and upper stages, if required. When a complement of experiments can be integrated on a single satellite, STP will contract for a spacecraft and manage that program from its inception through data retrieval.

The following table provides a history of STP for the past 10 years.

Space Test Program Launch History

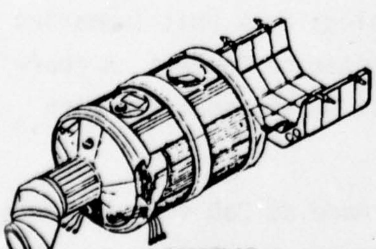
SPACE FLIGHT NUMBER	BOOSTER/UPPER STAGE	PAYLOADS
P67-1	THOR/BURNER II	2
S67-3	THORAD/AGENA	3
S68-2	NASA THORAD/AGENA	1
P68-1	SLV-3A/BURNER II	10
P67-2	TITAN IIIC	4
P69-1	ATLAS F/TRI OVI	4
S69-2	NASA THORAD/AGENA	1
S68-3	TITAN IIIC	3
S69-4	THORAD/AGENA	1
S70-3	NASA THORAD/AGENA	1
S70-4	THOR/BURNER II	1
P70-1	THOR/BURNER II	2
P70-2	ATLAS F/DUAL OVI	9
P71-2	THORAD/AGENA	4
S71-3	THORAD/AGENA	2
S71-3	THORAD/AGENA	2
P72-1	ATLAS F/BURNER II	5
S73-7	LPC-509	1
P73-3	ATLAS F/2 SOLIDS	1
S73-5	TE-M-479	3
P72-2	ATLAS F/AKM	5
S73-6	TE-M-516	7
P74-1	TITAN IIIC/SOLIDS	4
P76-5	SCOUT	1
S74-2	TE-M-521	7

LAUNCH DATE CALENDAR YEAR: 67, 68, 69, 70, 71, 72, 73, 74, 75, 76

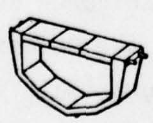
TOTAL FLOWN	25
ASCENT FAILURES	3
S/C FAILURES	0
NOT KNOWN	1

With the potential provided by the Space Transportation System, and the plan to utilize space on both NASA and DoD Shuttle missions, the opportunity to "fly" experiments will be vastly increased. The carriers which will be considered for DoD payloads are shown in the following figure.

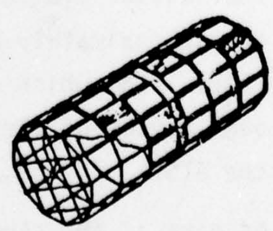
POSSIBLE CARRIERS



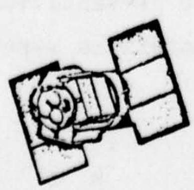
SPACELAB



STANDARD TEST RACK



LONG DURATION EXPOSURE FACILITY

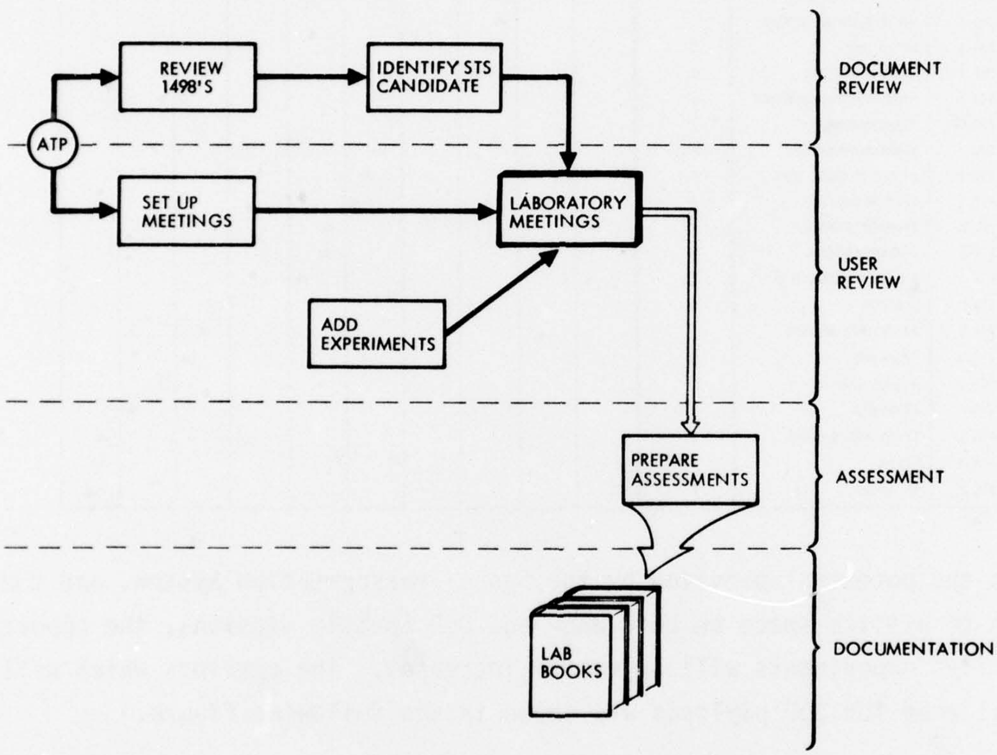


MULTIMISSION MODULAR SPACECRAFT

In addition to integrating payloads with other primary missions, STP will continue to pursue the use of "free flyers" when that method appears to be the best solution for the proposed mission.

The study which TRW performed is outlined in the flow diagram which follows.

ASSESSMENT PHASE STUDY FLOW DIAGRAM



The initial evaluation of DoD's technical needs was obtained by screening the approximately 20,000 active Research and Technology Work Unit Summaries (Form 1498) which are on file in the Defense Documentation Center. A thorough review of these produced approximately 100 that were compatible with the STS.

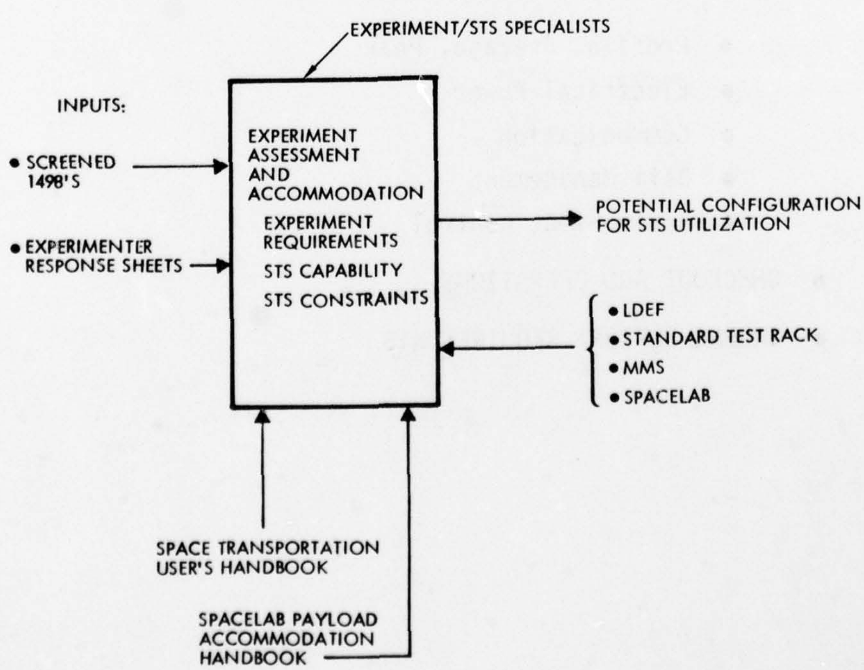
As part of the study task, five presentations were made at DoD installations throughout the country. Presentations were made at: The Naval Research

Laboratory, Washington, D.C., 12 July 1977; Air Force Technical Applications Center, Patrick AFB, Florida, 14 July 1977; Space and Missile Systems Organization, Los Angeles, California, 28 July 1977; Air Force Geophysics Laboratory, Hanscom AFB, Massachusetts, 2 August 1977; Air Force Aeronautical Laboratories, Wright-Patterson AFB, Ohio, 4 August 1977.

The purpose of these presentations was to acquaint the technical and management personnel within these agencies with the capability of the Space Transportation System, the added potential for experimentation which the STS provides and outline the study being performed. The participants were also encouraged to discuss their endeavors with the study team to determine if STS could be utilized to enhance their investigations.

The information obtained from the screening of 1498's and inputs received from the presentations formed the body of the source material for the study. This information, coupled with information relating to the Shuttle, the Spacelab and other payload carriers, was used by TRW to perform the experiment assessments. The experiment specialists, who have performed many studies on Spacelab and Shuttle payload accommodation, then analyzed the selected experiments and produced the assessments contained in the study.

ASSESSMENT TASK



Where possible, the Experiment/STS specialists contacted the DoD investigators to expand the information available so that the assessment would be more meaningful.

The items that were analyzed in preparing the assessments are enumerated below:

EXPERIMENT/APPLICATION DEFINITION

- MISSION REQUIREMENTS
  - Objectives
  - Orbit Requirements
  - Flight Dates and Duration
- PHYSICAL CHARACTERISTICS
  - Mission Equipment and Support Equipment
  - Weight, Volume, Size
  - Configuration, Deployed and Stowed
- CREW REQUIREMENTS
  - Number and Skill
  - Timeliness
- UTILITY SUPPORT REQUIREMENTS
  - Profile, Average, Peak
  - Electrical Power
  - Communication
  - Data Management
  - Environment Control
- CHECKOUT AND OPERATIONS
- GROUND SUPPORT REQUIREMENTS

STUDY ORGANIZATION

The study was performed under the direction of Major Carl Jund, the manager of the planning activity for Space Test Programs at SAMSO. He was assisted by Mr. Larry Weeks, The Aerospace Corporation, who provided technical support and guidance to TRW during the assessment phase.

The study was managed for TRW by Mr. Robert Elkins, with Mr. Thomas Hanes as deputy.

The specialists who performed the technical assessments are listed below, with a brief description of their qualifications.

In the event questions arise regarding the content of this report, please contact either Major Carl Jund at (213) 643-1121, or Mr. Larry Weeks at (213) 648-6236.

RESUMES

Dr. Nathaniel L. Sanders - Lead Scientist

Dr. Sanders' experience includes the management and planning relating to the performance of scientific experiments on spacecraft as well as participation as an experimenter. He has participated in numerous STS related studies. He has been with TRW for 17 years. His recent experience includes an assignment as the Assistant Project Manager for Experiment Accommodation on AMPS (Spacelab) Phase B Study.

Prior to that, he was the Assistant Project Manager for Experiment Integration, and Magnetic Control for Pioneers 6 through 11 (Jupiter).

Mr. Robert L. Hammel - Space Processing Specialist

Mr. Hammel has extensive experience in the Space Processing field. He has been the study manager for a Phase B study for NASA/Marshall Space Flight Center dealing with the definition of Space Processing payloads for early Spacelab flights.

Mr. Hammel was in charge of the TRW study for MSFC, "Concepts and Requirements for Materials Science Manufacturing in Space Payload Equipment Study," and the follow-on, "Space Processing Application (SPA) Payload Equipment Study." These studies concentrated on conceptual design of SPA payloads and engineering analysis of integration of these payloads into the Shuttle/Spacelab system.

Dr. Robert F. Doolittle - Space Physics Specialist

Dr. Doolittle has been involved in most aspects of space physics during his career. He has been on the staff at San Diego State University and has done research in the area of charged particle track detectors. He has been in charge of many company sponsored programs having wide application in the space physics field.

He was a staff scientist on HEAO working primarily on experiment integration. In this capacity Dr. Doolittle was thoroughly familiar with all electrical and mechanical interfaces of experiments, as well as their scientific objectives and characteristics.

Dr. Robert L. Wax - Ionospheric Physics Specialist

Dr. Wax has had extensive experience with the Space Shuttle system. Beginning in 1966, he worked on the second revision of he NASA Blue Book of Candidate Experiments for the Manned Orbiting Laboratory. He also did work on the final Blue Book version during 1970. In 1971-72, he participated at NOAA in the study of experiments for the Plasma Physics and Environmental Perturbation Laboratory (PPEPL) in Boulder under the chairmanships of J. R. McAfee and W. Bernstein. In 1973, he worked with the Martin Marietta Corporation in Denver to help produce the "Preliminary Concepts from Woods Hole Atmospheric and Space Physics," which involved the fitting of the 1973 Woods Hole recommendations into an AMPS-like configuration.

Mr. T. E. Hanes - Deputy Study Manager

Mr. Hanes joined TRW as a Special Consultant following retirement from NASA in 1975. His last assignment at NASA was as Director of Skylab Office administering closeout of the program after successful completion of the mission. He assisted the Skylab Program Manager, as one of six second-level assistants, from Skylab preliminary program definition through the entire life of the program. He was primarily responsible for integration of approximately 200 scientific, technological and applications experiments into the Skylab program. At TRW, Mr. Hanes has worked on the NASA Cost Reduction Alternative Study, the Atmospheric, Magnetospheric and Plasmas in Space (AMPS) payload, and as a specialist in procedural matters for all of our STS and Spacelab studies.

Dr. G. T. Inouye - Senior Scientist

Dr. Inouye has had extensive experience in the accommodation of instruments for space experiments. His academic background is in ionospheric physics and he has participated as magnetometer experimenter on spacecraft and rocket flights. His areas of special expertise are in magnetics and spacecraft charging. Most recently, he has worked on the AMPS (Shuttle) Payload Definition Study and on spacecraft charging problems relating to the DSCS II, FLTSATCOM, and TDRSS spacecraft programs.

MEDIUM LEVEL ASSESSMENTS

The 32 detailed assessments that were prepared during the study are contained in this section.

The information presented in these assessments is general in some areas and covers concepts and integration and accommodation techniques rather than specific design and interface information.

All the material in this section follows a similar format to assure that the same criteria was applied to each experiment. The level of detail will vary depending on the depth of existing information.

The outline for each assessment follows:

- 1.0 EXPERIMENT IDENTIFICATION
- 2.0 REQUIREMENT BACKGROUND
- 3.0 EXPERIMENT APPROACH
- 4.0 ASSESSMENT FOR STS FLIGHT
  - 4.1 Experiment Considerations
  - 4.2 STP Integration Considerations
- 5.0 RECOMMENDATION(S) AND REMARK(S)

The assessments in this section are grouped by Laboratories, and agencies, essentially in the order that response sheets were received from interested investigators.

INDEX OF MEDIUM LEVEL ASSESSMENTS

INDEX OF MEDIUM LEVEL ASSESSMENTS (CONT'D)

INDEX OF MEDIUM LEVEL ASSESSMENTS (CONT'D)

FAR ULTRAVIOLET IMAGING AND PHOTOMETRY

1.0 EXPERIMENT IDENTIFICATION

Dr. G. Carruthers, Principal Investigator
NRL Code 7123
Washington, D. C. 20375

2.0 REQUIREMENT BACKGROUND

Response to STS Utilization Presentation (Response Sheet No. 2)

This experiment was submitted to NASA for Spacelab 1 through STP, related work under Project No. RR 034-06, Work Unit No. A01-63.101, at Naval Research Laboratory.

DD Form 1721 has been submitted to STP.

3.0 EXPERIMENT APPROACH

The intention of the investigator is to measure the ultraviolet emission from diffuse sources such as the airglow, aurora, interplanetary, stellar and interstellar sources, as well as man-made emissions from chemical releases and from rocket exhausts. The instrumentation is able to form an image of the area being viewed on film. The wavelength region from 105 nm to 200 nm will be covered by two nearly identical instruments.

The instrumentation consists of two Schmidt cameras which use image intensifiers to produce images on film. One of the cameras covers the range from 105 nm to 160 nm and the other the range from 123 nm to 200 nm. The Schmidt telescopes form the UV image on the face of the image intensifiers, the UV produces photoelectric emission, the photoelectrons pass through an image preserving electron multiplier section which employs both high electrostatic voltages and high magnetic fields, and finally the intensified beam of electrons impacts on the film to form the permanent image of the scene being viewed by the telescope.

The plan of operation is to decide before the flight on a set of possible targets to be viewed by the cameras, but also to be prepared to change the program in order to view temporary targets such as aurora or man-made emissions if they are within the viewing capability of the instruments. A total of as many as 3 targets per orbit is planned with as many as 8 exposures for each target throughout the duration of the flight. In order to facilitate the viewing of different targets, it is desirable that the instruments be mounted somewhere so that the maximum viewing range can be provided consistent with costs. A crew member will be responsible for the pointing of the cameras and for the initiation and setting of the exposures. Only data needed to determine the status of the instruments need be viewed by the crew. Housekeeping data and the scientific data on the film can be retrieved after the flight.

Figure 3-1 shows a sketch of the instrumentation. It consists of two cylindrical telescopes about 20cm in diameter and 61cm long connected together by a rigid structure which serves to keep them aligned in the same direction.

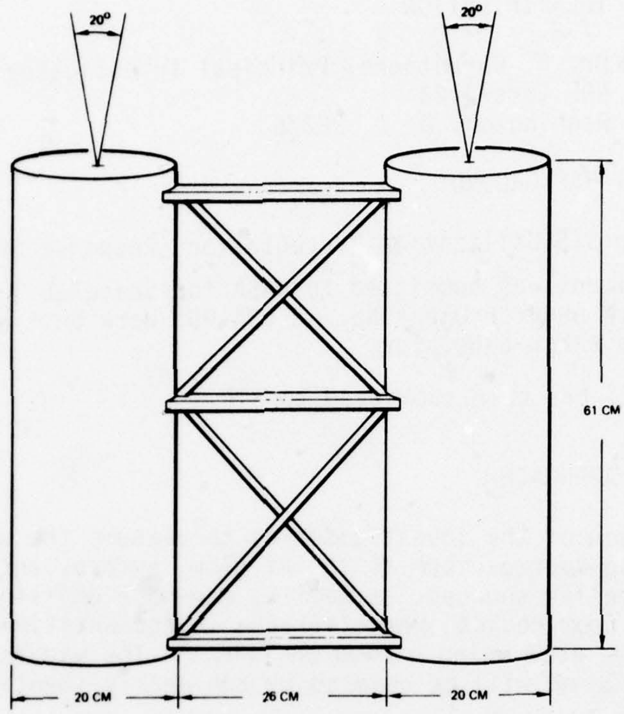


Figure 3-1. Sketch of the Schmidt Telescopes

The center lines of the two telescopes are separated by 46cm. Electrical power and housekeeping data connectors will be supplied as required for STS mounting. The total instrument weight is 50 kilograms. Each of the telescopes must have a clear viewing field of at least 20° full cone angle.

The instruments must be pointed with at least 2° accuracy and the stability requirements are that drifts of less than 1 arc minute occur during a 100 second exposure time. The instrument must be held 3-axis stabilized in inertial coordinates during stellar exposures, and 3-axis stabilized in earth view coordinates during earth fixed target exposures or during horizon exposures.

The average power used by each of the cameras is 5 watts with a peak value during advancing of 10 watts. The instruments take 28 volts. It is unlikely that both of the peak rates would be simultaneous so that for the entire instrument the average power would be 10 watts with the peak being 15 watts.

Only low data rates (less than 100 bps) are required for the housekeeping and status data. The major scientific data is on the film that is recovered after the flight.

Instrument commands consist of selecting wavelength filters, choosing exposure times, power on-off, high voltage on-off, film exposure initiate, film advance, and possibly temperature regulation. All of these commands would be committed by the crew from the aft flight deck or the Spacelab module.

The instrument is designed to operate in the temperature range from 10° to 30°C and to be stored in the range from 0° to 30°C. It cannot be turned on until the pressure drops to 10^{-5} Torr. It is somewhat sensitive to energetic particle radiation but is designed to operate in orbit for at least 7 days without the need for additional shielding. The optical surfaces can be contaminated by dust and condensates. The image intensifiers produce large magnetic fields which fall off as the radius cubed, but are still at the 100 gauss level on the surface of the boxes.

There is a definite reflight potential for these instruments. The optics would probably have to be cleaned and perhaps reworked, and the film would have to be replaced. These Schmidt cameras are very similar to cameras that have already been flown on rocket flights.

4.0 ASSESSMENT FOR STS FLIGHT

This experiment is well qualified for flight on the Shuttle. It must be performed in the space environment, and it needs the active control of the crew.

There are 4 areas of possible concern. The first is the pointing requirements of the instrument, in particular the stability requirement of 1 arc minute for a 100 second duration. The Orbiter stability is given as 0.1°, so that it appears that this instrument must be mounted on some sort of flight support equipment which will be discussed below. The second possible problem is the stray magnetic field produced by the instrument. It will be necessary to take care that this instrument is not positioned near another instrument that would suffer from the high stray magnetic fields. The third problem involves the recovery of the film. During the descent portion of the flight the temperatures inside the cargo bay can be quite high, and it may be found necessary to require that an EVA be performed in order to remove the film before the reentry. The final problem is that if the instrument is fixed to the orbiter structure without an active pointing system, a very large fraction of the orbit time would be taken up by the special pointing requirements of this experiment. As a worst case with 24 100-second exposures per orbit about 44% of the orbit time would be taken up by this experiment. This means that the orbiter would not be able to change attitude during these exposures. The use of a separate pointing system would alleviate this problem. See the discussion below.

4.1 Experiment Considerations

4.1.1 Design Suggestions

It is suggested that the principal investigator consider placing his instrument on some sort of pointing platform such as the NASA/GSFC Small Instrument Pointing System (SIPS) or the NASA/LeRC Annular Suspension Pointing Systems (ASPS). Either of these systems seem to be capable of meeting his accuracy and stability requirements with the added advantage that they alleviate the flight operational constraints on the orbiter's attitude during the performance of the experiment.

While STS/Spacelab have no requirement limits on DC magnetic fields, there are many instruments that are sensitive to such fields. Care must be taken when placing these Schmidt cameras in a payload so that their stray magnetic fields do not adversely affect other instrumentation. As a suggestion only, the investigator should consider if it is possible to do some shielding of the stray magnetic fields so that there would be less interaction with other instruments.

All other aspects of the instrument design can be accommodated by the Shuttle Transportation System or by the Spacelab system, and there should be no problem in placing this experiment on some STS payload.

4.1.2 Operation Restrictions

During the flight, the instrument should not be operated until the pressure and dust contamination environment of the orbiter cargo bay have fallen to their nominal flight values. A crew member is needed to check on the status of the experiment, and to initiate the exposure sequences. The amount of crew time devoted to these operations would be from 5 to 15 minutes per orbit and either the mission specialist or the payload specialist would be able to do the operations since they are relatively simple. If the instrument is placed on a pointing platform, the crew member would have to be familiar with its operation so that he could point the instrument at the required target. There is a possibility that the pointing directions could be preprogrammed and done nearly automaticallyor alternately commanded from the ground. However, there will be target of opportunity events such as chemical releases and rockets for which the pointing must be adjusted during the flight. These events would have lead times of at least one orbit so that the crew activities could be planned around them.

In the present configuration the instrument is evacuated before the launch and the cover is released after reaching orbit altitude by a squib. The investigator is considering using a motor on the cover so that before reentry the optical system can be recovered and saved from possible contamination. Since contamination is likely during reentry, the investigator should decide on a cost basis if the motor would be effective.

The instrument is capable of operating only in a vacuum of less than 10^{-5} Torr, and so there are restrictions on the ground operations. The investigator wishes to have access to the instrument as close to the launch date as possible because it is necessary to expose some of the flight film to calibration sources in his laboratory. This requires that the entire instrument be transported to the laboratory for calibration. Subsequently, the instrument is returned to the payload. Because of the aging effects on the film, it is desired to do this calibration at the last possible moment before flight. The investigator would like to perform the calibration a few days before flight, but could tolerate times up to two weeks. These times appear to be feasible given the STS ground operations schedules, however, the investigator would probably have to make special coordination efforts with the Leval II/I integrators.

4.1.3 Experiment Support Equipment

The experiment needs power, some data lines and probably a pointing system. These services can be provided by an appropriate STS flight. There may be a need for a special panel on the aft flight deck for the control of the instrument and also for a status display. If the experiment flies on a Spacelab flight, the controls and displays can be accomplished through the CDMS if the instrument is designed to be RAU compatible, which apparently it has been. The major piece of support equipment would be the pointing system. This instrument is of such a size that it will easily fit inside a SIPS canister since it occupies only about one-quarter of the canister's viewing area and 1/12th of the volume. The size of the instrument also is compatible with the LeRC Annular Suspension Pointing System. With either the SIPS or the ASPS, the instrument would have to be flown on a Spacelab pallet and would be able to use the Spacelab CDMS for its small data and control requirements.

4.1.4 Experiment Costs Considerations

The following cost considerations should be made by the investigator: 1) The use of a motor to recover the instrument during reentry; 2) the use of some pointing platform in order to obtain his required stability vs. a change in the requirement, 3) the cost of an EVA to retrieve his film before reentry vs. the possible damage to the film during the high reentry temperatures, 4) the use of high data rate counting electro-optical systems instead of the film system since the Spacelab CDMS should be able to handle high digital data rates. The extra costs of such a system would be balanced by a reduction of the risk of losing film data during reentry, or for an EVA for film retrieval.

The alternate to an EVA retrieval of the film packs is a thorough thermal design which takes into account the reentry heating profile. It seems quite likely that a thermal canister could provide the required transient thermal protection for the film. A cost trade would then reveal the most cost-effective technique.

A thermal enclosure is planned for the SIPS which utilizes a variable conductance heat pipe system. During reentry, the system is commanded "off" and will provide a good deal of thermal isolation. Should this system provide the necessary isolation, it would undoubtedly prove to be the most cost-effective technique.

4.2 STP Integration Considerations

This instrument is well enough developed, and compatible enough with the STS that there should be little problem with its integration. The instrument should be able to fly early in the STS program and was indeed proposed for the first Spacelab flight. Since it is a continuation of rocket flight instruments, there should be nc need for extensive testing or simulations. Assuming that some pointing system like the SIPS will be an integral part of the Spacelab system, there should be no problem integrating this instrument with the flight support equipment. Figure 4-1 shows a conceptual layout with the Schmidt cameras mounted in one of the SIPS canisters. This instrument should account for a very small fraction of the orbiter launch cost because of its small size and weight.

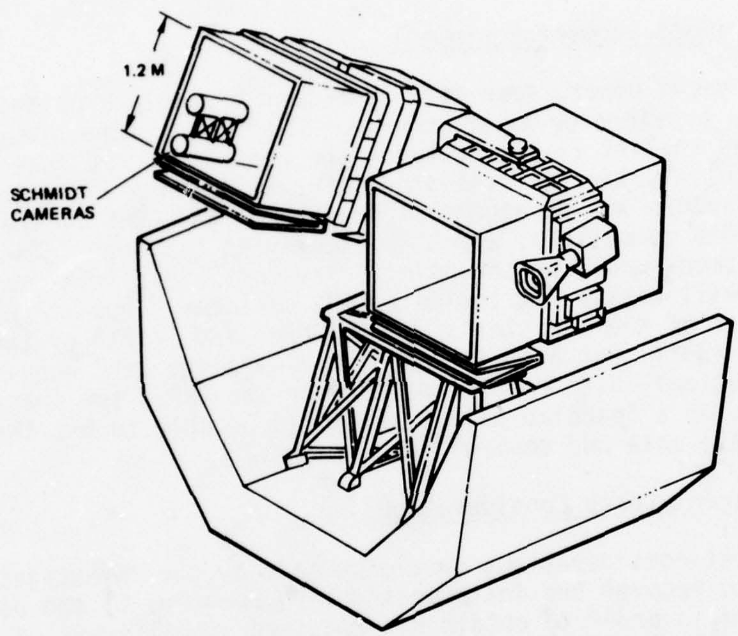


Figure 4-1. Conceptual Layout of Schmidt Cameras in a SIPS Cannister

5.0 RECOMMENDATIONS AND REMARKS

1) This instrument is compatible with the STS and must be flown in space to obtain data, it should fly on an STS sortie mission.

2) This instrument should be mounted on a pointing platform in order to obtain its required stability, and to generate less impact on the orbiter attitude requirements. Any of the available platforms would be suitable for this instrument because its weight is not great and its pointing requirements are not stringent.

3) This instrument would most easily be accommodated on a Spacelab flight where it could be monitored and controlled through the CDMS, however, the instrument could be designed to fly on other payload carriers such as the Standard Test Rack.

RADIO INTERFEROMETER SATELLITE LINK EXPERIMENT

1.0 EXPERIMENT IDENTIFICATION

Dr. S. H. Knowles, Principal Investigator
Naval Research Laboratory Code 7132
Washington, D.C.

2.0 REQUIREMENT BACKGROUND

Response to STS Utilization Presentation (Response Sheet No. 4)

Work Unit Number: DN620165

"Very Long Baseline Interferometry with a Realtime Satellite Data Link" by Knowles, S. H. and Waltman, W. B.

3.0 EXPERIMENT APPROACH

Very long baseline interferometry in the radio astronomy study of cosmic radio sources provides resolving power which is unattainable with smaller antenna systems. Fig. 3-1 shows a typical radio astronomy interferometer setup to investigate a cosmic radio noise source. The signals picked up by the two spaced receiving antennas have a difference in the time-of-arrival of characteristic noise waveform envelopes as well as in the instantaneous phase of the rf signal.

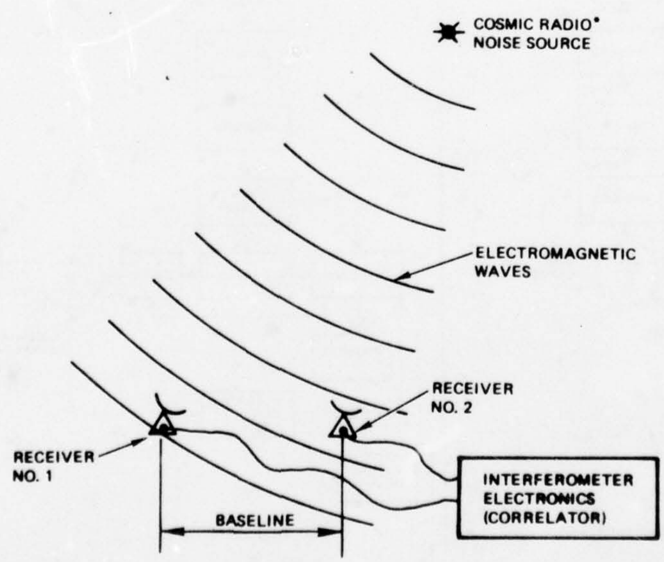


Figure 3-1. Radio Astronomy Interferometer

These time differences, as measured, are proportional to the length of the baseline separating the two receiving antennas, and therefore the angular resolving power of the system also increases at the length of the baseline. In the past, the signals received by spaced receivers have been correlated by recording the received signal on video (wide bandwidth) tape at each site. Subsequently, the taped signals are correlated after the tapes are made available at a single location. There are many drawbacks to this technique which are minimized or eliminated if the two signals are available at a common location in realtime. The proposed experiment shown in Fig. 3-2, has been implemented on a short-term basis via the Communications Technology Satellite (CTS) in the manner shown in Fig. 3-3. The proposed experiment would utilize a geostationary satellite for a continuing radio astronomy program. Time sharing with other functions is permissible if, perhaps, 48 hours per month were made available for this experiment on a long-term basis. An improvement of this experiment would be the provision for the two-way transmission of a phase-stable common local oscillator signal which would permit the implementation of phase-coherent interferometry. In this technique, the hydrogen maser clock at one of the two receiver sites would not be needed.

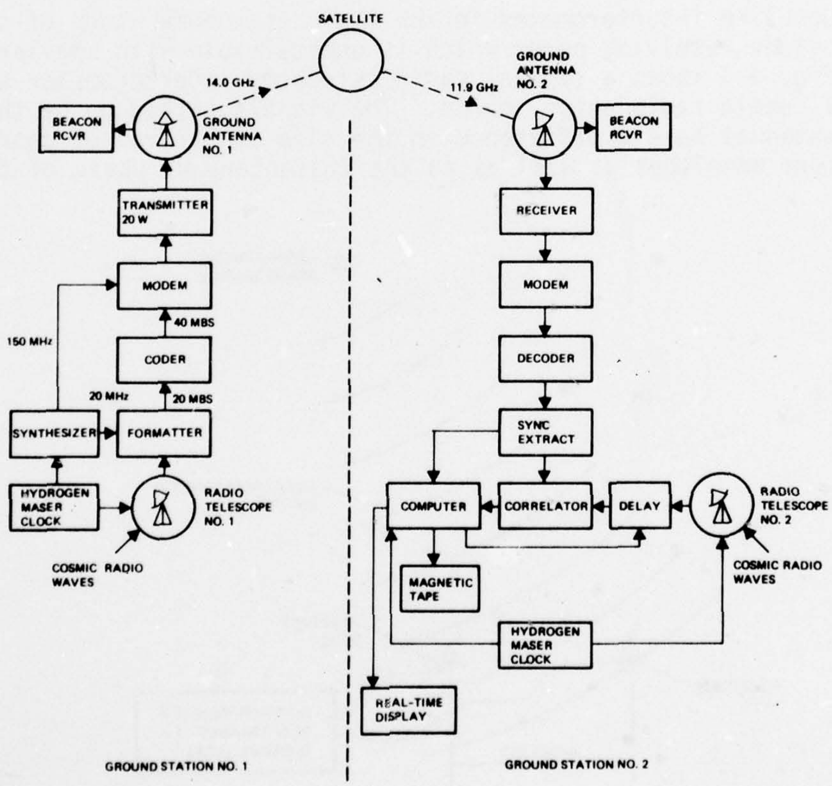


Figure 3-2. Satellite-Link Interferometer Block Diagram

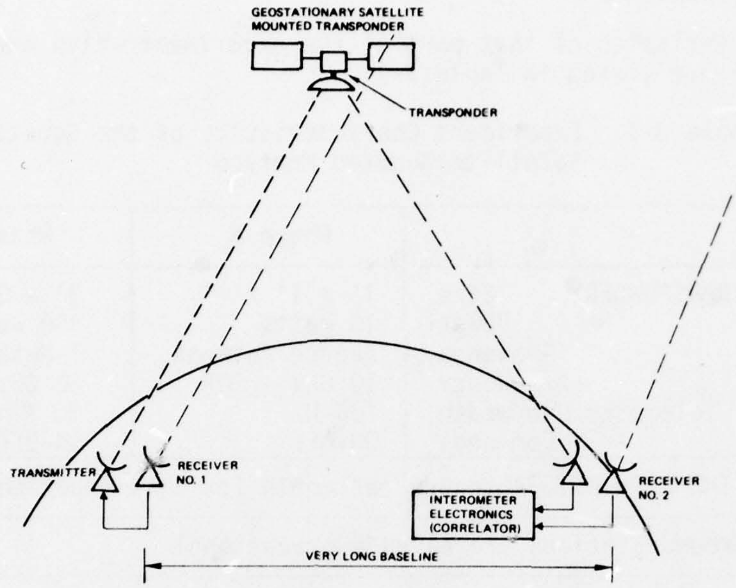


Figure 3-3. Proposed Very Long Baseline Interferometer Experiment Using a Geostationary Satellite Mounted Transponder

3.1 Plan

The proposed experiment, therefore, includes two steps:

a. A low power phase stable reference signal ($\sim$10 GHz) transmission between widely separated stations. This phase of the experiment provides a common local oscillator signal for the two stations which are recording a cosmic radio noise source. The signals are recorded locally at each station on a wide bandwidth video tape recorder and the two tapes are processed for time correlations at a later time.

b. In the second phase of this experiment, the capability to retransmit the video signal from one station to the other is added. This eliminates the requirement to tape record at each station and permits the correlation, interferometry to be performed in realtime. The wide bandwidth, 50 MHz, implies that the transponder be high-powered (20-200 watts) as in the CTS system to minimize ground station costs.

3.2 Design Configurations

The characteristics of that part of the experiment which would be supported by the STP are listed in Table 3-1.

Table 3-1. Experiment Characteristics of the Geostationary Satellite Mounted Portion

	Phase A	Phase B
TRANSPONDER: Size	1' x 1' x 1'	3' x 3' x 3'
Power	10 watts	300 watts
Antenna	Beacon Antenna	1 Meter Dish
Frequency	10 GHz ± 20%	10 GHz ± 20%
Telemetry Bandwidth	100 Hz	50 MHz
Commands	ON-OFF	ON-OFF
TIME SHARING: 48 hours per month for this experiment		
Ground Stations are already operational		

4.0 ASSESSMENT FOR STP FLIGHT

4.1 Experiment Considerations

This experiment depends on the ability of the STP to provide the capability for putting experiments into geosynchronous orbit. This experiment should be added to the list of those which do require this capability. As far as the STP supported portion of this experiment is concerned, the hardware has already been developed as evidenced by the short-term implementation via the CTS spacecraft. On CTS, because it is provided with a 200 watt TWT system, both the wideband experiment, Phase B and the narrowband experiment, Phase A, have been implemented. For STP, it is more feasible to initially implement Phase A, and to expand to Phase B when the higher powered capability becomes available.

The operational requirements for this experiment are minimal, requiring only an on-off command. The experimenter is willing to time-share his satellite usage to a minimum of 48 hours per month.

4.1.1 Experiment Design Suggestions

The phase-coherent interferometer aspect of this experiment in the form of a roundtrip White Sands-Shuttle-TDRSS-White Sands propagation experiment of signal phase stability might be a good preparatory experiment to get the experimenter's "feet wet" in terms of getting a working relationship going on the STP program. Such an experiment would require minimal equipment, which is already available, be set up at White Sands. That location is only required to eliminate Washington, D.C. to White Sands ground link phase instability errors. Even this restriction may be eliminated with some sort of calibration or compensation technique.

A second possible experiment configuration, the Shuttle-TDRSS link with its 50 MHz bandwidth capability, was suggested to the experimenter. The White Sands-TDRSS ground station location would be a limiting factor in this case, and the experimenter felt that it was better to wait for the direct geosynchronous link capability. One of the requirements for this experiment is that the signal levels at the receiver site as well as the signal levels transmitted from the other station be sufficiently large that the required signal bandwidth of 50 MHz is achieved. All of the antennas, at the transmitter, on the satellite and at the receiver, must be properly sized in terms of the available transmitters and receivers—their available power levels and effective signal-to-noise characteristics.

4.2 STP Integration Considerations

Since this experiment requires a geostationary satellite, the STP integration considerations relative to the Shuttle are those having to do with the launching of the satellite into geostationary orbit from the Shuttle. The integration of the transponder into the satellite itself is a process which has been already implemented in numerous unmanned spacecraft, and should pose no exceptional problem.

4.2.1 Miscellaneous Considerations

The delivery/lead time, preparations including advanced testing/simulations are minimal since the hardware is readily available and because the preliminary experiments are currently being performed

5.0 RECOMMENDATIONS

The pre-Phase A experiment described in Section 4.1.1 could be implemented with no additional equipment on the Shuttle. The only STP requirement is an allocation of access to realtime telemetry in the White Sands-Shuttle-TDRSS-White Sands link for a few hours at a time. The purpose of this would be mainly to set up a working relationship with the STP program as a precursor to the main experiment objective, but may also have some engineering or scientific fallout. This experiment should be added to the list of those requiring a STP capability to provide geosynchronous orbit payloads. The experiment related equipment to be placed on the satellite are state-of-the-art hardware that have been flown on the CTS spacecraft and presents no unusual STP integration problems.

GAMMA-RAY MONITOR FOR SPACE SHUTTLE

1.0 EXPERIMENT IDENTIFICATION

Dr. J. D. Kurfess, Principal Investigator
Naval Research Laboratories, Code 7128
Washington, D. C. 20375

2.0 REQUIREMENT BACKGROUND

Response to STS Utilization Presentation (Response Sheet No. 5)

DD 1498, Accession #DN 480107

A Hard X-Ray/Gamma-Ray Observatory for the Space Shuttle (Proposal to NASA, 3 December 1973) by J. D. Kurfess and H. Friedman.

DN 480107 describes the NRL balloon and rocket research programs. The experiment assessed here is an outgrowth of those programs.

3.0 EXPERIMENT APPROACH

The objectives of the Gamma-Ray Monitor experiment are:

(a) to observe the Shuttle Gamma-Ray environment.

(b) to study high energy processes in solar flares.

(c) to monitor gamma-ray emissions from celestial sources.

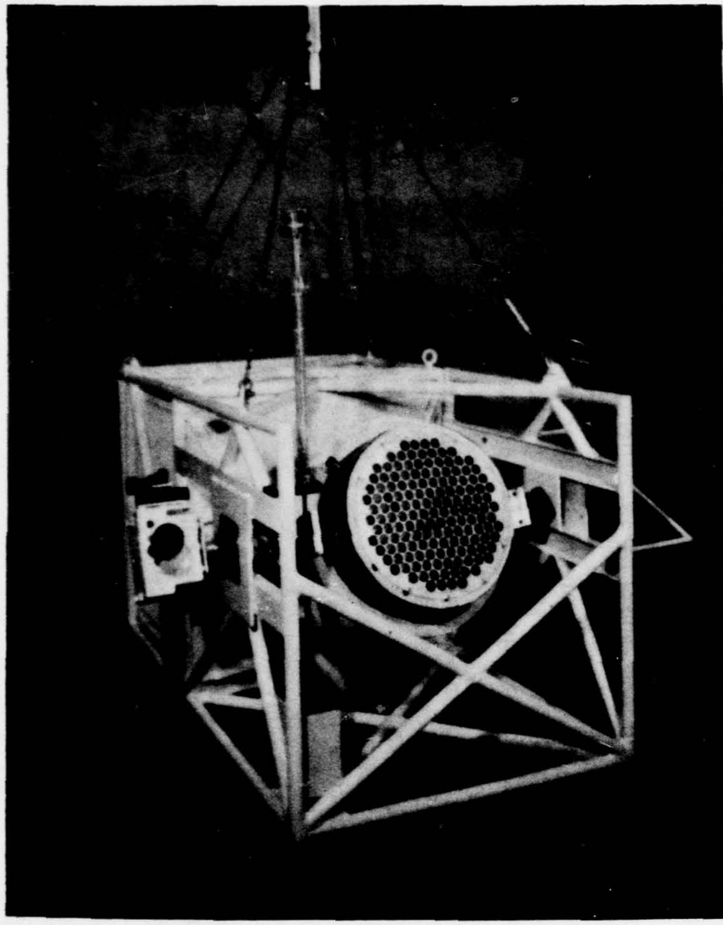
(d) to monitor upper atmosphere and the near-earth gamma-ray environment.

The gamma-ray environment on the Shuttle is expected to be quite severe. This background environment arises from: (1) the background produced in the orbiter and detector material by spallation products generated by cosmic rays and by trapped protons in the South Atlantic Anomaly, and: (2) the time-dependent background resulting from changes in geomagnetic latitude.

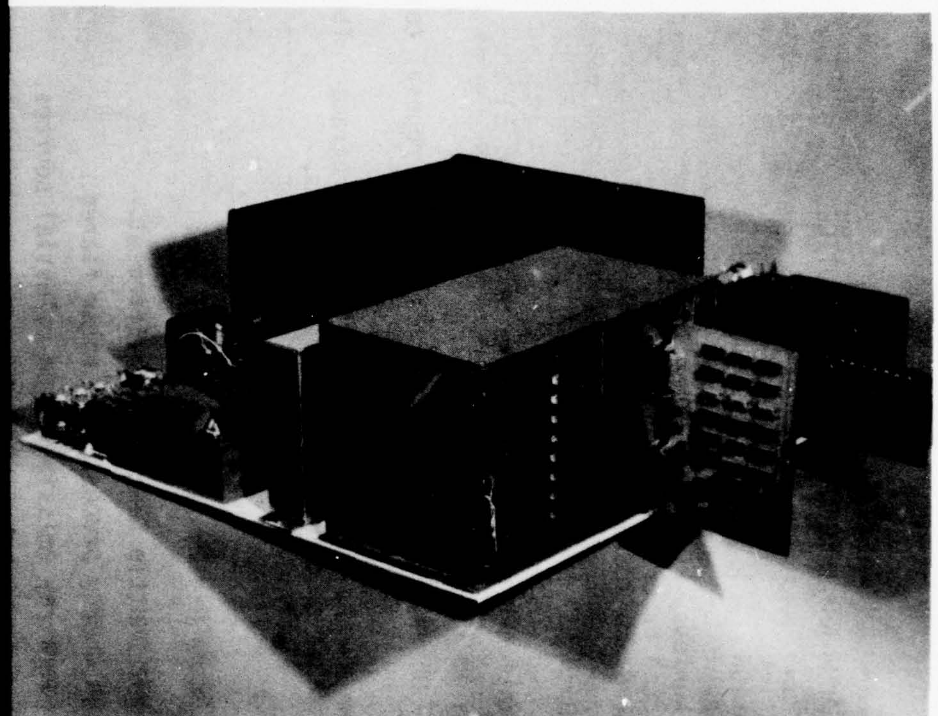
This is a major concern of gamma-ray experiments in the 0.1 to 10 MeV energy range. Thus, one of the primary objectives of this experiment is to observe this environment and investigate observational techniques which will minimize the effect of this background on the measured results. The knowledge gained from these measurements can then be used to enhance future solar and celestial gamma-ray experiments operating in the Shuttle environment. Time will be allocated to map the locally produced background in the Shuttle.

In order to study gamma-ray sources in the presence of this background, the instrument will alternate measurements of the source with background measurements by offsetting the pointing from the source on a 1-2 minute time scale.

The experiment instrumentation consists of a "phoswick detector system," associated electronics and an orienter system. This instrumentation will be derivatives of a system flown by NRL on balloons. Figure 3.0-1 shows the balloon-borne orienter and electronic equipment.



AN X-RAY DETECTOR MOUNTED IN
ORIENTER SYSTEM FOR BALLOON FLIGHT



EXPERIMENT ELECTRONIC INCLUDING MICROPROCESSOR
FOR DATA ACQUISITION AND ORIENTER CONTROL

Figure 3.0-1. Experiment Configuration for Balloon Operation

The "phoswich" detectors which are sensitive to gamma rays in the 0.1 to 10 MeV energy range consis of large sodium iodide-thallium activated (NaI (Tl)) scintillation crystals optically coupled to cesium iodide-sodium activated (CsI(Na)) scintillation crystals. These phoswich assemblies are then, in turn, optically coupled to seven ruggedized photomultiplier tubes. The CsI section provides active anticoincidence and helps define the active NaI volume via pulse rise time or shape discrimination. The pulse shape discrimination is also used to distinguish between gamma ray and neutron energy losses greater than 10 MeV in the NaI portion of the phoswich. The detector system is shown in Figure 3.0-2.

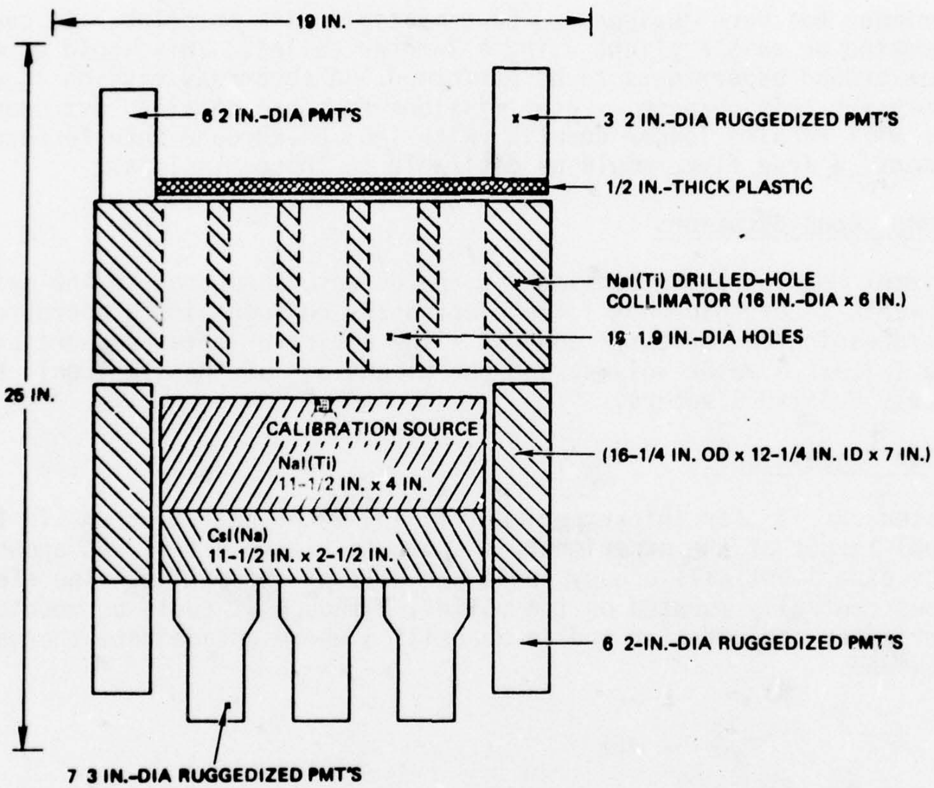


Figure 3.0-2. NRL Gamma-Ray and Neutron Detector

The lower annular NaI scintillation section is viewed by six ruggedized photomultiplier tubes and provides anticoincidence discrimination against charged particles entering from the sides of the instrument. The upper NaI section has holes drilled as indicated in the figure to define the detector aperture or field of view. A one-half inch plastic scintillator viewed by three photomultiplier tubes covers the entrance aperture and completes the 4π active shielding for charged particles.

The orienter system is a computer controlled pointing system developed by NRL for its balloon flights. Differential measurements are performed by pointing the detectors to a solar or celestial gamma-ray source and by making a comparison of the count rates to those from an off-source (background) pointing.

A schedule has already been planned for this experiment as follows:

- Detail Design — Oct 77 to May 78
- Fabrication — Jun 78 to Jun 79
- Test — Jul 79 to Jan 80
- Deliver to Integration — Jan 80

4.0 ASSESSMENT FOR STS FLIGHT

This experiment has been designed to be compatible with Spacelab. It can readily be accommodated on an STP flight using a 3-meter pallet. This would permit the Shuttle background experiments to be performed. A seven-day mission should be satisfactory for this purpose. Later missions for more detailed astronomical observations will require longer duration with less background interference. For these reasons, a free flyer would be desirable in those missions.

4.1 Experiment Considerations

The experiment configuration consists of the detectors mounted in the orienter plus a separate electronics box. The electronics box contains a microprocessor for data processing and orienter control. The entire orienter and detector system occupies a 1x1.3x2.2 meter volume, and the dimensions of the electronic box are approximately 0.3x1x0.3 meters.

4.1.1 Design Suggestions

The suggested carrier for this experiment is a 3-meter pallet on an STP flight. A conceptual layout of the experiment is shown in Figure 4.1-1. As shown in the figure, the experiment will occupy about one-half of the pallet. The electronics box is shown centrally mounted on the pallet, although it could be mounted anywhere near the orienter and detector and in a position where an adequate thermal interface can be provided.

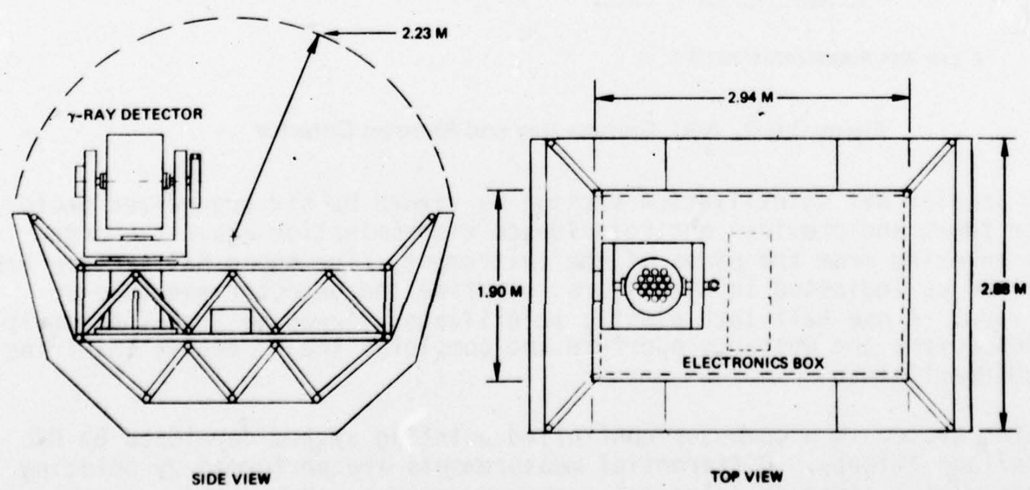


Figure 4.1-1. Conceptual View of Gamma Ray Detector and Orienter Mounted to Spacelab Pallet

The detectors will be maintained at a temperature of 20 ± 5°C by passive insulation and thermostatically controlled heaters provided by the experiment. The electronic box will have to be mounted on a cold plate.

All the resources required by the experiment can be directly supplied from the orbiter without need for Spacelab components with the exception of the pallet and cold plate.

Pointing

Relatively course pointing accuracy is adequate for the gamma-ray measurements in this spectral region. The experimenters propose to provide the orienter as part of the experiment. The orienter is a two-axis system and is a direct adaptation of an existing orienter which has successfully flown on many balloon borne flights. It consists of a bi-axial orientation platform, controlled by the microprocessor. The angular resolution on each shaft is limited to approximately 0.1 degrees by a 12-bit shaft angle encoder used in a digital positioning system. This approach can result in substantial cost savings, but special attention should be paid to the qualification of this device for Shuttle use. This orienter, as is the case with most balloon pointing systems, is centrally stem supported from above and is designed to operate in gravity. This, of course, will have to be modified. Further significant design changes may be required to beef up the structure to withstand the Shuttle launch environment.

A pointing accuracy of about 1° is required. This will be provided by the experiment's own star/sun sensors. The orienter control system has a microprocessor which performs all the computations required to orient the detectors. The microprocessor will require orbiter attitude references to perform these computations. These can be obtained from the Orbiter Guidance, Navigation and Control computer. Offset pointing maneuvers for background measurements will also be generated by the experiment's microprocessor.

Electrical & Data

The experiment requires 60 watts of power on the average and 100 watts peak. This can be readily provided by the orbiter. If power regulation better than the 28 $\pm$ volt range provided by the orbiter is required, this will have to be provided internal to the experiment.

The data handling requirements for this experiment, (i.e., 5.4 kbps normal and 5.4 kbps for 10% duty cycle) can be accommodated through the orbiter data processing and software subsystem. This data are obtained from the experiment via a single redundant serial PCM data channel. About 100 commands will be required from ground to the experiment via the orbiter.

4.1.2 Flight and Ground Restrictions

Any Shuttle orbit in the altitude range of 280-450 km and with inclinations less than 57° is adequate although the lower inclinations are more desirable since the background variations are more severe at higher inclinations. Some real time monitoring and control of the experiment are desired although the operation of the experiment can be completely automatic. In any case, minimal payload specialist training ($\sim$ 40 hrs) will be required. It is anticipated that the payload specialist will have complete control of the experiment throughout the mission; but no monitoring or physical activities are required of the specialist. Ground controllers will have the primary responsibility for the performance and observing program of the experiment. The ground controller should be capable of near-real time analysis of significant data to assure the health and maintenance of the experiment. He should also be in communication with other observers to assure a quick response to solar or astrophysical phenomena of particular interest.

Flight operation activities of the type described above can be conducted from the POCC at JSC. The "host" mode of operation is suggested. In this mode, POCC provides the facilities for data monitoring, payload commanding and voice communications with the Mission Control Center (MCC). The user provides all the payload operations personnel to support all the real time crew activities. JSC provides a liaison position to familiarize the user with the POCC.

No unusual ground operation restrictions are anticipated.

The experiment should not be flown with other experiments that use radioactive materials. The total flux of line gamma rays at the experiment envelope must be kept below 10^{-3} photons/cm^2-sec.

4.1.3 Other Considerations

No safety hazards are anticipated.

The orienter is launched and landed in a stowed position with all rotation axes pinned. To prevent a safety hazard in the event of failure of the stowed configuration, the orientation system should be designed to withstand the crash loads in an unstowed position. The experiment remains within the orbiter bay closed door envelope. No hazardous materials or components have been identified.

Optional Shuttle services requiring additional costs to STP over and above those basic Shuttle services for this experiment might include:

- Payload mission planning services, other than for launch, deployment and entry phases. This may be required to support the operations discussed in Section 4.1.2.
- Payload data processing.

We have assumed that the approach described in this assessment could be performed with the basic three-man orbiter crew and without the need for Spacelab or other special equipment. If these assumptions prove to be incorrect, then these services will also accrue additional costs.

4.2 STP Integration Considerations

The conceptual layout for this experiment was shown in Figure 4.1-1.

In order to mount the orienter on the pallet so that the instrument can attain its required 15° clear field of view without interference from the Orbiter in all pointing directions, it is necessary to mount the instrument on an elevating platform, as shown in the figure. The raised platform is attached to the pallet at standard equipment hard points on the pallet side panels and with structural support members to hard points on the floor panel of the pallet. In this configuration, the instrument fits within the 2.23 meter radius static payload envelope of the orbiter for all detector orientations.

The total experiment weight including the platform is approximately 450 kg.

Cabling to the appropriate orbiter stations for data, commands and power will have to be provided by STP or the user. Furthermore, as discussed previously, a cold plate will be required for cooling the electronics.

No unusual pre-launch testing is anticipated. Integration testing will be performed using NRL supplied GSE. Small radioactive sources will be required to perform these tests and provisions for storage of these sources at the integration site will be required.

The schedule shown in Section 3.0 is compatible with any STP flights after mid-1980. The experiment can be delivered to integration in January 1980. It would take approximately six months for integration into a payload.

5.0 RECOMMENDATIONS AND REMARKS

This experiment was proposed for a Spacelab flight and is thus well suited for an STS flight. It can be flown on a 3-meter pallet using power, command and data handling services from the orbiter. If the balloon-borne orienter proposed for this experiment can be modified for use on the orbiter, it will be a significant cost saving over the use of a Spacelab supplied pointing system.

An alternate method of accommodating this experiment could be provided by the "smart" Standard Test Rack (STR) presently being studied by STP. This rack could provide the mounting and pointing of the detectors in the orbiter bay as well as power, data, commands and thermal control for the electronics. This facility, however, is still in the study phase.

This experiment is also recommended for follow-on flights using a free-flyer to permit longer observation of solar and celestial events.

LASER EFFECTS AND HARDENING OF SATELLITE MATERIALS AND COMPONENTS

1.0 EXPERIMENT IDENTIFICATION

J. T. Schriempf & L. C. Towle
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Code 6410
Washington, D. C.

2.0 REQUIREMENT BACKGROUND

Response to STS Utilization Presentation (Response Sheet #6)

3.0 EXPERIMENT APPROACH

Selected materials and components, either presently utilized or contemplated for use on military satellites, will be irradiated with a focused high energy laser beam. The unpressurized pallet structure in the shuttle bay will be utilized for mounting the experiment apparatus. The material specimens (targets) will be mounted on an instrumented carousel and upon deploying out of the bay, rotated into the target area for irradiation. Post-flight damage assessment will be performed on earth to supplement several on-site parameters measured during the actual exposure.

4.0 ASSESSMENT FOR STS FLIGHT

If a gas or chemical laser source is used, the large amount of gases will be vented during experiment operations. This experiment should be conducted on an STS mission where optical or surface contamination is not a critical concern to another experiment.

4.1 <u>Experiment Considerations</u>

The experiment will require development of the carousel and positioning devices and some crew training for familiarization. The STS interfaces, in addition to structure, are power and data handling. Data processing such as temperature measurement as a function of irradiance time is desired but may be self contained by the experiment equipment itself.

4.2 <u>STP Integration Considerations</u>

Because of the need to develop special equipment for this experiment, there will be a significant lead time required. In addition, preliminary ground testing will be required in order to establish experiment and operation parameters. This testing will also provide one gravity samples for comparison with the flight specimens.

One feature of accommodating this experiment will involve locating the laser beam and target area with respect to other structures including the Orbiter to preclude self-inflicted dammage. A jettison device for the target carousel will be necessary.

Special venting controls for the laser should be considered in order to reduce contamination, but would increase the time and cost of resolving accommodation and interface details. If the package requires classification, it is recommended that operating and data parameters not be processed by STS avionics.

Estimated operational times are one-half hour per test and a total of four hours needed on a given flight. A typical weight is 4000 kg; 2 KW/AC of power will be required.

5.0 RECOMMENDATIONS AND REMARKS

An alternate concept, being considered by the Investigators, would substitute a thermal pulse derived from a solar optical source to replace the laser irradiation. This obviates the possible contamination due to the laser operations. However, it is recommended that further consideration will have to be given to control of material offgassing.

This mode also generates a requirement for solar tracking with a reasonable pointing accuracy. It may also be necessary to deploy the optical source above the pallet. With a laser source, there are no STS flight operations constraints.

HEAVY IONS IN SPACE

1.0 EXPERIMENT IDENTIFICATION

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NRL/Laboratory for Cosmic Ray Physics, Code 7020
Washington, D. C. 20375

2.0 REQUIREMENT BACKGROUND

Response to STS Utilization Presentation (Response Sheet No. 7)

Experiment Number NRL 702

Heavy Ions in Space

HZE Radiation in Space

3.0 EXPERIMENT APPROACH

The purpose of this experiment is to investigate the three components of the heavy nuclei, N-O-Ne, in the upper atmosphere. The three components of interest are those in galactic cosmic xrays, those in the radiation belts and a recently discovered anomalous component. In the past five years, spacecraft observation has indicated that an enhanced flux of N, O and Ne is present in the upper atmosphere. Exploration of this anomalous component has mainly been performed at energies less than 30 MeV/nucleon. At higher energies, the large abundance of normal cosmic-ray nuclei masks the effect.

The Heavy Ion in Space experiment will separate these components by means of a plastic track detector flown at low latitudes. These detectors are passive and processed after return from orbit. If the anomalous component nuclei are singly charged (as some evidence suggests), these nuclei would have high rigidity for a given energy per nucleon and thus could not be present at latitudes below 30°. Ordinary galactic cosmic-rays are fully stripped and therefore cannot penetrate the geomagnetic field at these low latitudes. Therefore, by examining the spectrum of the N-O-Ne nuclei at these latitudes, the separation of the anomalous component should be possible.

Furthermore, the use of the plastic track detectors permits the examination of the heavy nuclei component in the radiation belts. The energy and charge of an incident particle on these detectors are determined from the damage produced in the plastic. The effects of heavy nuclei on these detectors are not masked by the high fluxes of radiation belt electrons and protons.

The expected flux of the 100 MeV/nucleon anomalous component of heavy nuclei is between 70 and 800 particles/m^2 ster. yr. for N, O and Ne. Therefore, the detection of these particles requires a long exposure for the detectors.

This experiment can be ready for integration in two to three months. The experiment has already been proposed for a LDEF flight.

4.0 ASSESSMENT FOR STS FLIGHT

The Heavy Ion in Space experiment is an excellent candidate for an LDEF flight. It is completely passive and requires no services from the STS. The low inclination, low altitude orbit required and its required time in orbit are completely consistent with LDEF mission plans.

4.1 Experiment Considerations

4.1.1 Design Suggestions

Mechanical, Pointing, Field of View

The instrumentation for this experiment consists of 9 modules, each 30 cm (11.8") x 40 cm (15.7") and 7 cm (2.8") deep. These modules provide one square meter of detector area. Each detector consists of layers of plastic and nuclear emulsions interleaved with brass. Three different types of plastic are used. The total experiment mass is 120 kg (264 lbs.).

The experiment must look away from the earth with a pointing accuracy of $\pm 20°$. A 90° clear field of view is required. These requirements can readily be met by mounting the modules in trays in the LDEF end frames to view along the LDEF longitudinal axis. Since the LDEF is gravity-gradient stabilized, this axis will be aligned with the local vertical as shown in Figure 4.1-1. A magnetic damper is employed by LDEF to provide viscous damping for transients. When first injected into orbit, the vehicle will undergo large oscillations and the pointing requirement of $\pm 20°$ will not be met. However, within 8 days, a steady state will be reached and the direction of the longitudinal axis will be oriented to within $\pm 2°$ of the local vertical and the oscillations about this axis will be reduced to within 5°.

LDEF can accommodate end mounted experiments by means of end trays. Six end trays are mounted on one end of LDEF and eight end trays on the other. Two different kind of end trays are available, "corner" trays and "center" trays. Corner trays are about 28-1/2" x 28-1/2" square and center trays are about 40.75" x 33.5" in area. Both types come in 6" and 12" depths. The 10 Heavy Ion in Space experiment modules can be accommodated in two center trays as shown in Figure 4.1-1. Each of these trays can support a mass of 175 lbs.

Thermal Control

Thermal control for the instrument should be provided by the experimenter. Some control will be required since the detector should never reach temperatures greater than 30°C. This should be attainable by passive means on orbit, however, during reentry and ground operations, other provisions may be necessary.

Data Handling and Power

The basic experiment does not require any power or data handling. However, the experiment would be improved if the time history of the temperature in the vicinity of the detectors while in flight, were obtained.

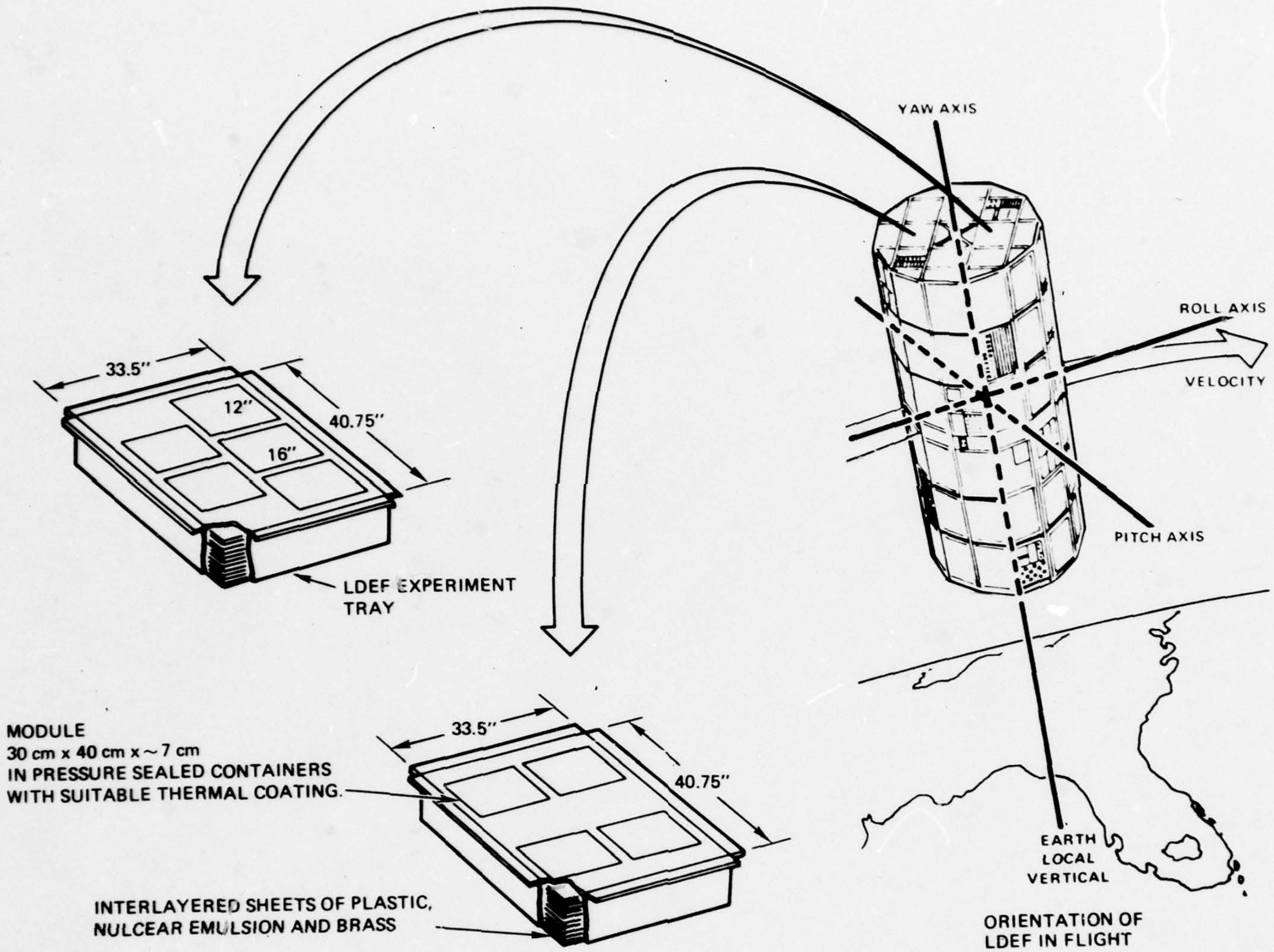

Figure 4.1–1. Conceptual Layout of Modules for Heavy Ions in Space Experiment in an End Center Tray (Two Trays are Required for the 9 Modules)

SHUTTLE EFFECTS ON PLASMAS IN SPACE

1.0 EXPERIMENT IDENTIFICATION

Dr. E. P. Szuszczewicz, Principal Investigator
Naval Research Laboratory
Upper Air Physics Branch, Code 7127
Washington, D. C. 20375

2.0 REQUIREMENT BACKGROUND

Response to STS Utilization Presentation (Response Sheet No. 8)

This is experiment NRL-701, Project No. RR033-02-42, Task No. RR0330242, Sponsored by Naval Electronics Systems Command (PME-106)

A Form 1721 has been submitted for this experiment.

3.0 EXPERIMENT APPROACH

The investigator proposes to use two pulsed plasma probes to examine the local plasma environment of the Shuttle cargo bay during an early STS flight. There may be serious local contamination near the STS orbiter. The two pulsed plasma probes will be able to quantify the extent of the modification by the orbiter of the ionospheric electron density, electron temperature, electron density fluctuation power spectrum and the level of ambient-particle contamination. The probes should be mounted to sample the ambient, if possible, when the cargo bay is turned into the ram direction. It is also possible that the probes could be mounted on a boom or mast deploying them to 7 meters above the top of the cargo bay. It is also desirable that the probes be mounted as near the top of the cargo bay as possible.

Figure 3-1 shows a plane view drawing of the instrumentation. The cylindrical and loop probes make the same basic measurements. The weight of the two units combined is 6.8 kg, the volume is 8400 cm^3, the average power is 6 W, the peak power 7W, the duty cycle is 33% with the orbit average power of 2 watts. The total data rate is about 25 Kbps.

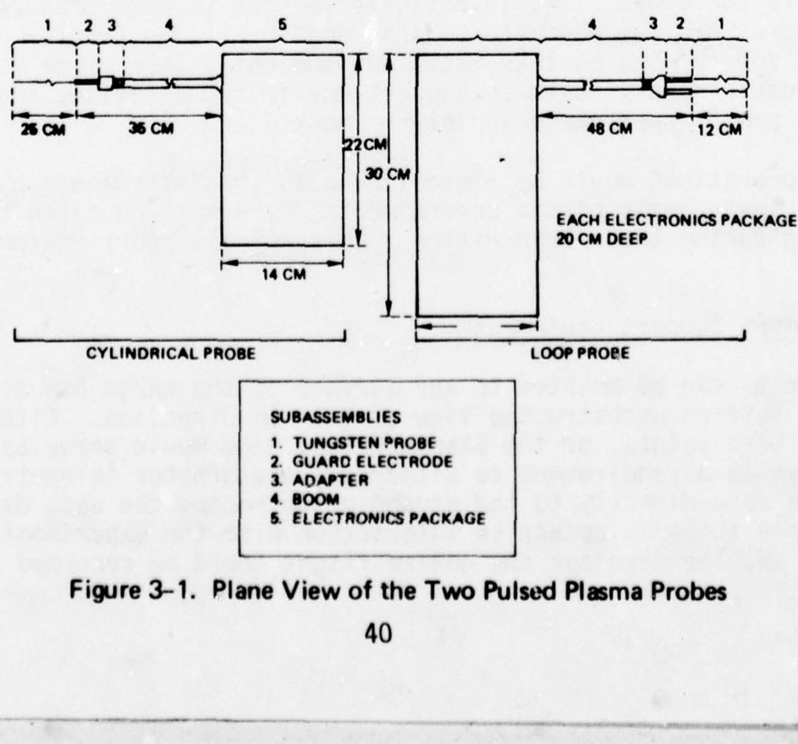

Figure 3-1. Plane View of the Two Pulsed Plasma Probes

4.0 ASSESSMENT FOR STS FLIGHT

This instrumentation appears to be compatible with the STS and is well qualified to fly early in the program. There are strong scientific reasons that this or some similar instrumentation be flown early in the STS program in order to assess the contamination of the local environment, particularly the ionospheric environment, that is caused by the Shuttle. There is a large class of experiments that plan to investigate the ambient plasma behavior from a Shuttle-based platform. Accordingly, there is a pressing need that the Shuttle contamination be studied before these other experiments are in their final design stages.

4.1 Experiment Considerations

4.1.1 Design Suggestions

The instrumentation exists and is well adapted to the measurements that are to be carried out. To make the full range of observations in the near-Shuttle environment, the investigator should consider the design of a mast or boom that could be mounted to the Shuttle and which would extend vertically above the cargo bay. Either a telescoping rigid boom or an astromast could be considered for accomplishing the vertical scanning.

The investigator will have to provide his own thermal control if the experiment is located at the top of the cargo bay. This could be accomplished by using a high reflective paint and providing heaters inside the electronic boxes.

4.1.2 Operation Restrictions

There is a requirement that the instruments make observations when the cargo bay is pointed in the orbital velocity ram direction, X-POP Y-vertical or Y-POP X-vertical. There are no requirements as far as orbital altitude or inclination are concerned. The investigator wishes to make measurements over as wide a range of orbital parameters as is possible. The Orbiter crew would be responsible for turning on the instrument and doing some minor status checks. If an extendable mast is used to support the instrumentation, the crew would carry out a pre-planned extension of the mast.

The ground operations would be minimal because the instruments operate fully only in the ionospheric plasma environment. Care must be taken to keep the probes clean during their integration. No special support equipment would be required.

4.1.3 Experiment Support Equipment

The instruments can be mounted to any surface in the cargo bay so long as they are able to have an unobstructed view of the ram direction. Either a pallet, the orbiter hard points, or the Standard Test Rack would serve as a mounting point. There is a requirement to either use the orbiter telemetry system to transmit the data directly to the ground or to record the data during the flight. Since there is no active interaction with the experiment once it has been turned on, the data for the entire flight could be recorded and recovered after the flight.

The instruments have no need of active pointing. The power and weight are minimal and would make no significant impact on any Shuttle flight. The experiment could receive power directly from either the orbiter or the Spacelab power supplies at 28 volts. The data could be made RAU compatible if the experiment is flown on a Spacelab flight, or could be put directly on the Orbiter multiplexer.

4.1.4 Experiment Cost Considerations

This experiment equipment already exists. The only added cost would be the design and construction of the mounting mast and the cabling required to attach the instruments to the electrical power and data handling systems of STS/Spacelab.

4.2 STP Integration Considerations

The integration of this experiment onto STS would be relatively easy. The instrumentation can be placed in a large number of locations. Figure 4-1 shows a conceptual layout of the instruments. There is a requirement that when the cargo bay is in the ram direction, no other equipment be mounted so that the plasma probes are obscured from the direct flow of the ionospheric plasma, nor lie within 50 cm of the probes themselves. This requirement could be softened somewhat if the probes are mounted on an extendable mast.

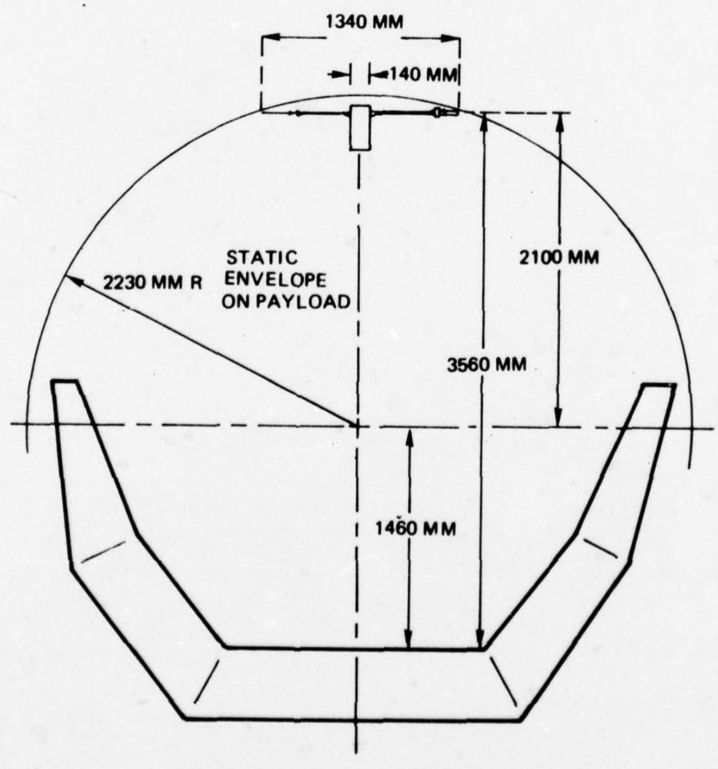

Figure 4-1. Conceptual Layout for the Pulsed Plasma Probes in the STS Cargo Bay. A Nominal Location Between Stations ℄ = 576 in. to 1000 in. would be Desirable.

5.0 RECOMMENDATIONS AND REMARKS

1) This experiment is fully compatible with the STS and because it is designed to answer questions regarding the Shuttle environment which are important to many experiments, it should be flown early in the STS era.

2) The investigator should examine the possibility of using either the RMS or preferably an extendable mast in order to raise his instrumentation 10 meters above the floor of the cargo bay. A jettison device would have to be provided as a part of the extendable mast.

3) Because of the nearly self-contained nature of the experiment and its lack of requirements on the STS, there should be no major problems in its integration.

CRYSTAL GROWTH AND HOMOGENIZATION OF
SEMICONDUCTOR AND LASER MATERIALS

1.0 EXPERIMENT IDENTIFICATION

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2.0 REQUIREMENT BACKGROUND

Response to STS Utilization Presentation (Response Sheet #14)

3.0 EXPERIMENT APPROACH

Two cylindrical samples each of Nd - doped laser glass and polycrystalline AlSb will be processed in a zone melter. Container wall effects, unavoidable in one-g processes, are a likely cause for compositional inhomogeneity in Nd-doped glass. This limits its performance in high-power laser applications. The semiconducting compound AlSb is a potentially high efficiency solar cell material. The objectives of these experiments are to achieve better homogenization of the laser glass and produce more perfect single crystal AlSb than presently possible in a one gravity environment.

4.0 ASSESSMENT FOR STS FLIGHT

This investigation requires a float zone melter of the type being considered by the NASA-MPS program or an equivalent facility.

No serious problems are expected in integrating this experiment with other zone refining/crystallization experiments that use the same equipment. The principal areas of concern are the initial melting of seed-sample interface of the AlSb experiment and the melt width obtained by resistive heating. Use of alternative means of heating the melt zone such as radio frequency (rf) or laser methods may be necessary. A maximum melt length not greater than π D represents the upper melt zone length for stability.

4.1 Experiment Considerations

The experiments are to be performed in ultra high purity argon at a pressure of $10^{-5} N/M^2$. Acceleration level during processing should be 10^{-3} g or less, including g-jitter. Operating temperatures of the zone heater are 1200 $\pm$ 12°C for the AlSb specimens and 1500 $\pm$ 15°C for the laser glass specimens.

Unprocessed samples must be placed in contamination-resistant containers using "clean room" procedures in order to minimize surface contamination. If this precaution is not taken, serious surface degradation could occur during processing.

The apparatus design should afford shock protection for processed samples to minimize physical damage during landing and subsequent handling. The only STS flight constraint during processing is maintenance of low acceleration levels.

The type of experiment lies within capability of the apparatus being considered by NASA. If available, through NASA, all necessary support equipment will be available for use and no rework or retrofit need be contemplated. Time-lining of the experiment protocol makes it desirable to utilize the maximum time available in orbit for processing. The melt-growth rate for typically 10 cm. long samples will require at least several hours per specimen processed. Rates to be studied have yet to be specified. Parallel specimen processing while shortening the percent of mission time required, would impact the float zone apparatus design and operating power requirements.

Figure 4.1-1 illustrates the basic elements needed of a linear and rotational enclosure assembly which are required for this experiment. Figure 4.1-2 is a schematic representation of the drive unit/heater combination and auxiliary equipment. Typical performance values being considered by NASA for a zone melter apparatus are listed in Table 4.1-3.

The necessary STS interfaces to the apparatus are power and structural mounting of the facility, controls, and data acquisition will be desired. Acceleration level as a function of zone length travel time is very desirable so that a correlation of specimen perfection per unit of processed length as a function of acceleration level can be made.

4.2 STP INTEGRATION CONSIDERATIONS

Based upon using MPS apparatus yet to be developed, the experiment and attendant equipment are straightforward, and there are no particular requirements that impact normal payload integration lead times.

In order to obtain verification of the enhancement of material properties due to low gravity processing control, specimens processed in 1 g. are necessary. Utilization of equipment identical to that onboard the STS will be required in order to control process variables as much as possible.

5.0 RECOMMENDATIONS AND REMARKS

It is recommended that this experiment be processed in the pallet zone melter, located in the unpressurized portion of the Shuttle bay if that flight configuration is the initial version developed. It would require less automatic sample change out and be somewhat more flexible to perform the experiment in the Spacelab pressurized module, but the required power and potential hazard of releasing antimony vapor into the atmosphere may mitigate against this operational mode.

An alternate experimental approach to that considered would be to process one-half of the specimen under one gravity conditions and process the remaining on orbit. In this manner, sample material and process variations would be minimized, aiding in the assessment of the effect of reduced gravity on processing these types of materials.

EFFECTS OF SPACE ENVIRONMENT ON SPACECRAFT MATERIAL

1.0 EXPERIMENT IDENTIFICATION

Mr. E. N. Borson, Principal Investigator
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Materials Sciences Laboratory, A-6/1421
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2.0 REQUIREMENT BACKGROUND

Response to STS Utilization Presentation (Response Sheet No. 15)

Work Unit Numbers: 31451959, 31451962, 682J0401, 682J0404

3.0 EXPERIMENT APPROACH

Solar cells and thermal control coatings, optical materials, and other materials contemplated for use on spacecraft will be exposed to the space environment using the Long Duration Exposure Facility (LDEF). Some in-situ measurements of specimen environment will be made during the 6-9 month mission duration, but most experiments will be conducted on earth after LDEF retrieval to determine the extent and type of degradation.

The immediate objectives are to understand the changes in the properties of the materials and compare these changes with predictions based on ground-based laboratory experiments. The longer term objectives are to improve the performance and usage of existing materials and to decrease the lead times for the application of new materials.

4.0 ASSESSMENT FOR STS FLIGHT

4.1 Experiment Considerations

The experiment will be conducted in the LDEF and consists of three standard trays, 1.3m x 1 m. Each tray will be equipped with an Experiment Power and Data System (EPDS) for the in-situ measurements. Any additional instrumentation will be fabricated by the Principal Investigator. Two Vacuum Exposure Control Canisters (VECC) will be used with each tray to protect certain specimens during launch, initial deployment and before retrieval of the LDEF. The remaining 1/3 of each tray will house specimens not requiring surface fidelity.

The experiment trays will be located on the leading edge, the trailing edge, and in the greatest solar view angle area. The purpose of one tray in the ram direction and another in the wake is to compare the effects of enhanced atomic oxygen and particulates on the materials. There is no constraint on the STS with respect to any aspect of this experiment. The LDEF and the VECC are described in Appendix A.

4.2 STP Integration Considerations

The experiment is straightforward, and there are no particular requirements that require abnormally short or long delivery lead times.

At 28.5° orbit inclination, the requirement for greatest solar viewing is essentially satisfied by the tray in the ram position; therefore, only two trays are needed.

Integration of this experiment with other LDEF experiments should pose no problems, since the experiments are confined to trays, are protected where necessary during outgassing or "dirty" phases of the mission and are independent from other experiments. Therefore, the experiment packages can be considered to be independent of STP except for integration into LDEF, retrieval and subsequent removal for shipment to the Principal Investigator.

5.0 RECOMMENDATIONS AND REMARKS

It is recommended that this experiment be placed in the LDEF since it provides experimenters with an opportunity to recover specimens that have been exposed for long periods in space. Although many materials appear to be satisfactory for a variety of spacecraft applications, there is insufficient knowledge of their physical and optical properties after long term exposure to space. Laboratory tests do not simulate the actual space environment with sufficient fidelity to enable accurate prediction of property changes as a function of exposure time. In addition, this cooperative effort among appropriate DoD laboratories affords an economical approach to large scale testing of many new materials presently being developed for spacecraft usage.

OPTICAL COUNTERMEASURES DEMONSTRATION
(Satellite Survivability Program)

1.0 SOURCE

Dr. Robert M. Cooper, Principal Investigator
Aerospace Corporation, Material Sciences Laboratory
and Lt. Vic Slaboszewicz, Project Officer
SAMSO/YAS
P. O. Box 92960
Worldway Postal Center, Los Angeles, CA 90009

2.0 REQUIREMENT BACKGROUND

Response to STS Utilization Presentation (Response Sheet No. 16)

Identifying Number: not known

Details of experiment are classified.

3.0 EXPERIMENT APPROACH

3.1 Objectives

The objective of this test is to demonstrate the performance of optical countermeasures against lasers. A secondary objective is to obtain measurements of laser beam degradation caused by atmospheric turbulence and absorption. The countermeasures to be demonstrated are under development by the Air Force Materials Laboratory and by SAMSO.

3.2 Experiment Description

Tests will be conducted in conjunction with laser radar trackers situated at MIT Lincoln Laboratory (43°00'N, 72°00'W) and Holloman AFB (32°51'N, 106°06'W). At least 10 individual encounters with each of the two test sites will be required. The orbital period, inclination, and ascending node should be selected to maximize the number of encounters between the payload and the two test sites.

The tests will consist of acquiring, tracking, and illuminating the payload package with the laser tracker. Measurements of intensity will be made with radiometers located on the payload package and on the Shuttle Orbiter during operation of the optical countermeasures. Individual encounters will last approximately three to five minutes. Typically, there will be four encounters per day.

3.3 Orbit

The orbit should be circular with 400 km maximum altitude and at least 45 degree inclination. Polar or near polar inclination is acceptable. As described previously, the ascending node, inclination, and period should be selected to optimize the number of passes over the two laser test sites. Line-of-sight elevation angles during the tests are to be at least 60 degrees. The flights should be conducted in late summer or fall so that cloud cover over the MIT Lincoln Laboratory site is at least minimum.

3.4 Configuration

The configuration of the test equipment relative to the Shuttle orbiter is shown in Figure 3.1. The payload package must be deployed on a boom away and downward from the Shuttle Orbiter and separated by 15 meters or more. The weight and volume of the boom-mounted package are 50 kg and 70 x 70 x 50 cm^3. The package may further deploy short, retractable booms 2 to 3 meters long.

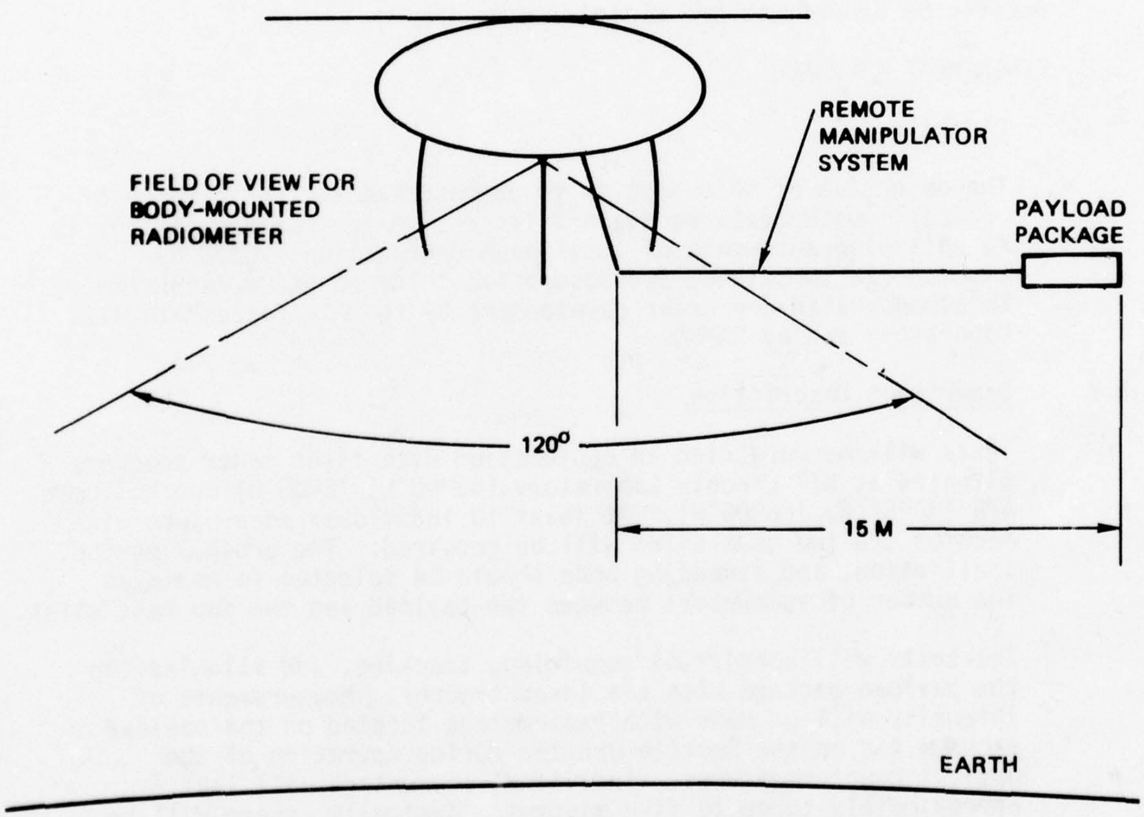

Figure 3-1. Configuration on Shuttle Orbiter of Optical Countermeasure Demonstration

There will be three small radiometers hard-mounted on the Orbiter that require a 60-degree half-angle, downward-looking field of view. These packages are 12.5 x 15 x 17.5 cm^3 in dimension, weigh 3 kg each, and should be as widely distributed on the Orbiter as possible.

Other requirements are:

Power:	50 W @ 28 VDC
Telemetry:	40 digital channels @ 1 sec
	5 wideband analog channels @ 10 KHz
Command:	10 momentary commands

4.0 ASSESSMENT FOR STS FLIGHT

No major problems are anticipated in carrying this experiment onboard the Shuttle Orbiter, possibly as an add-on or "piggyback" payload. Its small weight and compact size (when stowed) and its compatibility with orbital altitudes and inclinations of many typical Shuttle missions facilitate accommodation of this experiment. Demands on crew operation are minimal.

However, the following areas are of some concern and require further investigation:

- Although laser irradiance levels are presumably below materials vulnerability thresholds, analysis is required to assure that test irradiances of the Shuttle Orbiter and other payloads are below the hazardous level and that dangerous crew exposure can be avoided.

- The number of encounters with the two laser test sites at sufficiently high elevation angles as function of orbit characteristics and mission duration requires additional analysis.

- Remote placement of the 50 kg experiment package on a long deployable and retractable boom (e.g. the RMS arm or a payload-unique boom) requires analysis as to dynamic loads due to Orbiter maneuvers, maneuver constraints, boom tip deflections, deployment system design and Orbiter safety considerations.

4.1 Experiment Considerations

4.1.1 Deployment Boom

The experiment calls for deployment of the 50 kg payload package to 15 m or, preferably, a greater distance from the Orbiter cargo bay. The Remote Manipulator System (RMS) could be used for this purpose, as originally suggested, having a fully extended length of about 15 m. However, if the RMS is needed for other payload elements, it would then be available to the laser experiment only during a part of the mission. The time allocation depends on the overall Orbiter payload composition and requires further study.

To avoid RMS allocation conflicts, a second RMS arm could be installed to be assigned exclusively to the laser experiment. However, this would accrue extra cost and weight which would be chargeable to that experiment.

Another alternative would be the use of a dedicated, payload-unique deployment boom permanently attached to the experiment package. This would avoid the remotely controlled attachment/detachment procedures necessary with RMS, eliminate time allocation conflicts and permit payload deployment to distances greater than 15 m by appropriate boom design. The boom weight, including deployment mechanism, would be only 25 to 50 lb, compared with 800 to 900 lb for the RMS and end effector.

Possible candidates for this application would be existing coilable lattice booms (Astromast, manufactured by SPAR Aerospace Products, and Able Boom, manufactured by Able Engineering Co., both of Santa Barbara, Calif.) consisting of three continuous fiberglass/epoxy longerons with transverse battens and stiffening cables (Figure 4.1-1). These booms are designed for applications such as lightweight deployable antenna masts, as instrument support booms on spacecraft and for deployment of large solar arrays. A lightweight solar array extension boom currently being developed by Able Engineering under Lockheed contract has dimensions and characteristics that could be used for the laser experiment application:

Boom length:	105 ft (32 m)
Cross section diameter	14.5 inches
Weight	36 lb.
Bending strength (M critical)	100 ft-lb
Bending stiffness (EI)	2.5×10^7 16 in^2

However, depending on the desired deployment distance and the magnitude of the bending moments exerted on the cantilevered boom during orbiter roll and yaw maneuvers, a boom design of larger cross section diameters and greater bending strength may be required, as discussed below. The concern is with boom integrity under maximum acceleration loads rather than with tip deflections which can be minimized by refraining from thruster operation sometime prior to and during laser test site encounter events.

4.1.2 Boom Bending Moments Due to Orbiter Maneuvers

For a long cantilever boom with a 50-kg tip mass the dynamic bending loads due to Orbiter rotational maneuvers are more severe than those due to translational maneuvers. Maximum rotational accelerations during RCS thruster firing are 1.5 deg/sec^2 (for the 900 lb primary thrusters) and 0.04 deg/sec^2 (for the 25 lb vernier thrusters) according to data from the Shuttle Payload Accommodations Handbook, JSC 07700, Volume XIV, Revision D (Change #15), p. 3-38. With boom deployment in or near the direction of the Orbiter y-axis, the maximum angular accelerations in roll and yaw are of primary concern.

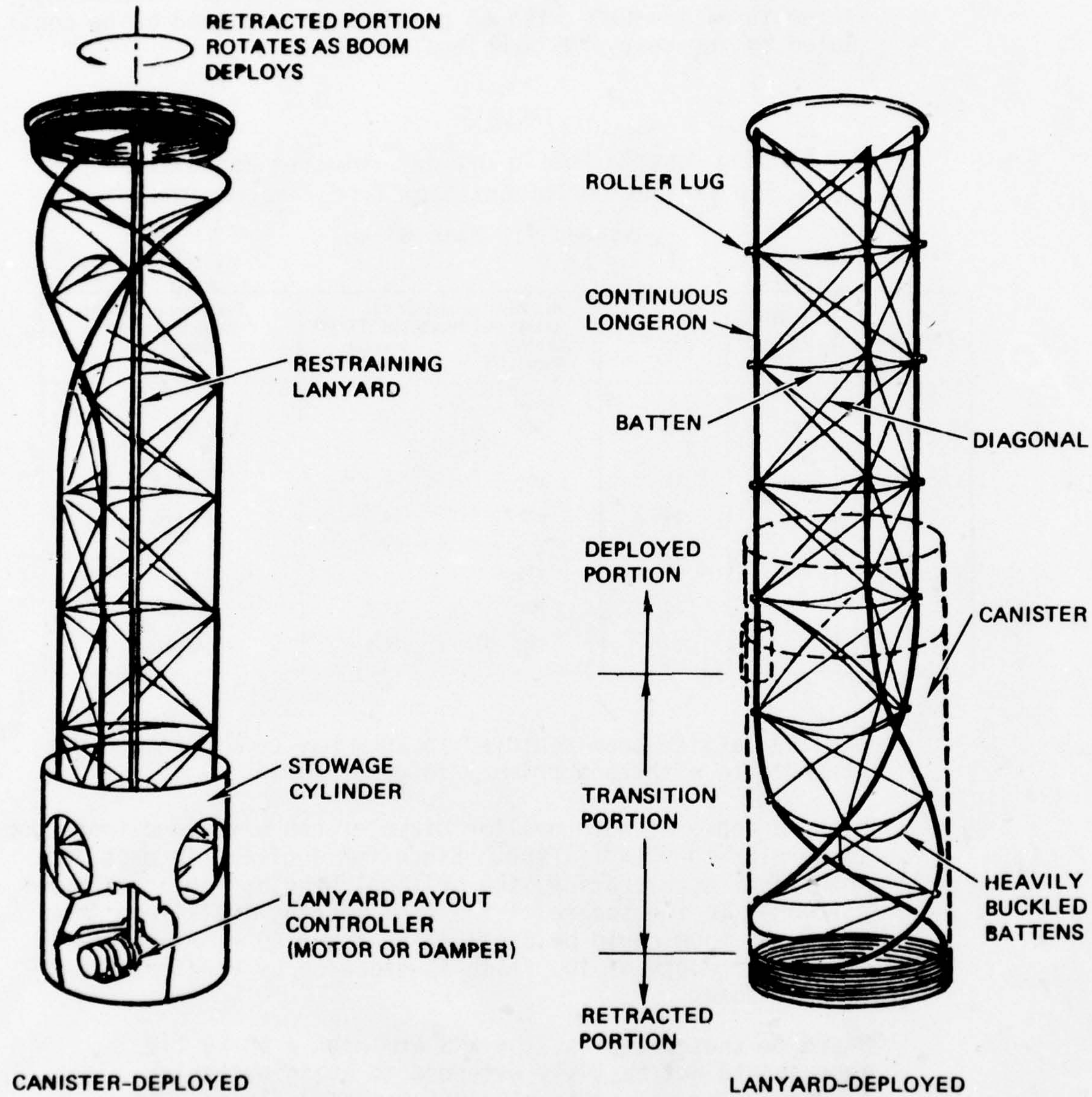

Figure 4.1-1. Continuous-Longeron, Coilable Lattice Boom (Courtesy Able Engineering Co.)

Table 1 lists maximum bending moments due to maneuver loads acting on (a) the fully extended RMS arm (15 m length) and (b) lattice booms deployed to various lengths (15 to 30 m). The bending strengths of these booms also are indicated. Note that the load due to primary thruster firing slightly exceeds the bending strength of the RMS, with 60 percent of that load being contributed by the heavy RMS arm itself.

TABLE 1

Bending Moments Due To Orbiter Rotation Maneuvers
For Various Deployment Boom Types and Lengths

(Assumed Tip Mass 50 kg)

Boom Type	Bending Strength (ft.lb)	Deployment Distance (m)	Max Bending Moment To Rotational Maneuvers (ft-lb) Primary Thrusters	Vernier Thrusters	Fraction of Moment Contributed by Boom Mass
RMS Arm	500	15	563	15.0	0.60
14.4" Lattice Boom	100	15	227	6.0	0.013
		30	920	24.5	0.026
29" Lattice Boom	800	15	236	6.3	0.051
		26	730	19.4	0.085
		30	992	26.4	0.097

The 29" lattice boom deployed to 26 m has sufficient bending strength to withstand primary thruster firing.

Lattice booms of much smaller diameter can withstand loads due to vernier thruster firing. Since for a given tip mass and rotational acceleration, the critical bending moment is proportional to the square of the boom length, it is seen that the 14.5" boom could be deployed to about 60 m before the bending strength of 100 ft-lb is exceeded by vernier thrust dynamic loads.

Based on these results, the RMS arm with a 50-kg tip mass should not be fully extended to avoid excessive bending loads during primary RCS thruster firing. If a dedicated lattice boom of large diameter (30") is contemplated to support the experiment package it could be safely deployed to 26 m. Deployment of the package to only 15 m would be possible with a boom of 19 inch diameter.

Restriction of Orbiter maneuvers to the use of vernier thrusters only would reduce the bending loads by a factor of nearly 40 and thus permit the use of booms of much smaller diameter, but would be operationally unattractive.

To avoid excessive bending moments the boom can be retracted whenever the primary thrusters are to be used for maneuvering the Orbiter. The repeated retraction/deployment cycles that would be necessary during the mission would complicate the operational sequence and may pose reliability problems. This operating mode should therefore be avoided.

4.1.3 Encounters of the Two Laser Test Sites

Careful selection of Shuttle orbit characteristics is required to maximize the number of test site encounters at elevations greater than 60 degrees. Figure 4.1-2 shows a set of daily Orbiter ground tracks for 45-degree orbit inclination. Because of its proximity to the maximum latitude of these tracks, the Lincoln Lab test site is encountered once to twice daily in spite of the small visibility circle (radius = 4 degrees) corresponding to elevation angles ≥ 60 degrees. At the lower latitude of the Holloman AFB test site the local ground track inclination is steeper and, consequently, the average number of daily encounters is appreciably smaller.

By proper choice of orbit parameters the day-to-day drift of the ground track can be adjusted such that during a short Shuttle orbit mission (less than 7 days), the number of daily Holloman encounters can be improved without noticeably affecting the Lincoln Lab encounter frequency, because of the ground track pattern geometry. An orbit inclination increase to about 48 degrees raises the Holloman encounter frequency but lowers the Lincoln Lab facility encounters. Conversely, a reduction of the orbit inclination to 42 or 43 degrees increases the frequency of Lincoln Lab encounters at the expense of Holloman encounters. According to the SAMSO Project Office, the Lincoln Lab encounters are of greater importance than those of Holloman, and this is aided by the more advantageous geographical location of Lincoln Lab relative to the ground track pattern.

The total number of useful encounters would be much increased if the experiment were to be performed at elevation angles less than 60 degrees, as indicated by the size of the visibility circles in Figure 4.1-2.

The geometrical factors discussed in the preceding paragraphs also indicate that polar or near-polar orbits are less well-suited to produce an adequate number of test site encounters than orbits of intermediate inclination.

4.1.4 Operation Restrictions

4.1.4.1 Crew Safety

A principal concern is that of crew safety during laser irradiation of the Shuttle Orbiter from the ground. Even with irradiance levels sufficiently low to avoid material damage to the Orbiter and its payloads, crew members must probably be protected against

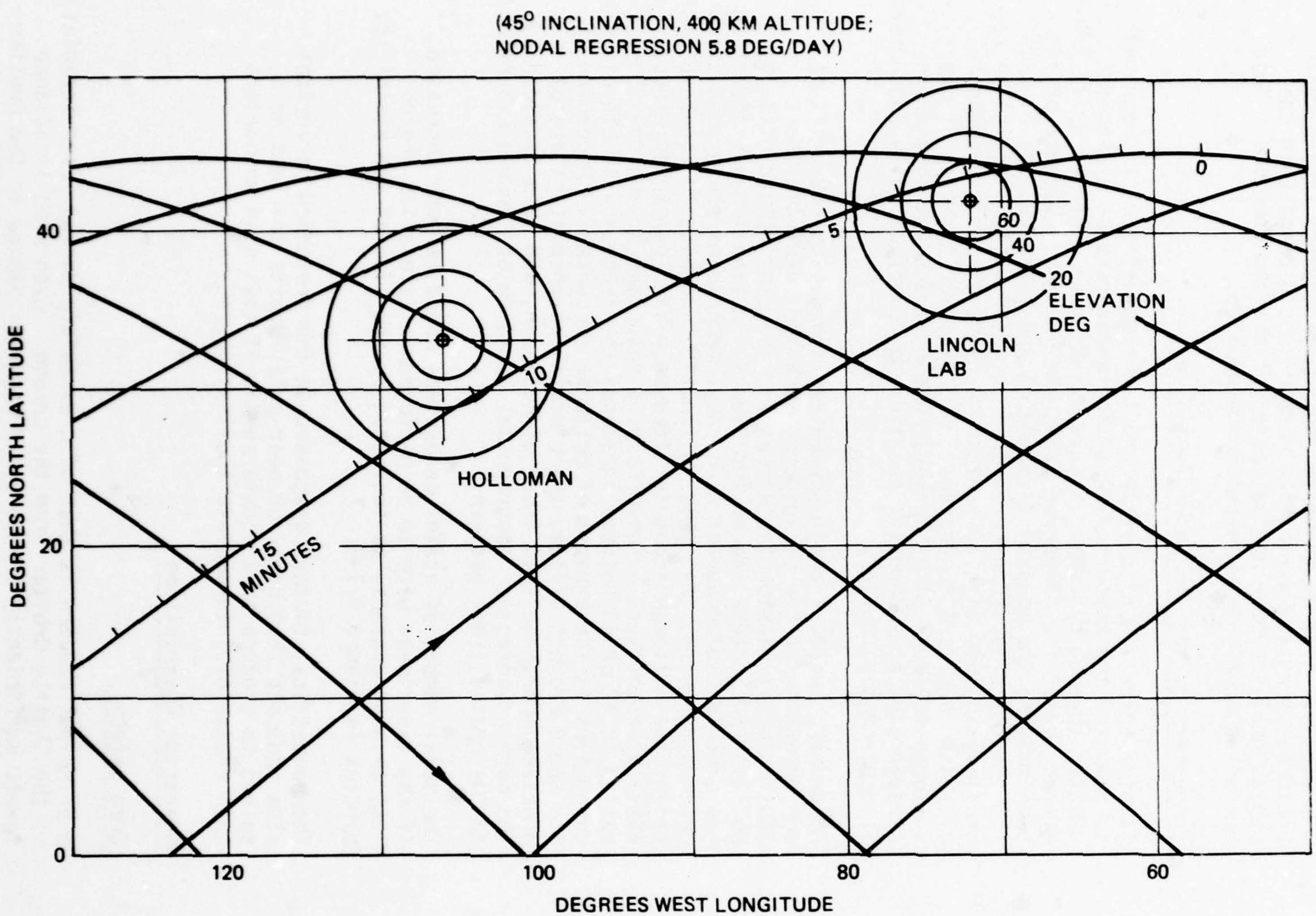

Figure 4.1-2. Orbiter Ground Tracks on Third Day from Launch and Visibility Contours of Lincoln Lab and Holloman AFB

direct exposure and especially against eye injury that could result from looking in the direction of the laser test site at the time of laser operation. To avoid inadvertent exposure during the few daily encounters and the few minutes of laser test firing during each event, it is proposed to install window blinds that will be closed during each encounter event. Assuming that the blinds are to be operated manually, audible and/or visual warning signals must be provided to alert the crew to assure that the blinds be closed prior to each active encounter.

4.1.4.2 Restriction on Orbiter Maneuvers

Depending on the payload support boom design characteristics it will be necessary to restrict orbiter maneuver accelerations to avoid damage to the deployed boom. In the event that a boom design of low bending strength is adopted, only vernier RCS maneuvers will be permitted at times when the boom is deployed. Repeated retraction and redeployment of the boom may become necessary to permit greater maneuver flexibility.

Even with a boom designed to withstand primary RCS thruster firing, major orbit change maneuvers using the 6000 lb OMS engines cannot be performed unless the payload is retracted first.

4.1.4.3 Payload Jettison Requirement

If due to some malfunction the payload boom cannot be fully retracted and secured at the end of the mission it will be necessary to jettison the boom and payload in order not to jeopardize the Orbiter's safe return. This requirement is placed on any deployable equipment carried by the Orbiter.

4.1.5 Crew Operations

The crew will participate in the operation of this experiment by deploying the experiment package, performing checkout functions, activating and deactivating the payload at each encounter, operating experiment-related support equipment in the orbiter's crew compartment as required (including tape recorders for data storage) and effecting payload retraction at the end of the mission. The crew will also verify by communication with mission control that the experiment is deployed, checked out and ready to operate.

4.1.6 Orbiter and Experiment Package Orientation

The required Orbiter orientation is earth pointing with the cargo bay open to nadir. This orientation which must be maintained during the entire mission is consistent in general with pointing requirements of other earth observation payloads.

Details of the experiment package orientation (e.g. the need to keep it pointed at the laser test site during an overflight event) have not yet been defined. Orientation changes, if required, would best be provided by the experiment package itself.

Orientation requirements are not critical. In the absence of dynamic deformation of the deployment boom (no RCS maneuver), orientation requirements such as $\pm$ 1 degree of pointing accuracy can be readily met by the proposed boom design.

Thermal deformations of the fiberglass/epoxy lattice boom are minor and can be established by on-board measurements if desired.

4.2 STS Integration Considerations

4.2.1 Conceptual Layout

Figure 4.2 shows a conceptual layout of the experiment package and payload-unique deployment boom in stored and deployed configuration on the Shuttle Orbiter. The equipment is placed on a Standard Test Rack, mounted in the forward part of the cargo bay, such that extension of the experiment package to one side of the Orbiter (starboard) provides as much lateral clearance as possible from the cargo bay and the wing structure and avoids obstruction of RMS motion. The fully-deployed 30-inch diameter lattice boom is assumed to extend to 85 ft (26m) length. The stowed boom contained in a 75 inch by 34 inch diameter stowage and deployment canister and the experiment package (28 x 28 x 20 inches) attached to it are stowed in a retention cradle parallel to the Orbiter x-axis. From this position it is rotated to an orientation normal to the x-axis, and slanted with respect to the x-y plane, before the lattice boom is deployed. In addition to the deployed experiment package three radiometers are carried by the orbiter spaced at 20 ft intervals along the cargo bay (not shown in drawing).

4.2.2 STS Interfaces

Mechanical interfaces with the Shuttle Orbiter were discussed above and involve the experiment mounting and retention fixtures plus the RMS arm and end effector, unless a dedicated deployment boom is provided for this experiment.

Electrical interfaces include the STS power supply, command channels, data handling and telemetry, and crew display panels that present payload status data, caution and warning indications. All of these support requirements are quite modest and can be readily accommodated by the Shuttle power supply and avionics subsystems. Of particular interest are the data handling and telemetry interfaces which will be discussed below.

4.2.3 Data Handling and Telemetry

The telemetry requirements of the experiment, stated in Section 3.2, translates into a maximum bit rate of 1.5 Mbps during the short active operating periods of several minutes, averaging four times per day. The total data volume for a 7-day mission is estimated to be of the order of 2.5×10^9 bits. The Shuttle data handling subsystem provides adequate capacity to record all digital and analog channels of the payload data either for temporary storage, with intermittent data dump to ground stations, or for post-flight data retrieval and evaluation. Channel capacity via Ku-band link to TDRSS, with bit rates of 2 Mbps and 50 Mbps, is adequate to provide real-time telemetry of payload data to the ground. Details of data handling requirements, formats, interface equipment and operating sequences for this experiment still need further definition.

4.2.4 Cost Considerations

The small size and weight of this payload permits accommodation on the Shuttle Orbiter at a minimum launch charge. Use of a dedicated RMS arm (at a weight of 800 to 900 lbs), which would greatly increase the installation cost and transportation charges, can be avoided by the approach discussed in Section 4.1. The cost savings may be of the order of $1 million.

5.0 RECOMMENDATIONS AND REMARKS

The experiment can be readily integrated with and operated from the Shuttle Orbiter because of its small size and weight, its compatibility with orbit characteristics typically used by the Shuttle, its modest demand on crew time and skills, infrequent operating times and modest pointing requirements.

It is recommended that questions of possible interference with other experiments and possible hazards to the crew due to intensive laser illumination for short time periods be further investigated to define adequate safety and protection procedures and equipment.

CONTAMINATION FROM SATELLITE PROPULSION SYSTEMS

1.0 EXPERIMENT IDENTIFICATION

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2.0 REQUIREMENT BACKGROUND

Response to STS Utilization Presentation (Response Sheet #22)

Identifying number and title: PE 62302F, Proj. 3058, Rocket Propulsion Technology

Supplementary information obtained from Don Young and Lou Molinari, Propulsion System Division, JPL, Pasadena, CA. (concurrent NASA-sponsored flight test program definition study, Oct. 1977- Sept. 1978)

3.0 EXPERIMENT APPROACH

3.1 Objective

The objective of this test is to perform quantitative measurements of rocket exhaust plumes under vacuum conditions in earth orbit and to characterize contamination effects of critical satellite components in close proximity of the rocket, such as solar cells, optical surfaces, thermal coatings, etc.

The principal concern is to determine whether existing analytical models of rocket exhaust flow and contamination effects are realistic and quantitatively accurate.

3.2 Experiment Description

A test facility installed in the Shuttle Orbiter cargo bay will be used to operate various propulsion system specimens in low earth orbit and to map the exhaust plume, using an array of appropriate detectors and measuring equipment placed at various locations relative to the exhaust nozzle and the main flow of exhaust products. Tests will be performed in short, continuous or pulsed operating cycles depending on thruster type.

Current plans project a series of six test missions each devoted to a different propulsion system test specimen, as listed below in the order of the most likely test flight sequence:

1. 25-lb_f monopropellant hydrazine thruster (MSFC)
2. 800-lb_f liquid bipropellant thruster using monomethyl hydrazine and nitrogen tetroxide as propellants, similar to the Shuttle primary RCS thruster (JPL)
3. 8-cm mercury ion thruster (LeRC)

4. 1000-lb_f solid propellant rocket (JPL)

5. 30-cm mercury ion thruster (LeRC)

6. 25 lb_f GO_2/GH_2 thruster (LeRC)

Dimensions, weights, propellant mass, plume characteristics, experiment power requirements, heat dissipation and other basic data of these thrusters require further definition for a more detailed assessment.

Detectors and measuring equipment to be used will depend on the thruster type being tested but will probably include the following:

- mass spectrometers
- surface collectors
- quartz crystal microbalance
- Langmuir probe and Faraday cups (to be used in ion engine tests)
- solar cell specimens

Thrust level measurements may also be included, e.g., in the tests of high-thrust propulsion systems.

To avoid undesirable Orbiter attitude perturbations during the firing of these rockets, alignment of the thrust axis with the Orbiter center-of-mass is required, since concurrent firing of the Orbiter's RCS thrusters for the purpose of nulling perturbing moments will not be permitted. This restriction is necessary to preclude possible exhaust interference with contamination measurements of the test specimen.

Before initiating the test series the experiment platform will be raised from the stowed position in the cargo bay to a height of 2 to 6 ft (depending on the thruster type) above the door mold line and locked in place. This is necessary in order to (a) eliminate any influence on the rocket contamination measurements due to traces of other contaminants surrounding the Orbiter hull in a thin layer, (b) to reduce the effect of cargo bay surfaces on the rocket exhaust flow field, and (c) to avoid interference with, and contamination of other payloads carried by the Orbiter.

Details of the experiment design, the platform dimensions and layout, the test equipment and the plume mapping procedure remain to be defined. A test planning and design study intended to provide such data will be initiated by JPL in October 1977 under NASA/OAST funding. Several man-years of study effort are projected.

3.3 Orbit

The principal requirement is to perform the test at altitudes above the sensible atmosphere. This means that, in general, any Orbiter flight of opportunity with sufficient spare payload weight and cargo bay space could be used to accommodate the propulsion tests. Orbit characteristics are generally of no concern.

3.4 Test Data Acquisition

Each test will be performed in a preprogrammed sequence, with the Orbiter crew only performing the tasks of raising the test platform to the required position above the cargo bay, turning the propulsion system on and off and monitoring the test while in progress. Test data will be recorded onboard the Orbiter and returned to the ground for post-flight analysis. A requirement for on-orbit checkout and trouble-shooting by the Orbiter crew of the specialized test equipment and the propulsion system specimen is not envisioned. Except for the required power source, test data recording and remote control circuits and displays no major electrical interface with the Orbiter system will be required. The experiment is largely self-contained.

4.0 ASSESSMENT FOR STS FLIGHT

The Shuttle Orbiter provides a convenient testbed for this experiment, facilitating realistic rocket exhaust measurements under vacuum conditions, and easily accommodating hundreds of pounds of test equipment and the propulsion test specimen at a low transportation cost. By the intended preprogrammed, automatic sequencing of thruster firing and measurement procedures this nearly self-contained experiment only requires a minimum of crew involvement. Principal areas of concern are:

- Provision of safeguards against possible hazards inherent in carrying appreciable amounts of propellants in the cargo bay and firing propulsion systems in close proximity to Orbiter structures and other payload elements.
- Availability of adequate power (3 to 4 KW) for operating the large (30 cm) ion thruster over an extended period.
- Dissipation of waste heat, e.g., about 1 KW prior to and during operation of the large ion thruster. This may be of critical concern because of the tight thermal control required for the quartz crystal microbalance being used in the test.
- Maintenance of the Orbiter's attitude when operating large 800 to 1000 lb rockets if the thrust axis is not accurately aligned with the center of mass. Two-axis gimballing of these rockets may be required to minimize perturbing moments.

4.1 Experiment Considerations

4.1.1 Safety

Safeguards are necessary to guard against inadvertant firing of the test rockets before the experiment platform has been erected to the operating position; against exposure of sensitive payloads to the test rocket exhaust plume; against the possibility of spilling corrosive, combustible, and toxic propellants into the cargo bay and against heat from the large chemical propulsion thrusters or the 3 KW ion engine affecting sensitive equipment in the cargo bay. Some of these hazards can be reduced to an acceptable level by appropriate design of the test facility, by interlock provisions, redundant safety features and thruster enclosures, and by safe test operating procedures through adequate crew training. Monitoring displays and caution/warning indicators at crew stations also are essential.

4.1.2 Monitoring of Background Contamination

The possible effect of contaminants in the Orbiter environment on sensitive measurements of the thruster exhaust plume can be determined by scanning the detectors through the region surrounding the test specimen before initiating the firing test. Any noticeable background levels can then be subtracted from the contaminant flow measured during the test operation.

Time variations of contaminant distribution should also be monitored to detect such effects as decay of exhaust concentrations after Orbiter RCS system firings.

4.1.3 Preprogrammed vs. Adaptive Test Program

As currently envisioned by test planners, the thruster firing and plume mapping operations will be conducted in a pre-programmed sequence. Different sequences will be designed for the different propulsion systems to be tested. This approach is simple and reliable, requires little or no participation by the Orbiter crew, and minimizes communication with investigators on the ground. All test results will be recorded onboard the Orbiter for post-flight processing and analysis.

This approach, favored because of its simplicity and low cost implications, however, does not permit the use of adaptive techniques where the experimental sequence can be influenced by the outcome of preceding steps and the capacity of the human operator for improvisation, factors generally considered a principal asset when planning Shuttle-borne experiments.

Further study of alternate approaches is recommended to determine:

- Whether a fully preprogrammed test meets all safety requirements.
- How much cost and complexity is saved by adopting a preprogrammed procedure.
- Whether the cost of repeating an unsuccessful or incomplete test on another Shuttle flight of opportunity is sufficiently small to justify the economical but more failure-prone pre-programming approach.
- How long a waiting period, on the average, is to be expected between Shuttle flights of opportunity based on current traffic models.

4.2 STS Integration Considerations

4.2.1 Multi-Purpose vs. Application-Tailored Test Platforms

The test program includes propulsion systems of great diversity and thrust level ranging from a 1-millipound (8cm) ion thruster to a 1000-pound solid rocket. Dimensions and weights of the thruster specimens, complexity of the system components and subtlety of plume mapping techniques similarly vary over a wide range.

The cost trade-off between a single multi-purpose test platform for this diversity of test objects and developing test platforms tailored to different classes of test objects requires further study.

Test equipment commonality includes items such as:

- Platform and deployment mechanism
- 2-axis gimbal mount for thrust vectoring of large rockets, including control electronics.
- Scanning boom(s) for plume mapping instruments and detectors (not necessarily required).
- Data handling interface equipment
- Test sequence programmer

Support Equipment tailored to individual test items will include the following:

- Mounting and support brackets
- Power Supplies
- Thermal control equipment, shields and radiators
- Data acquisition and data handling modules
- Propulsion system control circuits

4.2.2 Conceptual Layout

Because of the very preliminary status of the test program definition, the layout of the test facility can be presented only in rough outline. However, from the foregoing discussion of test objectives and procedures the following general design requirements and preliminary configuration aspects are apparent:

(1) The support platform must be designed for stowage on a standard test rack and for deployment to a height of about 5 to 8 ft. above the stowed position. A scissors-type deployment linkage is a promising candidate. This deployment mechanism may be required to permit locking the platform at several discrete positions above the stowed position.

(2) The platform must be able to accommodate the largest rockets contemplated in the program, i.e., the 1000 lb_f solid motor with a typical length of 30 to 40 inches and a variety of propellant storage and feed systems.

(3) A two-axis gimbal mount may be required in some instances to align the thrust vector with the Orbiter's center of mass. However, alignment accuracy is modest (probably $\pm$ 1 degree).

(4) The preferred location of the platform center in X-direction is close to the Orbiter C.M. (typically, within $\pm$ 5 ft. of the C.M.) at least for the high-thrust propulsion systems in the test series. This permits thrust vector orientation within about 30 degrees from the Z-axis and, thus, minimizes plume impingement on Orbiter structures or on objects in the cargo bay.

(5) An articulated boom may be required to scan contamination sensors along and across the thruster exhaust plume. The diversity of thruster sizes and exhaust plume characteristics calls for large variations of scan motions and coverage range which must be accommodated by the boom design. These booms must be safely stowed prior to platform deployment. (Note: According to information received from JPL's Propulsion System Division, the maneuverable scanning boom may be omitted to reduce cost and complexity of the experiment.)

(6) As a safety provision, the entire deployable experiment platform must be jettisonable if the retractor mechanism fails to operate. The deployable scanning boom also must have a jettison provision.

Figure 4.2-1 shows a conceptual layout of the experiment platform in stowed and deployed positions. The 800 lb bipropellant rocket and propellant tanks (Experiment 2) are shown as a sample propulsion specimen.

In this layout it is assumed that some other cargo occupies the rear portion of the Orbiter's cargo bay and extends forward just beyond the center of mass (assumed at Station 1150). As illustrated, the propulsion test platform is placed between cargo bay stations 980 and 1070 forward of the center of mass. Thus the thruster must be installed at a forward tilt angle (approximately 30 degrees) from the Z-axis to achieve near-zero thrust vector offset from the C.M. A two-axis thruster gimbal mount is shown in the drawing which will be used for in-flight thrust axis alignment if necessary. (Further analysis is required for a specific platform installtion and for specifics of the Orbiter mission to determine whether this added complexity might not be avoidable).

A two-axis gimballed test equipment deployment boom of the STEM type, attached on the starboard side of the platform (to avoid interference with the Remote Manipulator Arm) is provided for mapping the exhaust contamination flow field in three dimensions to distances of 8 to 10 ft.

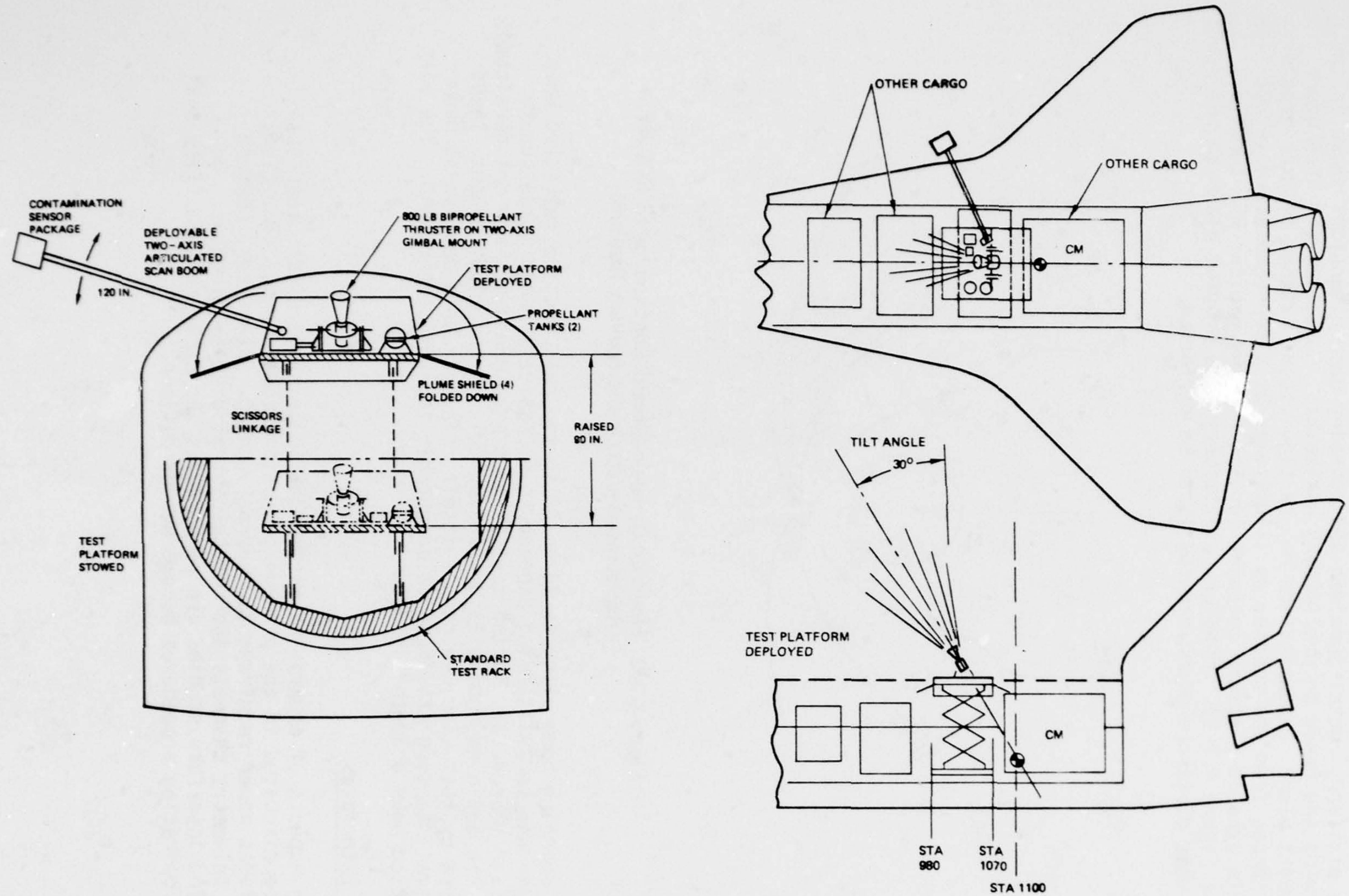

Figure 4.2-1. Conceptual Layout of Propulsion Test Platform
(Example 800 lb Bipropellant Thruster Experiment)

The platform, rocket specimen and scanning boom dimensions shown in the drawing are based on preliminary estimates. Actually, the area where exhaust plume contamination may be significant extends to much greater distances from the nozzle, as illustrated by a representative set of flow field dynamic pressure contours for the 800 lb_f Orbiter RCS thrusters (see Figure 4.2-2). However, preliminary test program plans do not specify to what distances plume mapping is to be performed.

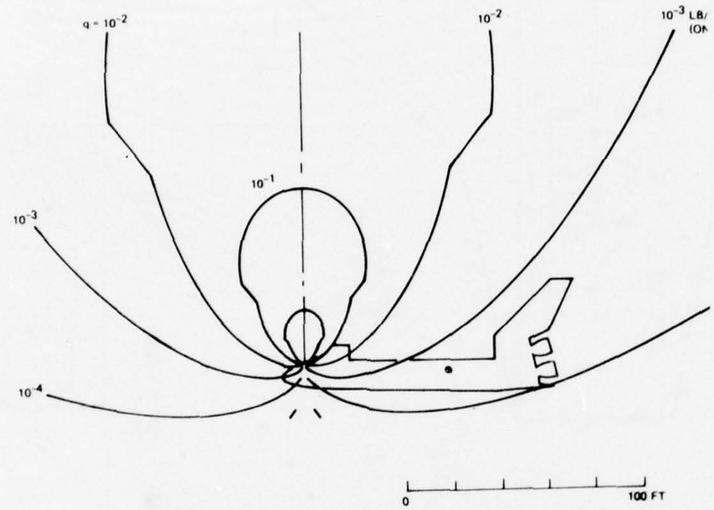

Figure 4.2-2. Flowfield Dynamic Pressure Contours for +Z RCS Jet (Representative 800 lb Bipropellant Thruster)

The scanning boom with its 20 to 30 lb tip mass must be retracted when not in use to avoid being exposed to excessive dynamic loads due to Orbiter maneuvers. During test firings such maneuvers are not permitted, and the deployed boom will be exposed only to the minor dynamic loads caused by test specimen thrust itself. Thus, a thin deployment boom of small bending stiffness is acceptable for this experiment. The estimated boom diameter is about 1/2 inch.

4.2.3 STS Interfaces

Many aspects of experiment accommodation on the Orbiter other than those discussed in the preceding sections remain undefined. This includes power requirements, thermal control interfaces, command and telemetry channels and remote control display circuits. Detailed interface studies are required as the experiment facility and its operating procedures become better defined.

4.2.4 Cost Considerations

Factors that aid in conducting this experiment at low cost have been discussed in the context of test facility design and operation. In summary, the following cost-saving considerations apply:

- Reuse of the facility for different propulsion specimens multi-purpose design reduces equipment and pre-flight preparation cost.
- Minimum demand on crew participation saves training cost and avoids interference with other crew duties.
- Short total operating time allows flexibility of scheduling during the mission and avoids interference with other flight objectives.
- Onboard storage of test data minimizes ground communication requirements.
- The experiment can take advantage of Shuttle flights of opportunity since mission characteristics are of little concern. This tends to reduce transportation cost.
- Many components of the test facility can be adapted from other flight programs and from propulsion test facilities on the ground.
- Weight and space requirements are reasonably small (estimated weight about 1000 lb, installation length about 5 ft on portion of test rack) to permit inexpensive STS transportation ($300 K to 400 K).

5.0 RECOMMENDATIONS AND REMARKS

Since information on this experiment series was too sketchy for a detailed assessment, it is recommended as a next step (even before the forthcoming experiment definition study by JPL is completed) that principal data on thruster dimensions, weights, propellant mass, plume characteristics, experiment power requirements, heat dissaipation, etc. be compiled as soon as possible and evaluated from an STS interface definition and experiment integration standpoint. This will aid in making preliminary estimates on STS integration, transportation and experiment operation costs.

Cost benefits aspects of the multi-purpose experiment facility design vs. tailored facility designs require further study as the diversity of test equipment to be used are better defined. Secondly, cost benefit tradeoffs between fully preprogrammed and adaptive test procedures are important as they affect crew functions and data handling and ground-to-Orbiter communication requirements.

ELECTRON INJECTION LIMITS

1.0 EXPERIMENT IDENTIFICATION

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2.0 REQUIREMENT BACKGROUND

Response to STS Utilization Presentation (Response Sheet No. 24)

Task Number JON 8809 1901 (AFWL).

3.0 EXPERIMENT APPROACH

The investigator wishes to measure the characteristics of trapping and dumping of high energy charged particles in the low altitude radiation belt. Several theories have been advanced which treat the behavior of the particles as they mirror along the magnetic field lines. In particular, it has been shown that one should expect that the number of particles that can be trapped on a given geomagnetic flux tube is limited by plasma wave instabilities. It is intended to inject particles into the geomagnetic flux tubes until this trapping limit is reached and then to examine the decay of the trapped particles. The injections are to be made in the southern hemisphere in order to take advantage of the weaker magnetic fields in that hemisphere and in order to be nearer to the equatorial trapping region. During the time of the injection, it is desired that the Shuttle be flying in such a fashion that it tracks the drift movement of the injected particles in the flux tube. Ground-based measurements in the northern hemisphere should be made to observe those particles that fail to be trapped and are precipitated out the other end of the flux tube. If trapping occurs, these ground stations could also follow the time history of the dumping.

In order to fill the flux tube with quasi-trapped particles, it will be necessary to try to inject the particles at pitch angles close to ninety degrees. The investigator has calculated that this will allow him to fill the flux tube with particles having equatorial pitch angles greater than 20°. The ideal experiment would be able to fill up to 0° pitch angle, and if the Shuttle could fly at higher altitudes, lower pitch angles could be injected.

The following instruments are to be mounted in the cargo bay of the STS: An electron accelerator with a capability of at least several amperes at 40 kilovolts is needed. The best system would to use a 400 kilovolt electron accelerator operating at 4 amps, however, such a system may be hard to design. In addition to the accelerator, there is a need for several diagnostic instruments in order to look for the particles that are leaving the geomagnetic flux tube. At the 40 keV energy level, electrostatic analyzers could be used, but at the 400 keV level, either solid state detectors or scintillators would be required. Since one of the points of interest is the pitch angle of the dumping particles, it will be necessary either to scan a single detector in pitch angle or to have several detectors mounted at different pitch angles. Since the whole experiment may be accomplished in a short time, the multiple detector mode would be preferable. The electrons in the flux tube drift eastward as they travel along the magnetic field lines. It is hoped that the Shuttle will be able to move at roughly the same velocity as this drift velocity, however, since the entire flux tube even at 400 keV, is only on the order of 40 meters in diameter, there may be difficulty locating the tube if the velocity of the Shuttle is not exactly the same as the drift velocity. For this reason, the investigator desires that several subsatellites be launched from the Shuttle into orbits close to that of the Shuttle so that during the experiment, the instruments mounted on the subsatellites will be able to make survey measurements of the dumping particles at several distances from the Shuttle. The subsatellite instrumentation would include particle detectors and magnetometers.

In the discussions that follow, the 40 keV system will be given greater emphasis because this system is closer to realization for Shuttle flights. The 400 keV system is not as definite even though it would be the better system to test the trapping limits.

The 40 keV accelerator has dimensions 1m x 1m on the base and is 2.57 m high. Its weight without energy storage capacitor banks is 328 kg. Each of the energetic particle detectors would weigh no more than 2 kg each and would fit within a box 10 cm by 10 cm on the base and 10 cm high. The magnetometers would weigh less than 1 kg and fit in a 3 cm cube box. The total number of particle detectors is TBD. However, there would probably be on the order of 10 to 20 units. The size of the subsatellites to carry the detectors could be rather small -- 50 cm in diameter and 10 cm high weighing no more than 50 kg each, including batteries and telemetry system.

The size of a 400 keV accelerator system is TBD, however, it would be significantly larger than the 40 keV accelerator. As a guess, one could say that it would be 2 m by 3 m on the base and 2.5 to 3 m high. The factors controlling its size would be the power dissipation, the exit beam divergence, and the accelerator high voltage. The weight would be no more than 500 to 1000 kg. The size of the diagnostic instrumentation would not change significantly for the higher energies.

The pointing requirements for either accelerator system would be on the order of 1° to 2°. The requirements for the diagnostic detectors would be the same. These pointing requirements are with respect to the geomagnetic field, and it is required that the orientation with respect to the magnetic field be held to these limits during the firing of the accelerator. The diagnostics mounted on subsatellites should be held so that at least one instrument is looking at 0° pitch angles, one at about 45° and a third at 90° pitch angles.

During the operation of the accelerator, the power requirements are extreme. For the 40 keV beam, the power requirements are on the order of 200 kW, or for the 400 keV beam, the power required is about 1.6 megawatts. Such power levels cannot be obtained directly from the STS and therefore, the investigator will provide specifications for a capacitor power bank that can be charged up slowly and fired rapidly to deliver the necessary instantaneous power. These capacitor banks should have energy storage capacity on the order of 32 kilojoules for the 40 kV system or 320 kJ for the 400 kV system. The details of the power system will be discussed under section 4.1. The diagnostic particle detectors would need about 2 to 5 W each. The magnetometers would take less than 2 W. All of the power can be delivered at 28 volts. The duty cycle for the experiment is extremely short -- no more than a few seconds for a given orbit. The number of times that the entire experiment is repeated depends upon orbital considerations and the possibility of finding an orbit that would be able to track a geomagnetic flux tub for several seconds. As a first estimate, it can be assumed that the experiment could be repeated once per day.

The accelerator will have regions within it that operate at extremely high temperature and that must dissipate large amounts of thermal energy. On the other hand, some of the control circuitry must be maintained in the temperature range from 0° to 40°C. The entire thermal design of the accelerator must be handled during the first stages of its overall design. The 400 kV accelerator has the more severe thermal problems of the two system because of its high energy dissipation. The diagnostics should be able to operate in the temperature range from -10° to 40°C. The accelerator will require a good vacuum environment for proper operation. No conductive volatile chemicals can be outgassed when the accelerator is exposed. The diagnostic instruments are sensitive to EMI, but when they are flown on subsatellites, there should be no problem.

The accelerator will produce a very high level of EMI for the short time that it is operating. The electron beam when it leaves the accelerator should not come in contact with any part of the vehicle. There is a possible problem during the operation of the accelerator if there is not enough electrical conducting surface exposed to the ambient ionospheric plasma. If the thermal electron current back to the Shuttle is not as large as the electron beam current leaving the accelerator, then the vehicle will begin to charge to large positive voltages. The investigator is considering adding a low voltage ion accelerator to his instrumentation in order to provide neutralization of the vehicle during the use of the electron accelerators. This ion gun would be no larger than the size of one of the energetic particle detectors, and would be mounted close to or on the electron accelerator. Some of the other problems associated with the ejection of large electron fluxes will be discussed in section 4.1.

The particle and magnetic field diagnostic instrumentation are standard spacecraft instruments that have been flown often. A 40 keV accelerator suitable for space flight is being developed for Spacelab flights by a joint Japan/USA team of scientists. This design should be available to the scientific community by 1980. A 400 keV accelerator exists for laboratory use, but at the present time, there has been no effort to modify it for operation in space.

When the experiment is flown, the investigator would like to refly several times in order to perfect the experimental technique and explore the full range of trapping stability behavior.

4.0 ASSESSMENT FOR STS FLIGHT

This experiment is difficult both in the design of the instrumentation and in the number of requirements that are put on the STS. If the instrumentation can be successfully built within the size, weight and power limits stated above, it should be able to be accommodated by the STS in the cargo bay.

There are several problem areas that would need to be studied before integration could take place: (1) Provision must be made for the short-term high power demands of the accelerator, (2) there must be an examination of the thermal problems associated with the accelerator, (3) the radiation environment of the Shuttle must be studied for the case when the accelerator beam accidentally hits part of the Shuttle, (4) the mission flight operations must be done carefully so that the desired tracking of a flux tube for the duration of the experiment can be accomplished, and (5) the tracking and telemetry reception of the several subsatellites must be studied.

4.1 Experiment Considerations

4.1.1 The design of a 400 kV, 4 amp accelerator suitable for spaceflight is the major problem to be overcome if the trapping is to be studied at high energy. The STS/Spacelab AMPS payload phase B study contracts have investigated the problem of mounting a 40kV, 2 to 10 amp accelerator on the STS. An accelerator similar to the AMPS one is going to be built by the Japanese for flight on the Orbiter. The two major areas of concern are power and thermal requirements on the Orbiter. In the Injection Limit Experiment, the accelerator will be fired for times on the order of microseconds to at most, a few seconds at the highest possible currents so that the maximum number of electrons can be injected along a given flux tube. This means that there are severe demands on the amount of power that has to be delivered to the accelerator during these times. The standard solution to this problem has been to supply, as support equipment, a large capacitor bank and several power converters. These are to take the 28 volt STS power and transform it first to 300 volts to charge the capacitor bank and then, take the 300 volt, high discharge current from the capacitor and transform it to a high current, 40 kV power level. The current state-of-the-art in power processing for pulsed sources is about 100 kilowatts per unit. The proposed 40 keV accelerator is within the capability of a combination of several of these power processing units. The 400 keV accelerator appears to be beyond the state-of-the-art for space qualified power processors. The capacitor bank for either system is probably less of a problem. Multimegajoule banks are in standard operation in the laboratory, and the conversion to spaceflight instrumentation is not complicated since the banks can be hermetically sealed. The only problem with the banks is the large size and weight. For the AMPS study it was found that typical capacitor bank weights and sizes were about 10 g/J and 2.1 cm^3/J. A 32 kJ bank has a weight of 320 kg and a size of 6.7×10^4 cm^3. The capacitor banks must be connected to the STS/Spacelab fluid cooling lines to dissipate the heat that is developed during the charging and discharging cycles. The size and weight of the 400 keV accelerator would be ten times as large. Either capacitor bank can be accommodated on a standard Spacelab pallet. For the 40 keV accelerator, it would be possible to mount both the capacitor bank and the accelerator with power processors on a single Spacelab pallet. It appears that the 400 keV accelerator would probably require two pallets, one for the accelerator and one for the bank and processors.

If a 400 kilovolt acceleration voltage is used, special care must be taken with the design so that the accelerator does not undergo high voltage breakdown. Probably the best vacuum that could be obtained even at the highest Shuttle altitude of 450 km would be on the order of 10^{-7} torr in the cargo bay, however, the pumping speed would be good with proper design.

4.1.2 Operation Restrictions

The required altitude for performing the experiment is as high as possible consistent with not having to pay for an OMS kit. This limits the altitude to about 450 km for a payload weight of 14,000 kg. The inclination should be about 28° or lower so that during the experiment, the Shuttle is traveling roughly west to east at a velocity on the order of 1 km/sec. Further, it is required that during the experiment, the Orbiter be able to hold its orientation with respect to the geomagnetic field direction. This last requirement is not serious since it is hoped that during the experiment, the orbital motion of the Shuttle shall be such that the geomagnetic flux tube associated with the injected electrons remains fixed with respect to the accelerator and the Shuttle. Therefore, the Shuttle can meet the experiment requirements by maintaining a position fixed in earth-centered coordinates. The problem of knowing when the Shuttle is flying so that the flux tube is fixed with respect to the accelerator must be solved by the investigator using maps of the geomagnetic field and his own magnetometer data. There is a possibility that the investigator would ask for small changes in the orbital altitude in order to synchronize with a flux tube before the beginning of the experiment. These changes can be consistent with baseline Orbiter capabilities for orbit maneuvers.

During the operation of the accelerator, care should be taken that the beam does not strike the Shuttle. This is a problem for this experiment because the investigator desires to inject at near to 90° pitch angle. This means that the electron beam will begin circling the magnetic field line in such a fashion that it returns to its point of origin in one cyclotron period which is a few microseconds at most. Hence, the beam will return to within a few centimeters of the spot on the Shuttle where it was emitted. It will be necessary to give the beam a sufficiently large velocity parallel to the magnetic field line to avoid hitting the Shuttle. This means that the pitch angle cannot be exactly 90°. Should the beam hit the Shuttle, there would be a burst of x-rays which could prove damaging to the crew members. Therefore, the investigator will have to provide a suitable safety margin in pitch angle. This radiation problem has been calculated for a 40 keV electron beam as one amp, the cumulative dose of several tens of seconds could cause damage to the crew. The problem for the 400 keV beam at 4 amps would be much more severe, and the danger to crew would be high.

The need for several subsatellites to locate the returning electrons requires that each be launched, tracked and monitored. The AMPS study has shown that simple spring-loading launching platforms can be provided which will satisfy the requirements of this experiment. The tracking and data reception of multiple subsatellites is a somewhat more difficult problem in that the STS provides only one detached payload telemetry link. This means that the subsatellites must be equipped with data tape recorders. During the experiment, they can record the data to be played back at a later time. Also, the subsatellites can be commanded into operation only one at a time, and tracked one at a time. The tracking may have to be done by K-band radar, although it may be sufficiently precise to locate the subsatellites by S-band. The tracking requirements are that the position of the subsatellite must be known to about 40 meters at distances up to several kilometers from the Shuttle. The detached payload telemetry rate is limited to 16 kbps. This is no problem as long as the data is on tape and can be played back at the desired rate.

If there are coordinated ground based observations in the northern hemisphere, then the Shuttle should be in voice communication with the ground observatories. The crew would be responsible for the subsatellites. The accelerator control panel should be about 60 x 30 cm and can be placed in the aft flight deck or in the Spacelab module. The amount of data reduction that will be done in orbit is minimal. The calculations of the position of the magnetic flux tube can be done on the ground. If multiple subsatellites are used, the data from each would be relayed to the ground, in sequence, for analysis after the firing of the accelerator. The total amount of data would be on the order of 50 kilobits per experiment sequence for each of the subsatellites, and up to 10 megabits per sequence for the accelerator status monitoring and control function. None of the individual data rates or total data requirements pose any problem for the STS system or for the Spacelab CDMS if it is used.

The accelerator and the subsatellite instrumentation can only be operated in a vacuum and therefore, only limited housekeeping functions can be checked during the integration period. The status of the particle detectors could be monitored, the magnetometers can be calibrated, and the status of the accelerator firing circuits could be checked. But the actual operation of the accelerator with high voltage can only be done before integration and after launch. The accelerator should have a cover to prevent dust falling into the large entrance aperture.

4.1.3 Experiment Support Equipment

The experiment needs the STS detached payload communications link. It needs a set of capacitor energy storage banks and power converters to supply the acceleration voltage and power. (Battery storage of accelerator energy has also been done successfully and could be considered for the 40 keV systems, but the 400 keV system would not be able to draw the necessary power through the internal resistance of the batteries.) Since there are several kinds of experiments including other accelerator experiments that have similar requirements for large power surges, it is open to negotiation, whether the storage banks should be considered as part of the investigator's responsibility or whether they should be considered part of the flight support equipment.

The necessary subsatellites can be easily mounted on the STS and deployed by a spring ejection mechanism. They would be spin stabilized and should have 20 to 30 watt-hr batteries. They would need tape recorders and an S-band telemetry systems.

There must be a magnetometer reference system on the Shuttle so that the accelerator can be pointed with respect to the geomagnetic field with 1 or 2 degree accuracy.

The experiment needs the fluid cooling loop for both the accelerator and the capacitor banks. The banks could possibly sit on the cooling plates if the Spacelab pallets are used to support the instrumentation, but the accelerator will need fluid cooling inside the body of the instrument. Spacelab does provide an experiment heat exchanger that can be used on a pallet. The instantaneous thermal generation of the accelerator would exceed the capabilities of the heat exchanger and of the STS cooling loop, but there can be sufficient thermal lag in the body of the accelerator so that the load on the cooling system would approach average heat dissipation of the accelerator. Most of the power supplied to the accelerator is transferred to the electron beam which leaves the vehicle.

4.1.4 Experiment Cost Considerations

This will be an expensive experiment to develop unless use can be made of an existing accelerator such as the Japanese one.

4.2 STP Integration Considerations

The integration of this experiment into the STS will be difficult. The minimal experimental system with the 40 kV accelerator, the energy storage banks and the particle and magnetic field diagnostics will require a mounting area equivalent to one full Spacelab pallet. Cooling to the accelerator will have to be provided, and a launch mechanism for a subsatellite would have to be built.

The delivery lead time would be directly related to the ultimate complexity of the accelerator system. At the present time, a 40 kV accelerator design would become available in the early 1980's. A 400 kV accelerator, if design were to start soon, could possibly be available in five to seven years. The energy storage banks and the subsatellite instrumentation could be delivered within two years after the start of the program.

There would be a need for testing of the accelerator system in a large vacuum chamber in order to determine its operating characteristics and also its production of EMI during the firing cycle. These measurements of EMI are necessary in order to determine if they might have some adverse effects on the STS avionics.

Figure 4-1 shows a sketch of the 40 kV accelerator system including capacitor energy storage banks mounted on a Spacelab pallet. Figure 4-2 shows a sketch of one possible configuration of a subsatellite that could be used in this experiment.

The following items must be purchased as optional STS services:

(1) Possible retrieval of the subsatellites. A cost tradeoff of retrieval vs. expended subsatellites should be performed.

(2) Heat exchangers for the accelerator and cold plates for the energy storage banks must be provided. Spacelab pallets could be used to mount the experimental hardware. Spacelab CDMS, or hardwiring into the Orbiter data system and aft flight deck, is necessary for control of the accelerator.

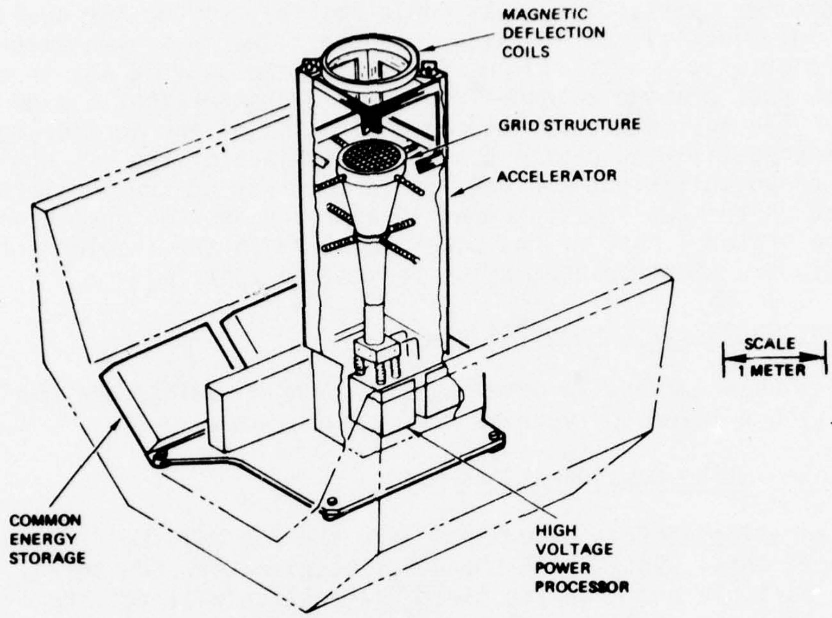

Figure 4-1. 40 Kilovolt Electron Accelerator Mounted on the Spacelab Pallet Together with its Energy Storage Banks

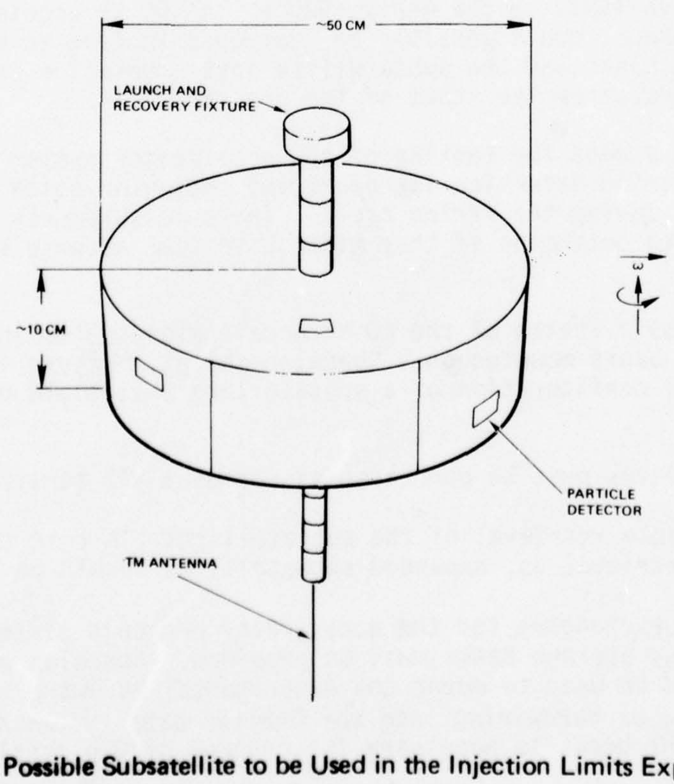

Figure 4-2. A Possible Subsatellite to be Used in the Injection Limits Experiment

(3) Mission planning support will be necessary in order to determine when the electron beam can be fired into a magnetic flux tube moving with the Shuttle or subsatellite.

(4) More than one day will be required to adequately perform this experiment.

5.0 RECOMMENDATIONS AND REMARKS

(1) This is a difficult and complex experiment requiring considerable use of STS facilities. It can only be performed during an STS sortie mission. While the integration of the experiment would be difficult, there seems to be no reason why it cannot be done. Its requirements are within the capabilities of the STS if a high power processor and energy storage bank are added either as flight support equipment or as experimenter provided instrumentation.

(2) It is recommended that the investigator consider the 40 kV accelerator at least for his first attempts at doing the experiment. The 400 kV accelerator, while having theoretical advantages in checking the trapping limit theory, appears to be beyond the present state-of-the-art for spaceflight qualified accelerators.

(3) It is recommended that the investigator seriously study the tradeoff between adding OMS kits to increase the orbit altitude and the ability to fill the trapping pitch angles only to 20°. If the majority of the injected artificial particles are at 20° pitch angles, there may result a purely artificial dumping mechanism that runs on the non-uniform distribution of the total particle population in phase space, i.e., the discontinuity of pitch angle distribution at 20° will drive the instability. For these reasons, it would be much better to orbit the Shuttle at much higher altitudes so that the particles can be injected at pitch angles nearer 0°. It may be that the cost of OMS kits may be small compared to the overall cost of doing the experiment.

(4) The investigator should verify that it is possible for the Shuttle or a Shuttle launched subsatellite to match the eastward drift velocity of the injected particles. He should also verify some experimental method of ascertaining when the tracking of the drift tubes can be done. There will be limits on the total number of times that the accelerator can be fired during a sortie mission because of the large amount of energy dissipated in each firing. Drift tube tracking may require instrumentation or ground support that has not been considered in the above assessment.

(5) There is a definite radiation hazard to the crew if a 2 amp, 40 keV electron beam strikes the Orbiter. Beams fired at 90° pitch angle should hit the accelerator or the Shuttle. The investigator must arrange the experimental geometry so that the beams do not hit the Shuttle.

DYNAMIC POWER SYSTEM

1.0 EXPERIMENT IDENTIFICATION

Robert C. Brouns, Principal Investigator
ERDA/Nuclear Research and Applications Division
Washington, D. C. 20545

Sponsor Agency: AFAPL/POE

2.0 REQUIREMENT BACKGROUND

Response to STS Utilization Presentation (Response No. 11)

Work Unit Number: 682J0701

KIPS Launch Vehicle Integration Characteristics, Sundstrand Energy Systems

3.0 EXPERIMENT APPROACH

Demonstrate the performance and operating characteristics of the dynamic power system in the space environment. Test will provide data to guide the development and demonstrate the feasibility of this technology.

4.0 ASSESSMENT FOR STS FLIGHT

Both the BIPS (Brayton cycle) and KIPS (organic Rankine cycle) are being considered for testing. As both systems derive their primary energy from isotope heat sources, and are presently designed for about the same power level, the assessment statements in this section apply to either system.

This experiment should be conducted on a free flyer, possibly the LDEF or the Multimission Modular Spacecraft. There are two concerns associated with STS launch:

1) The use of isotope heat sources requires special safety and handling procedures.
2) The thermal energy produced by the power system exceeds the heat rejection capability of the Orbiter both in-flight and prelaunch.

4.1 Experiment Considerations

4.1.1 Attitude Control and Pointing

There are no special pointing requirements.

4.1.2 Thermal

The heat generated by the power system exceeds the rejection capability of the Orbiter. Additional rejection capability or electrical loads will have to be provided. This problem is particularly significant during planned prelaunch operation. The heat generated is approximately 7000 W while the rejection capability is 1500 W during prelaunch and ascent. Serious consideration should be given to in-orbit startup of the system. If the system must be operated while in the Orbiter payload bay, consideration should also be given to reducing the amount of isotope used for early tests to keep within Orbiter heat rejection capability. A method must be developed for transferring this heat to the Orbiter thermal control system.

4.1.3 Safety

The safety requirements connected with having isotope power supplies in Shuttle payloads are being developed by NASA. In addition to the requirements, some design and operational suggestions will be promulgated. These should be considered in design of the experiment.

4.1.4 Design Suggestions

The radiator assembly planned is 5 ft. in diameter and 7 ft. in length, with significant empty space inside. The STS charging policy is based on length and weight. To reduce STS launch costs, the spacecraft and experiment equipment design and packaging should minimize the length and take advantage of the large diameter (15 ft.) Orbiter bay.

4.2 STS Integration Considerations

An isometric of the Dynamic Power System assembly is shown in Figure 4-1. The basic layout and the flight and ground support equipment are identified.

The final spacecraft and experiment system design should be influenced by the many deployment options and their costs. The experiment could be a co-passenger or power supply with IUS deployment or deployed with another stage on a mixed payload STS launch. The cost sharing tradeoffs will significantly influence length, packaging density and stage selection.

The above design suggestions and heat rejection considerations are applicable to any dynamic power supply system that uses an isotope heat source of about the same power level.

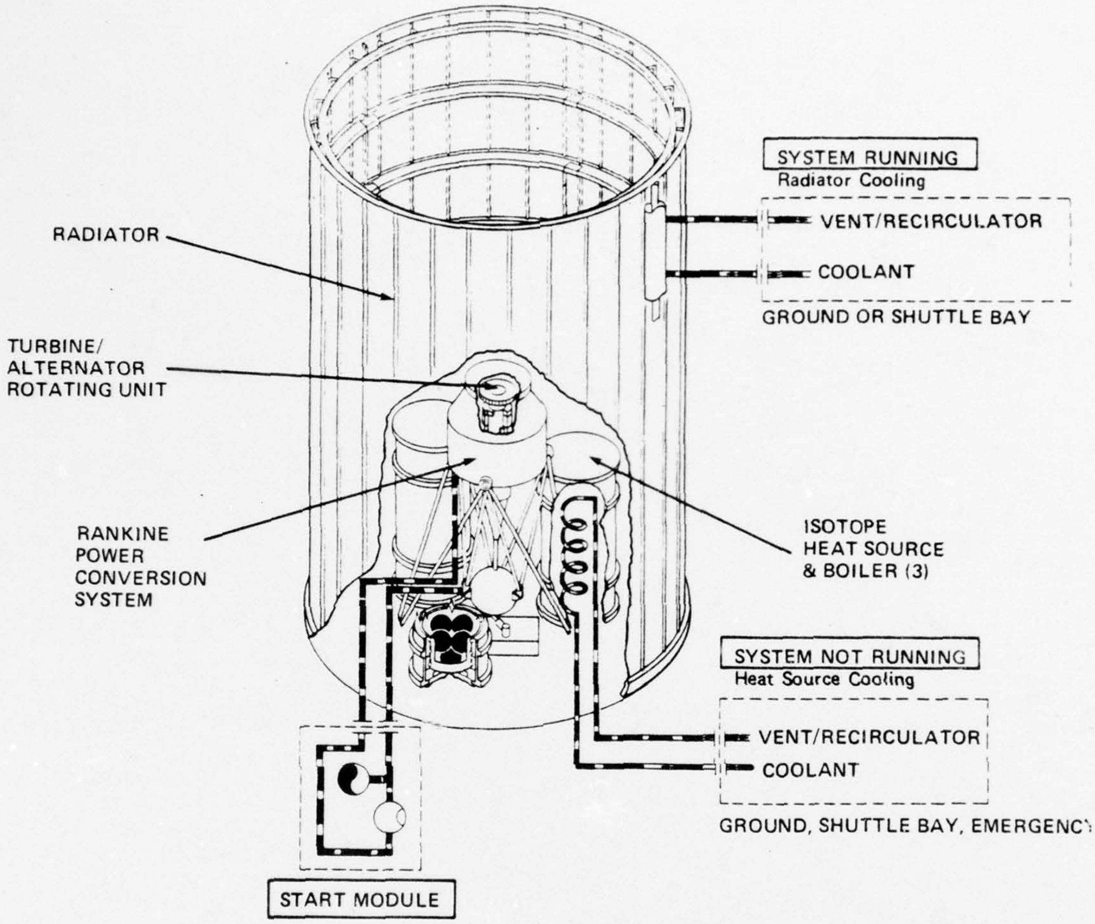

Figure 4-1. Dynamic Power Supply System

5.0 RECOMMENDATIONS

Consider launch costing model in basic spacecraft and experiment design and layout.

THERMAL ENERGY STORAGE EXPERIMENT

1.0 SOURCE

E. T. Mahefkey
AFAPL/POE-2, Aerospace Power division
Wright Patterson AFB, Ohio 45433

2.0 REQUIREMENT BACKGROUND

- Response to STS Utilization Presentation (Response Sheet No. 27)
- Identifying Number: Program Element 62203F, Project 31451900 and 63428 F/212 60310
- Sponsor Agency: AFAPL/POE-2 and SAMSO (SZSS)
- Related to Cryogenic Cooler Experiment (Response Sheet No. 28) and Cryogenic Radiator Experiment (Response Sheet No. 29)
- Ref. AFAPL-TR-77-10 "Demonstration Testing of a Vuilleumier Cryocooler with an Integral Heat Pipe/Thermal Energy Storage Unit", June '77

3.0 EXPERIMENT APPROACH

3.1 Background

The proposed experiment centers on in-space, zero-g testing of the performance of an eutectic salt energy storage device integral with a high-temperature liquid metal heat pipe. The system is designed for application with a Vuilleumier (VM) Cryocooler. The VM cooler requires a sustained high power (1-kW level) thermal input to its hot section. This can be supplied by a solar array when in sunlight, but stored thermal energy from a eutectic salt energy storage device is to be used during eclipse periods.

The flight experiment as defined by the principal investigator does not include the VM cooler but simulates the thermal energy consumption of that device by an appropriately designed heat sink. The VM cooler is the subject of a separate flight experiment which will be discussed elsewhere in this report.

The phase change processes in the Thermal Energy Storage Unit (TESU) and the liquid metal heat pipe integral with it may exhibit characteristics in the zero-g environment that are different from those demonstrated in the laboratory under 1g. Even very small differences may significantly affect the device's performance in the intended satellite application.

3.2 Experiment Objectives

Objectives of the flight experiment are:

1) Demonstration of the Heat Pipe/TESU system performance during steady state thermal cycling in zero-g.

2) Evaluation of zero-g start up of the high temperature liquid metal heat pipe from low temperature.

3) Evaluation of energy storage defects due to modifications of the eutectic salt freeze/melt pattern in the zero-g environment.

Post flight analysis will compare heat flow characteristics and temperatures at various points in the system as obtained from the flight test with those obtained in the laboratory to aid in further development and flight qualification of the device.

3.3 Experiment Design and Procedure

The schematic diagram shown in Figure 3.3-1 illustrates the integral Heat Pipe/TESU to be used in the flight test and the heat sink that simulates the VM cooler. The eutectic salt used in the TESU consists of a LiF-MgF_2-KF mixture with a melting point of 713° C. In steady state operation, the temperature varies over a small interval ($\pm$ 14°C) above and below the melting point. In the proposed flight test, the unit will be thermally charged by a thermostatically controlled electric heater with 300 W input power. The charge period is 40 minutes. The stored energy of 200 W-hr is then discharged into the heat sink through the liquid sodium heat pipe. The heat sink consists of a heat exchanger coupled to an appropriately placed radiator plate.

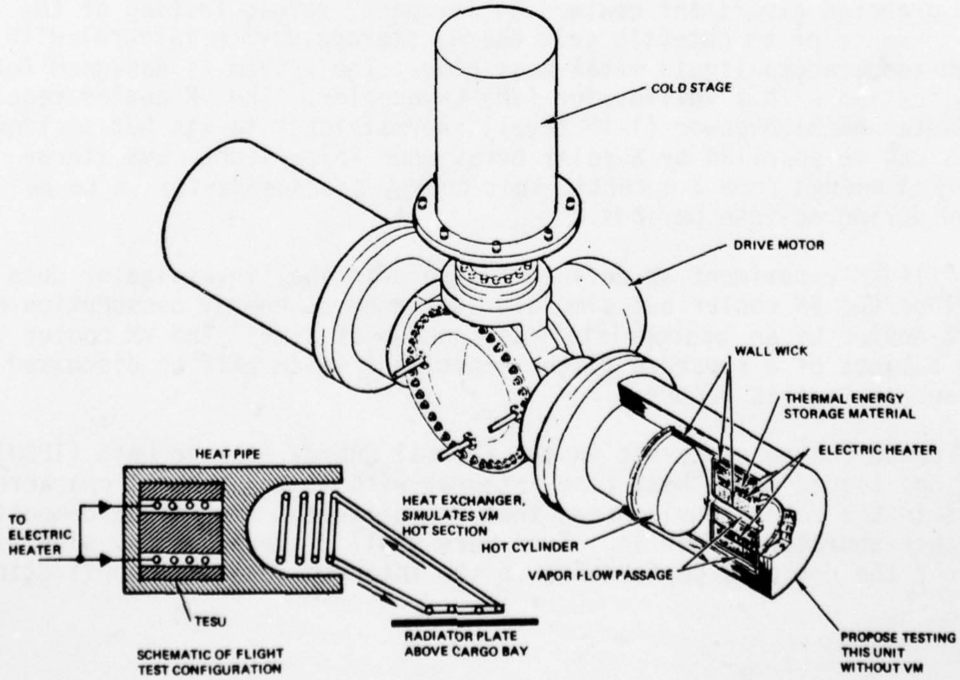

Figure 3.3-1. VM Cooler and TESU/Heat-Pipe System

The discharge period of the thermal cycle lasts for about 20 minutes. The cycle is then repeated. A small amount of stand by power is required to make up insulation losses prior to the discharge.

The thermal cycle duration may be extended to about 1.5 hours of observation. A total of at least 10 to 12 successive cycles under steady-state operating conditions is desired.

Various techniques and devices for controlling the heat dissipation through the heat sink are currently being considered, including louvers or other types of hinged radiation shields to cover the radiator during the charging process, or coupling to a pumped on-off liquid heat exchange loop that may be connected to the STS environmental control system. Ideally, no heat would flow from the TESU through the heat pipe into the heat sink during the charging cycle when the "heat valve" is shut off; the discharge begins when the TESU heater is turned off and the "heat valve" is opened.

Figure 3.3-2 illustrates several thermal cycles from the start up at cold temperature to the steady state condition. It is anticipated that temperature profiles obtained under zero-g conditions during the flight test (solid curve) do not differ greatly from those obtained in the laboratory (dashed curve). The main differences may be found in the start up phase in which the freeze/melt pattern may be significantly affected by the zero-g environment. Hot spots forming at the TESU/Heat Pipe interface due to irregular distribution of solidified salt under zero-g are of some concern. About 10 thermocouples placed at locations of main interest in the TESU/Heat Pipe system will provide data on the thermal profile. The data will be sampled at 2 minute intervals.

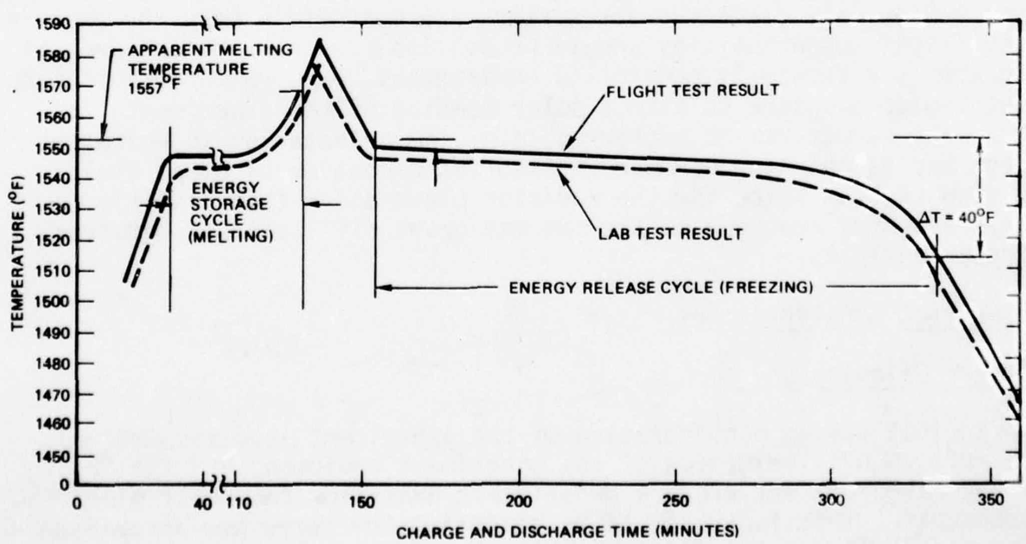

Figure 3.3-2 Thermal Cycle of TESU (Typical)

The principal investigation also has indicated an interest in making observations of system-in-flight performance under small acceleration loads, e.g., 0.1g., if achievable by Orbiter maneuvers (see below).

The specified test duration is about three days of which one day is allocated to achieving full thermal stabilization. No constraints in orbital characteristics have been identified, i.e., any Shuttle mission of sufficient duration would be suitable to carry this experiment.

The experiment can be fully automated and results recorded for post flight analysis. Crew participation is limited to turning the equipment on and off at the appropriate time. Override control of the automatic sequnce will also be required. One concern regarding safety is the possibility of a spill of molten eutectic salt, but this can be prevented by an emergency heater-shutoff provision.

The experiment is not sensitive to environmental conditions such as radiation, contamination and magnetic fields. However, any influences due to the thermal environment that might affect the sensitive equipment heat balance must be carefully avoided. Items of concern are avoidance of direct solar heating and any heat flux from surrounding structures, as well as the effectiveness of the thermal radiator.

The estimated weight of the experiment is about 45 lb, its dimensions are 1 ft diameter by 1 ft long (cylindrical). Average power requirements are 100 to 300 W. Data requirements remain to be defined. Also, a preferred mounting location has not been determined.

4.0 ASSESSMENT FOR STS FLIGHT

The experiment is well suited for STS accommodation at low cost because of its small size and weight and its moderate power requirements. It can be fully automated and requires almost no attention by the Orbiter crew. Since any orbit characteristics are compatible with the experiment, many flight opportunities should be available. However, one area of concern is a carefully controlled environment. An Orbiter orientation that avoids exposure to direct solar heating of the experiment or nearby structures is preferred, e.g., am orientation in which the cargo bay is pointing downward. Assuring a continuous clear field of view to open space for the radiator plate under this condition while avoiding exposure to the sun may prove difficult and requires further analysis.

4.1 Experiment Considerations

4.1.1 Orbiter Orientation

A principal design consideration of the experiment involves thermal control. Sun illumination of the experiment equipment and the inside of the cargo bay may produce undesirable heat flux into heat pipe/TESU components. This is avoidable by orienting the cargo bay in antisun direction. In low inclination orbits,the Orbiter Z-axis may be pointed at or near nadir for sun illumination avoidance, see Figure 4.1-1a. In sun-synchronous near polar orbits with ascending and descending nodes close to the terminator the best orientation would be one with the

Z-axis perpendicular to the orbit plane and X parallel to the velocity vector, see Figure 4.1-1b. To permit earth observation in this orbit while still avoiding direct sun illumination in the cargo bay, roll angles 45 to 60 degrees from the above described orientation are probably acceptable. This would make the experiment more compatible with mission objectives of other payload elements.

The radiator plate, with an area estimated as 0.5 to 1 ft^2, can be mounted parallel to the X-Z plane, above the location of the experiment equipment, with nearly unobstructed view of free space and no exposure to the sun. However, further study of radiator placement and pointing is necessary to take all pointing modes of a given mission profile into account.

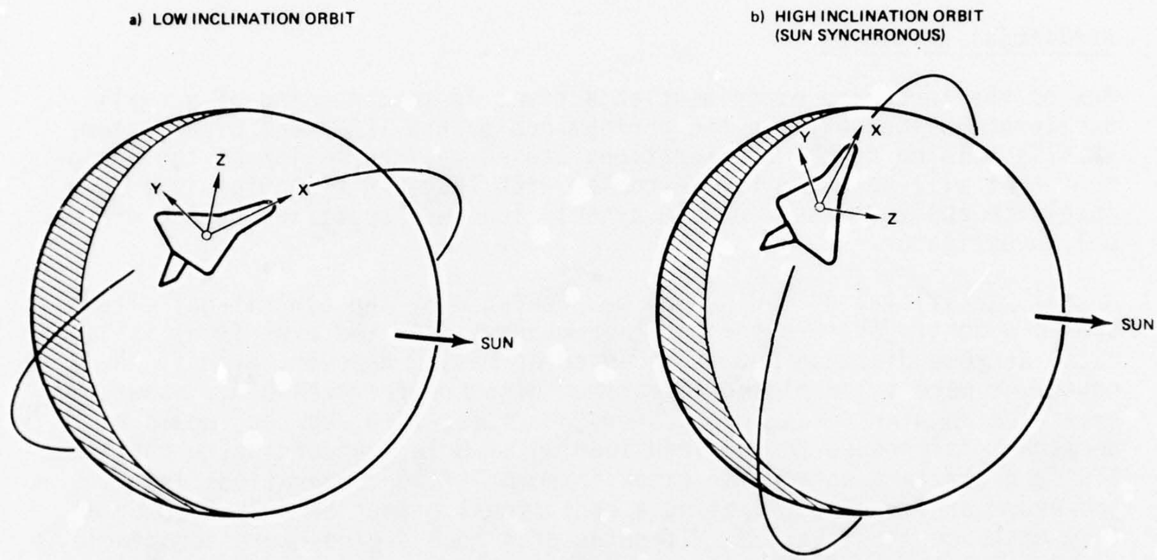

Figure 4.1-1. Preferred Orbiter Orientation for Sun Illumination Avoidance

4.2 Design Suggestions

Alternate methods to thermally isolate the experiment without requiring Orbiter orientations that may be unacceptable to other users include the following:

- Use of thermal coatings and insulation blankets on the experiment equipment and nearby cargo bay surfaces.
- Use of insulating mounts.
- Use of heat rejection methods other than a passive radiator, e.g., coupling to the cargo bay cooling system. (However, the added burden of 300 W. of waste heat may not be acceptable to that system, depending on other thermal loads.)

In order to determine (and record) the cargo bay thermal environment in which the experiment is conducted, it is suggested to place temperature sensors on nearby hot surfaces, e.g., adding a few thermocouples to those already used for internal instrumentation of the TESU and heat pipe.

A possible broadening of the experiment scope should be considered which would include the VM cryocooler rather than the simulated heat-sink. (See discussion of the VM cooler flight test.) Integration of the two experiments into an all-up system test could reduce the required total number of flight tests and probably would provide more comprehensive and realistic results. A more detailed review of the cost effectiveness of this approach and its other advantages versus technical difficulties involved in accommodating the system on the Orbiter is recommended.

4.3 Artificial g-loading

One of the secondary experiment objectives is to determine if a small acceleration load affects the performance of the TESU/heat pipe system. This is related to the accelerations caused by scan motion of the equipment that will be using the VM cooler with TESU and heat pipe in future satellite applications. A 0.1g dynamic load was specified by the principal investigator.

A small artificial g-loading can be generated by the centrifugal effect due to a continuous Orbiter rotation maneuver, if the experiment is located at some distance from the center of mass. However, even if the equipment were to be placed at maximum distance from the C.M., about 15m, excessive angular rates in pitch or yaw (i.e., 14.8 deg/sec) would be necessary to produce the desired loading of 0.1g. According to published STS data the largest angular rates in normal flight operations are of the order of 1 deg/sec, causing a centrifugal effect of only 0.0005g at 15 m distance from the C.M. If rates of 4 to 5 deg/sec were acceptable, centrifugal loads of the order of 0.01g could be produced, but only at a propellant consumption of 200 to 300 lb. Questions as to the permissibility of such manuevers, availability of enough maneuver propellant and usefulness of the small g level from the experimenter's stand point must be further investigated.

4.2 STS Integration Considerations

The experiment can be readily integrated with STS as previously discussed and requires little support equipment, preflight preparation, crew attention during flight and no ground station contact. Interface areas of concern are thermal control and heat rejection which may restrict Orbiter orientation during the mission (see above).

The average power consumption of 300 W. over a 3 day period (22 kw hrs) is only a moderate drain on Orbiter resources but must be evaluated in connection with power requirements of other users.

The experiment presents no significant safety hazard, but to preclude overheating due to thermostat failure, a caution and warning signal to the crew and manual override control must be provided.

Although the data volume requiring on-board storage has not been specified, the low sampling rate of the thermocouples providing the principal experiment output data indicates that a recorder capacity of 10^6 bits would be more than adequate.

The small total weight and size of the experiment and mounting fixtures are consistent with low STS transportation cost since the "getaway special" rate of $10,000 could apply in this case.

5.0 RECOMMENDATIONS AND REMARKS

This experiment will provide data which are obtainable only under the sustained zero-g environment of Orbital flight and under the influence of small g-loads that can be artificially imposed in this experiment. Utilization of the Shuttle Orbiter is recommended because of large transportation cost savings that would accrue (the special rate for small payload packages applies for this experiment) if repeated flight tests must be performed during system development.

Combination of this test with the VM cryogenic cooler test should be considered as a promising alternate possibility which may be more cost-effective than separate flight programs and will yield more comprehensive test results.

A concern exists with adequate control of the thermal environment, e.g., due to sun illumination of the cargo bay, and some restrictions on Orbiter orientation will have to be imposed. This requires further study.

Flight of this experiment as a "getaway special" should be considered.

MECHANICAL CRYOGENIC REFRIGERATOR EQUIPMENT

1.0 SOURCE

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and Maj. Peter Sivgals
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2.0 REQUIREMENTS BACKGROUND

Response to STS Utilization Presentation (Response Sheet 28)

Identifying Number: PE 63428, Project 212603

Performance requirements are classified

3.0 EXPERIMENT APPROACH

3.1 Background and Objectives

The Mechanical Cryogenic Refrigerator (MCR) is being developed as a possible candidate for use in cooling future missile surveillance sensors to very low cryogenic temperatures. It is an alternative to the Passive Cryogenic Radiator (PCR) also currently under development by the Air Force and being considered as a flight experiment that may utilize the STS, as discussed separately in this report.

Two different types of MCR equipment are being developed as candidate cryogenic coolers. One is the Vuillenmier(VM) cooler, the other is known as Rotary Reciprocating Refrigerators (R^3). A major concern with both of these coolers is long-life capability in an operational system. One model of the VM cooler has been flown previously and one is projected to be flown on SIRE. If the VM cooler does not show sufficient long-term performance in future ground tests then this flight demonstation will use the R^3 instead. At this time, the R^3 cooler is the more likely candidate for the flight demonstration, although design and development have not advanced very far.

Objectives of the flight demonstration include verification of thermodynamic characteristics of the system as well as mechanical characteristics of the gas bearings used to support the rotary equipment. Regarding the R^3 cooler, the operational system will use two modules operating in dynamic opposition to produce cancellation of reaction torques. The flight demonstration will include a check of any unbalanced torques generated in the space environment. Finally, the demonstration is to provide data from which long-life characteristics of the refrigerator can be deduced, although the STS Orbiter will have a mission duration of only one week (possibly up to three weeks in later flights).

3.2 Experiment Requirements

Power requirements of the operational MCR system range from 1500W (for the R^3 cooler) to 2000W (for the VM cooler). However, the flight demonstration model of either type of cooler probably will not require more than 1000W input power. Waste heat of the MCR process must be dissipated, possibly by using the Orbiter's heat rejection system. Otherwise, a separate heat radiator panel will be required. The heat rejection temperature is about 300°K.

Dimensions of the system have not been defined at this early experiment definition stage. System weight estimates range up to 350 kg.

Data requirements also remain undefined. A large number of data channels (up to 40 channels) is envisioned. No ground communication is required during most of the demonstration flight, except in the start-up phase. Probably several hours will elapse before the system reaches a steady state thermal condition. Subsequently, test data can be recorded on-board the Orbiter to be evaluated in post-flight analysis.

Test operations can be controlled automatically by a programmed sequence. Crew functions only involve turning the equipment on and off. However, some caution and warning signal for the crew is required to indicate if the system exceeds the operating maximum temperature range or otherwise requires emergency shut-off. No hazards in system operation have been identified.

No preference for specific orbit characteristics for this demonstration flight has been expressed by the principal investigator. Hardware will not be ready for launch at least until late 1982.

4.0 ASSESSMENT FOR STS FLIGHT

Information regarding equipment configuration, experiment procedures, support equipment requirements, STS interfaces and preflight preparation is still unavailable for detailed assessment of STS accommodation. However, it is apparent that the objectives of this demonstration require the zero-g environment of orbital flight.

Accommodation on the Shuttle Orbiter may prove cost-effective if sufficient spare power capability and heat dissipation capacity is available.

However, even with installation of an auxiliary power kit, the MCR experiment would require up to 25% of the added 840 kWh in a seven-day mission. Installation of auxiliary radiator panels aft of the primary radiators (to provide up to 2.2 kW of added heat rejection capacity) also must be considered as a probable extension of Orbiter capability for this experiment. The cost of both these auxiliary systems is chargeable to the experiment. The relatively large total experiment weight, estimated as about 350 kg, also is a matter of concern, but the cost impact will be less severe than that of the auxiliary power and heat rejection systems.

A major mismatch between the demonstration objectives and Orbiter capabilities seems to be the desired length of exposure to, and operation in the orbital environment. A seven-day maximum mission duration, as anticipated in the first years of Shuttle operations, may not be sufficient to accomplish some of the demonstration flight objectives, primarily those related to prediction of long-life performance.

5.0 RECOMMENDATION AND REMARKS

Further assessment of this flight demonstration as a candidate for STS utilization is recommended at the time when more specific experiment requirements are defined, e.g., when a decision as to VM or R^3 cooler flight demonstration has been made.

Accommodation problems of major concern are the great demand on the Orbiter's power supply and heat rejection system made by the proposed MCR flight demonstration. Addition of an auxiliary power kit and an auxiliary thermal radiator chargeable to this payload item will mean a significant cost increment above the basic STS transportation charge. The auxiliary radiator kit may be avoidable, however, if the total heat rejection rate of other payload items is reasonably low.

An area for further study also includes the question whether short-duration Orbiter flights are sufficient to provide results with regard to long life capability of the mechanical cry-coolers considered here.

PASSIVE CRYOGENIC RADIATOR EXPERIMENT

1.0 SOURCE

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and Maj. Peter Sivqals, Sponsoring USAF Office
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2.0 REQUIREMENTS BACKGROUND

Response to STS Utilization Presentation (Response Sheet 29)
PE 63428F, Project 2126
(Performance requirements are classified)

3.0 EXPERIMENT APPROACH

3.1 Background

The Passive Cryogenic Radiator (PCR) is being developed as a possible candidate for use in cooling future missile surveillance sensors to very low cryogenic temperatures in the 35 to 80°K range. This will be a 3-stage, heat pipe connected radiator of 100 to 150 sq. ft. area. It is an alternative to the Mechanical Cryogenic Refrigerators (MCR) also currently under development by the Air Force and being considered as a flight experiment utilizing the STS, as discussed separately in this report (see Item No. 28). The PCR which requires few if any moving parts for its operation is simpler in design then the mechanically driven devices, but the large panel size may present problems of storage and deployment as well as appropriate orientation so as to always face cold space. This problem which affects the use of the passive radiators on operational surveillance satellites is a matter of concern, as well, in conducting the proposed flight test on the STS Orbiter. The radiator is now only at the basic design stage and will not be ready for flight until 1982.

Related technologies include heat pipes, low thermal conductivity structures, cryogenic insulation and thermal control coatings.

3.2 Flight Test Objective

A weightless condition, achievable only in orbital flight, is essential to verify the simultaneous operation of all of the differently oriented heat pipes used in the radiator system. The principal flight test objective is to observe the radiator's effective cooling performance in the orbital environment and to compare it with predictions based on ground-based laboratory tests.

A secondary objective is to determine the performance degradation that may be caused by contaminant particles deposited on the radiator surface, e.g., through condensation of thruster exhaust products. Such condensation is unavoiadable because of the extremely low radiator surface temperature.

3.3 Experiment Requirements

The net power to be dissipated by the radiator is about 5 W. Adding the effect of parasitic heat sources, the total dissipation is typically 12 W. The very low radiator temperature necessitates the large 100 to 150 sq. ft. panel area. The coolant to be used in the heat pipes that couple the radiator to the heat source will be liquid oxygen.

Instrumentation consists primarily of temperature sensors distributed throughout the system. The total heat flux through the system must be carefully monitored to take into account any heat leaks from surrounding structures. The experiment duration is typically 3 to 4 days to ensure that the system will reach a stable thermal equilibrium.

The experiment requires deployment of the radiator panel from the stowed condition at the start of orbital operations. The test specimen radiator must be as large as that of the future operational system since a scaled-down radiator unit would not yield the desired verification of heat pipe performance under zero-g. Folding of the panel for easier storage in the cargo bay is undesirable since it, too, can adversely affect test realism.

An orbital altitude of at least 250 n.m. is required to reduce radiator surface contamination by atmospheric particles to an accepted level.

For a valid performance test, the radiator must always be oriented to face cold space. This can be accomplished either by using a steering mechanism or by flight in a sun-synchronous orbit. The orbiter must be oriented appropriately to shade the radiator against sun exposure.

Radiator surface contamination by particles emitted from the orbiter's environmental control and life support system (ECLSS) and by rocket exhausts must be minimized. This not only requires placement as far as possible from contamination sources, i.e., probably in the midsection of the cargo bay, but imposes restriction on the release of ECLSS waste during the active period of the test. Such restriction have previously been contemplated to protect contamination sensitive Orbiter payloads, e.g.,low temperature telescopes. Also, it probably will be necessary to restrict Orbiter RCS system operations during the critical test period to vernier thrusters as the plume impingement from the 900 lb primary thrusters appears prohibitive.

3.4 Experiment Support Requirements

Because of the early stage of the radiator system design, no experiment support requirements have been defined as yet regarding command, data handling and telemetry requirements, pre-flight preparations, crew tasks, ground control activities, etc. Crew tasks probably will be limited to system development and retraction and turn-on/turn-off of the experiment and its sensors, associated electronics and recording equipment. On board storage of recorded data is expected to be adequate, and will minimize telemetry requirements.

4.0 ASSESSMENT FOR STS FLIGHT

Verification of system performance under zero-g conditions can only be accomplished in orbital flight. However, accommodation of this experiment on the STS Orbiter appears to be difficult and costly. Difficulties are associated, primarily, with the extreme susceptibility of the cold radiator to contamination by water vapor and thruster plume condensation which will affect cooling performance. Of critical importance will be the amount of residual plume impingement upon the radiator assuming that it is located optimally , i.e., near the cargo bay center, and RCS operation is restricted to the vernier thrusters.

Secondly, storage of a 10x15 ft, or similarly dimensioned radiator panel that does not permit fold-up means that this experiment will occupy an inordinately large part of the Orbiter's cargo bay, with a proportionately large transportation cost, unless the space beneath or on both sides of the stored panel can be made available to other uses.

In addition, the radiator pointing and sun avoidance requirements tend to restrict the Orbiter orientation severely and are compatible with only a limited range of orbital characteristics. Near-polar sun synchronous orbits may be required. This reduces the number of flight opportunities.

Detailed analyses of these problem areas are required to determine whether accommodation in the Orbiter is practical and cost effective.

4.1 Experiment Considerations

Some radiator configuration, storage, deployment and pointing options that may resolve or alleviate Orbiter accommodation problems discussed in the preceding paragraph are illustrated in Figures 4.1-1, 4.1-2, 4.1-3.

Ways to conserve cargo bay volume in the vicinity of the stored, non-foldable radiator panel for use by other payloads and thus to reduce Shuttle transportation cost, are illustrated in Figure 4.1-1. The sketches show cargo bay cross sections with the radiator panel stored upright between small payloads or on the side of a larger payload unit. In the side-mounted arrangement, the maximum panel width can be accommodated more economically if a curved or segmented configuration is adopted. However, this may cause some Radiator efficiency loss and requires a tradeoff study.

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In the horizontal stowage arrangement also shown in Figure 4.1-1, the non-planar panel configuration would have the same advantage of more economical cargo space utilization. A change from the proposed, nearly square configuration to one with smaller width and greater length may facilitate radiator storage in some cases and help reduce the waste of useful cargo space associated with storing the bulky flat panel.

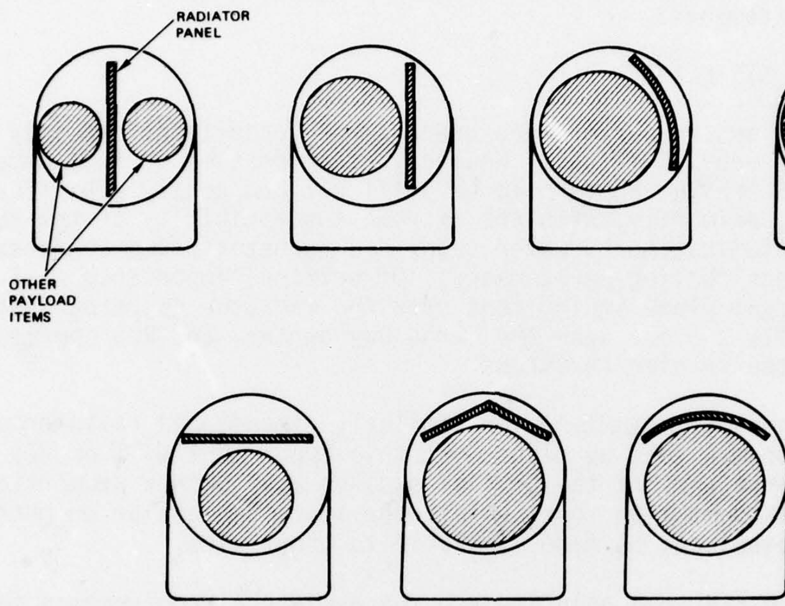

Figure 4.1-1. Radiator Panel Stowage Options (Schematic)

Figure 4.1-2 illustrates alternate panel deployment schemes, i.e., rotation vs. translation, consistent with the desired central location of the deployed radiator that minimizes RCS thruster plume impingement. These alternatives provide flexibility in placement of the stored panel along the cargo bay so as to reduce possible conflicts with placement priorities of other STS users. A main concern is the length dimension of the panel which accounts for nearly one quarter of the total cargo bay length (60ft).

Figure 4.1-3 shows the deployed panel above the cargo bay, tilted with respect to the Orbiter X-Z plane to avoid viewing parts of the wing structure. Tilt angle variation using the deploy/retract actuator as a single-axis or two-axis drive mechanism may be desired to permit some freedom of Orbiter orientation while maintaining the view of dark space and avoiding sun exposure. The sketch shows radiation shields deployed on three sides of the radiator to avoid viewing parts of the Orbiter structure under changing panel orientation.

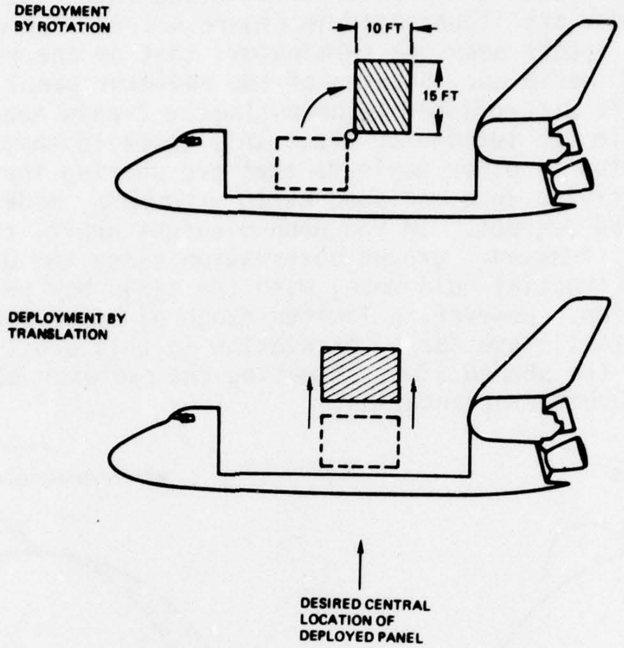

Figure 4.1-2. Radiator Stowage/Deployment Flexibility (Schematic)

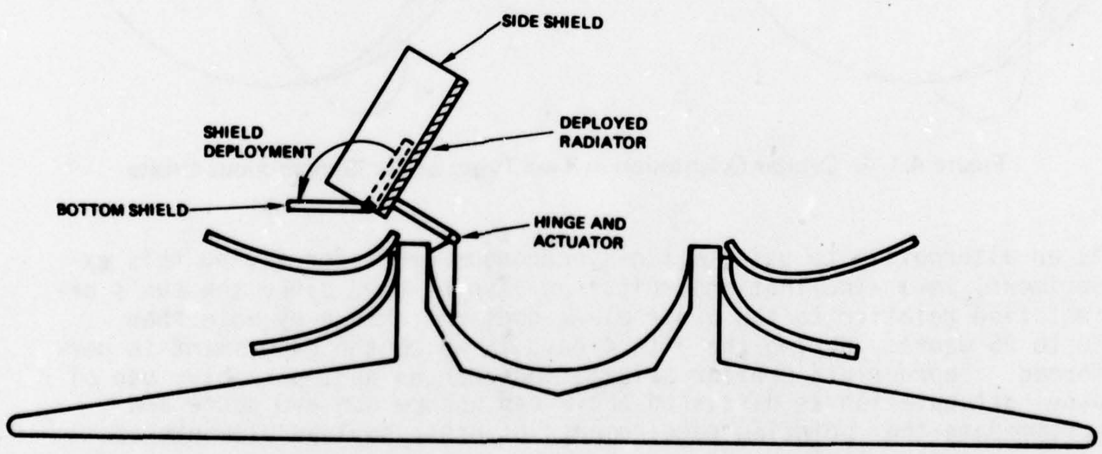

Figure 4.1-3. Conceptual Positioning of Deployed Radiator (Schematic Only)

Sun avoidance is easiest to maintain if the experiment is flown in a near-polar, sun synchronous orbit as suggested by the principal investigator. Two such orbits and Orbiter orientation requirements consistent with sun avoidance are illustrated in Figure 4.1-4. The orbit shown on the left has ground tracks near the terminator, that on the right near the noon - midnight meridian. Shading of the radiator panel by the Orbiter body and wings is accomplished by orienting the Z-axis nearly parallel to the sun line. In the terminator orbit this tends to hamper ground observation by other payloads that are sharing the mission unless the Orbiter is flying in a modified earth pointing mode with a bank angle of 45 to 60 degrees. In the noon-midnight orbit, the shading requirement also hampers ground observation since the Orbiter would be flying in an inertial hold mode, with the cargo bay pointing in anti-sun direction. However, a limited range of pitch angles would be permissable to facilitate earth observation in this orbit while maintaining the radiator on the shaded side. Steering the radiator can alleviate these restrictions on Orbiter orientation.

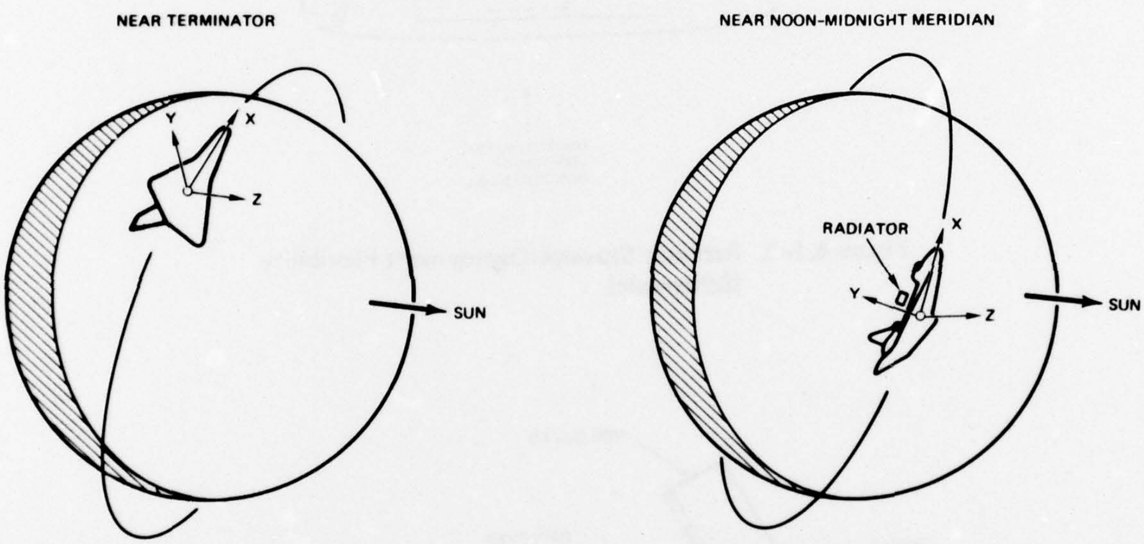

Figure 4.1-4. Orbiter Orientation in Two Types of Sun-Synchronous Orbits

As an alternative to using a sun-synchronous orbit for flying this experiment, lower-inclination orbits can also be used since the sun's orientation relative to the orbit plane does not change by more than 20 to 25 degrees during the 3 to 4 days in which the experiment is performed. Appropriate Orbiter orientation changes and, possibly, use of panel articulation as discussed above can assure sun avoidance and accommodate the pointing requirements of other payload elements at least to some extent.

4.2 STS Integration Considerations

Safety considerations demand that the radiator panel be jettisonable if failure of the retraction mechanism would interfere with closing the cargo bay doors at the end of the mission.

This can be accomplished by means of a pyrotechnically activated separation device installed near the panel deployment hinge. However, great care is required in the layout and operation of the ejection mechanism to preclude any possiblity of Orbiter damage during jettison, e.g., due to panel tumbling, etc.

Handling of the liquid oxygen used in the radiator heat pipes prior to and during the mission is a matter of concern, although only a small quantity will be required in this experiment. The liquid oxygen will be kept in a liquid nitrogen cooled storage tank until the experiment is initiated. Upon deployment of the radiator the LOX will be slowly transferred into the heat pipes. The chill-down process probably requires several hours. Upon completion of the experiment, the LOX contained in the heat pipes and any residual amount remaining in the storage tank will be vented overboard. Thus no accidental LOX spill can occur if jettison of the radiator panel should become necessary.

Use of liquid nitrogen having a boiling point at only 12.6° C higher temperature than LOX may be worth investigating as an alternate heat pipe coolant since it would be less hazardous in many respects, in preflight handling and during the mission. However, this alternative may already have been considered and dismissed by the principal investigator.

Other STS integration factors of the experiment appear to be only of a routine kind. The experiment is nearly self-contained and can be controlled by a programmed sequence once a steady state condition has been attained. The chill-down procedure probably will require active participation, or at least monitoring by a crew member. A caution and warning indicator to alert the crew to a rupture in the cryogenic supply tank or the heat pipes or other hazardous conditions will be necessary.

As previously noted, the principal concerns regarding STS integration are compatibility with Orbiter operations and with observation objectives and priorities of other payloads, as well as prohibitive cost penalties that may occur as a result.

5.0 RECOMMENDATIONS AND REMARKS

Beyond this preliminary assessment of experiment accommodation feasibility, more detailed analysis of key questions is required such as:

- Severity of RCS thruster plume and ECLSS contamination effects on Radiator performance measurements.
- Compatibility of Radiator pointing with Orbiter orientations required by other payloads.
- Cost implications of experiment accommodation difficulty.

Accommodation on a payload delivery/retrieval mission should be investigated as an alternative, where few if any other experiments would be carried that would pose a possible conflict in orientation requirement. A matter of concern in this case would be the cost of service charges for extra days in orbit until the radiator experiment is completed.

ADHESIVE/STRUCTURAL BONDING IN A SPACE ENVIRONMENT

1.0 EXPERIMENT IDENTIFICATION

Mr. David A. Roselius, Principal Investigator
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2.0 REQUIREMENT BACKGROUND

Response to STS Utilization Presentation (Response Sheet #30)

3.0 EXPERIMENT APPROACH

Adhesively bonded specimens of composite materials contemplated for use in space structures will be exposed to the space environment using the Long Duration Exposure Facility (LDEF). After retrieval, experiments will be conducted on earth to determine the extent and type of degradation.

4.0 ASSESSMENT FOR STS FLIGHT

There should be no problem in integrating this experiment with other experiments utilizing the LDEF facility. The principal area of concern is initial surface contamination of the specimens after deployment which may give rise to anomalous degradation results during subsequent exposure to the space environment.

4.1 Experiment Considerations

The experiment will be conducted in the LDEF and is completely passive. Measurements on the specimens will be conducted on the ground, some before and after they have been exposed to the space environment, and some only after exposure. (The only cost consideration is the retrieval of the LDEF after exposure.)

The specimens must be protected from surface contamination during launch, initial deployment phase and before retrieval of the LDEF. An experiment Vacuum Exposure Control Cannister (VECC), presently being developed, will insure surface fidelity during these times. There is no constraint on the STS with respect to any aspect of this experiment. The LDEF and VECC are described in Appendix A.

4.2 STP Integration Considerations

Integration of this experiment with other LDEF experiments should be accomplished easily, in that the experiment is confined to a tray (or trays) in which the specimens can be mounted independently of the LDEF. Therefore, the experiment package can be considered independent of STP except for actual integration into the LDEF, retrieval, and subsequent removel for shipment to the Principal Investigator.

5.0 RECOMMENDATIONS AND REMARKS

It is recommended that this experiment be placed in the LDEF since long-term exposure to the space environment is necessary for successful interpretation of the experimental results.

It is suggested that an alternate experimental approach be considered: by use of a system similar to the Experiment Power and Data System (EPDS), the VECC can be progressively opened during mission lifetime so as to expose successive areas of the tray to the total space environment. Thus, by replicating specimens and progressively opening the tray (for example: 1/3 at 2 months, 1/3 at 4 months, and 1/3 at 6 months), specimen degradation data as a function of exposure time could be obtained, potentially allowing extrapolation of the data to times longer than that of the LDEF mission duration.

STS - LDEF MULTIPHASE MATERIALS PERFORMANCE/CONTAMINATION EXPERIMENT

1.0 EXPERIMENT IDENTIFICATION

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2.0 REQUIREMENT BACKGROUND

Response to STS Utilization Presentation (Response Sheet No. 31)

Supports SAMSO/DoD: STS Payload Interface Contamination Considerations

3.0 EXPERIMENT APPROACH

The purpose of the investigation is twofold:

(a) to determine the degree and nature of the contamination to which STS Shuttle Bay Payloads are exposed during various mission phases, i.e., during launch, deployment, on orbit and during recovery and reentry.

(b) to determine the effects of the LDEF space environment exposure on thermal control coatings and other satellite and space system materials.

Seven duplicate samples of various materials will be exposed. The types of materials are:

- Thermal Control Coatings
- Polished Metals
- Front Surface Mirrors
- Second Surface Mirrors
- Optical Flats (UV-IR)
- Polymeric Films
- Solar Cell Covers
- Insulation Blankets
- Adhesives
- Transparent Thin Films

One of the duplicate sets of materials is exposed through all of the operational phases of the mission. Each of the other sets is selectively exposed during one of the phases, i.e., prelaunch/installation, launch, removal/insertion, orbital, retrieval, and reentry/recovery.

The samples are returned to earth for diagnosis and the material property measurements shown in Table 3.0-1 are performed. The nature and extent of any surface film and/or particulate contamination will be determined and correlated with the various phases of the overall flight.

TABLE 3.0-1

MATERIALS PROPERTY MEASUREMENT

THERMO-OPTICAL	SOLAR ABSORPTANCE EMITTANCE
OPTICAL	TRANSMISSION SPECTRAL PROPERTIES, UV-IR
ANALYTICAL	AUGER ELECTRON SPECTROSCOPY ELECTRON SPECTROSCOPY FOR CHEMICAL ANALYSIS SECONDARY ION MASS SPECTROSCOPY FRUSTRATED MULTIPLE INTERNAL REFLECTION SPECTROSCOPY DIFFERENTIAL SCANNING CALORIMETRY ELLIPSOMETRY SCANNING ELECTRON MICROSCOPE
PHYSICAL	WEIGHT LOSS % ELONGATION TENSILE STRENGTH MODULI YIELD STRENGTH
ELECTRICAL	DIELECTRIC LOSS DIELECTRIC CONSTANT VOLTAGE BREAKDOWN

4.0 ASSESSMENT FOR STS FLIGHT

This experiment is well along in its planning for an LDEF flight and it is clearly a candidate for that kind of STS facility. The experiment is self-contained and requires no services from STS.

4.1 Experiment Considerations

4.1.1 Design Considerations

Mechanical

The experiment consists of two concentric disks. The upper disk can be stepwise rotated about the common center. The sample set that is exposed to the environment is mounted on this disk. The selectively exposed samples are mounted on the lower disk and are shielded from the environment by the upper disk. These samples are selectively exposed to the environment through slots in the upper disk as the upper disk is stepwise rotated.

The disk diameter is about 10 inches in radius and 3 inches deep deep and weighs between 20 and 30 lbs. This easily fits into a standard LDEF tray which is 37.5 inches long and 49.5 inches wide and comes in varying depths of 3", 6" and 12". Each standard tray can accommodate up to 175 lbs.

Electrical

A very small amount of power is required to operate the stepping motor. Power (less than one watt) is used during each step about ten times throughout the entire mission. The small energy requirement can readily be accommodated by batteries that will fit within the weight and volume capability of one standard tray.

No external command, telemetry or power is required.

Thermal

Passive thermal control will be included as part of the experiment.

Areas not covered by samples will be coated with adhesively bonded low outgassing metallized polymeric films, FEP/Ag or FEP/Al or silica fabric thermal control coatings. Individual samples will be allowed to reach their own equilibrium temperature.

Samples under the sector wheel will be kept cool because of the low temperature of the sector wheel cover.

Operations

This experiment can be flown in any LDEF orbit and imposes no operational restrictions on LDEF. The stepping of the motor is automatic and pre-programmed.

The ground support equipment is nominal and all unique equipment is provided by the experimenter. This includes contamination protection before selective exposure and equipment needed to test out stepping motor and logic. The handling and testing of this experiment appear relatively straightforward.

After recovery, the instrument and samples are returned to the experimenter for evaluation.

Reflight of the experiment is anticipated. This can be done by simply cleaning the instrument and installing new samples.

4.2 STP Integration Considerations

A conceptual layout of the STS-LDEF Multiphase Materials Performance/Contamination experiment is shown in Figure 4.2-1. As shown in the figure, the experiment fits easily into one standard tray. Also shown in the tray is the electronics for the stepping motor and batteries. This experiment uses so little power that it will probably be possible to share power with another LDEF experiment. In that case, the power could be supplied by an Electrical Power and Data System (EPDS) obtained from Langley by STP. These units occupy one-third of a tray and cost approximately $50,000 each.

Scheduled LDEF flights permitting 6-9 month exposures for this experiment are also shown. This experiment could be ready for a 1980 flight.

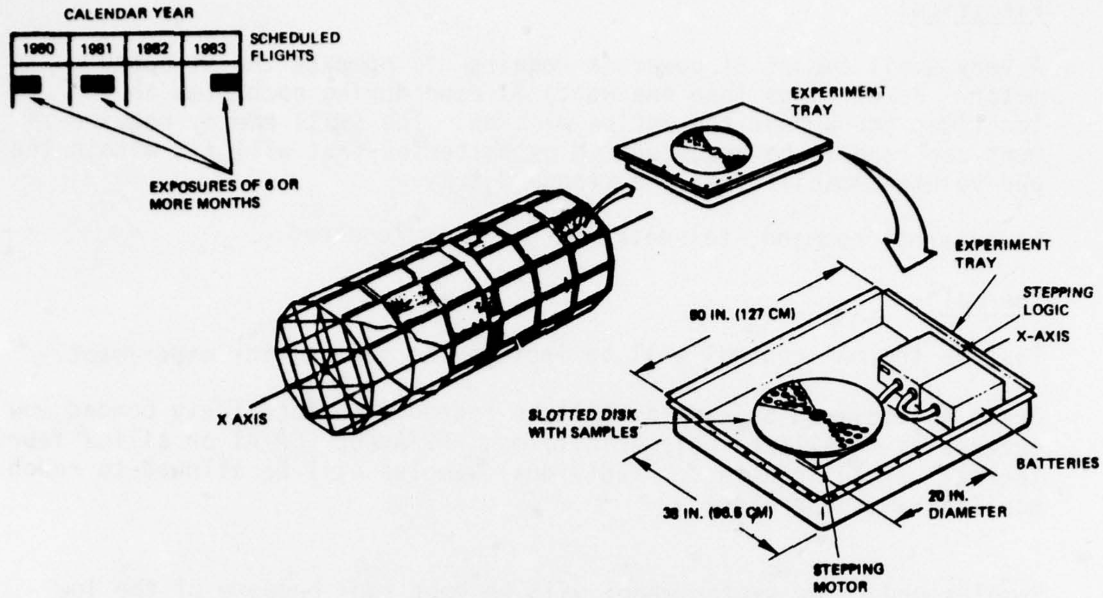

Figure 4.2-1. Multiphase Materials Performance/Contamination Experiment in Long Duration Exposure Facility

5.0 RECOMMENDATIONS AND COMMENTS

The STS-LDEF Multiphase Materials Performance/Contamination experiment is an excellent candidate for an LDEF flight in early 1980. No problems in integrating this experiment into LDEF are anticipated. The experiment can be accommodated easily in one standard LDEF tray and requires no STS services. The small amount of power need can be supplied by a dedicated battery or by an EPDS shared with another experiment.

SPACECRAFT CHARGING

1.0 EXPERIMENT IDENTIFICATION

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2.0 REQUIREMENT BACKGROUND

Responce to STS Utilization Presentation (Response Sheet No. 32)

Project Number 7661 at AFGL/LKB.

3.0 EXPERIMENT APPROACH

The investigator wishes to examine the charging properties of the Shuttle orbiter by ejecting energetic electrons and ions from the Orbiter and measuring how the Orbiter interacts with the ambient ionospheric plasma environment. Electron and ion accelerators capable of delivering currents of about one ampere at a few kilovolts will be used to eject charged particles from the Shuttle orbiter. A diagnostic package consisting of plasma measuring devices will monitor the levels to which the orbiter charges relative to the ambient plasma and will measure any anomalous behavior of the ionospheric plasma.

The accelerator package includes both the ion and electron elements. It is about 1 foot long and 6 inches in diameter. The diagnostic package consists of an electrostatic analyzer and a retarding potential analyzer in a 10 inch cubic box. There should be some separation between the diagnostic instruments and the accelerators; ten feet should be adequate. The weight of the accelerator package would be about 10 pounds, and the weight of the diagnostics would be about 5 pounds. Figure 3-1 shows a sketch of the two experiment packages.

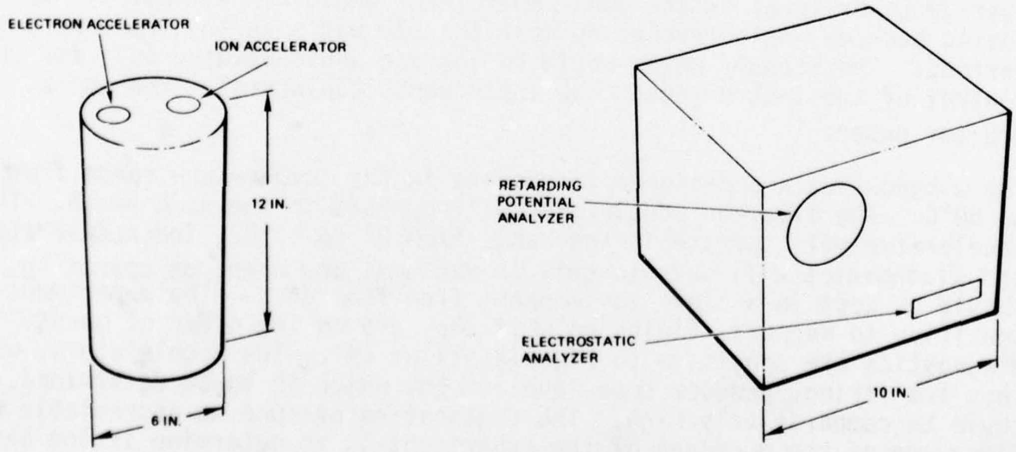

Figure 3-1. Sketch of the Instrumentation for the Spacecraft Charging Experiment

The operations conducted during the experiment consist of periodic firings of the accelerator for time periods up to 10 minutes. During these firings, the diagnostic instruments would make observations of the Shuttle potential and the ionosphere. The experiment may be repeated at as many different altitudes, attitudes and geographic locations as is possible. The operations would take about one hour per experimental period and there would be two periods per day for a total time of 2 hours per day. The operation of the experiment could be nearly fully automated with the crew involvement minimized to monitoring safety and status functions.

The accelerators need to be fired nearly along the magnetic field line so that the charged particles will travel well clear of the Orbiter. The diagnostic package's orientation with respect to the magnetic field line and the orbital velocity vector need to be known to at least 2° or better. 2° accuracy for the accelerators would be adequate.

The data rate from the accelerator during its period of firing would be about 1 Kbps. This rate would be comprised of status monitoring of the accelerator voltages, currents, temperatures and power supply. The data rate from the diagnostic package would be on the order of 2 Kbps including the electron and ion spectra and temporal variations in the plasma density. There is a need to know the orientation of the instruments to the geomagnetic field during operations. There would be on the order of 50 bilevel commands for the accelerator and diagnostics. These commands would be to control the operation of the accelerator -- such as voltage level, current range, heaters, etc.

The accelerators will need a variable power input depending upon the amount of energy in the charged particle beams. Under the assumption that the Shuttle orbiter is not easily charged by the ejection of the particle beam, power levels on the order of several hundred watts would be needed for the electron beam. The investigator estimates that for high power operations, the average power needed by the accelerators would be 300 watts and that the peak power could reach 2500 watts. The maximum duration of either the average power level or the peak power level would be 10 minutes. The diagnostic package would require no more than 10 watts during the experimental periods. The standby power would be low and would be used only for thermal control of the instruments. The instruments can operate from the 28 volt Orbiter power.

The diagnostics are designed to operate in the temperature range from -20° to 50°C. The electron accelerator will operate in the same range. The ion accelerator will operate in the range from 0° to 60°C. The accelerators and diagnostics will operate only in vacuums, and when not operating, should be kept in a clean environment free from dust. The experiment is sensitive to magnetic fields only if they are on the order of gauss. The diagnostics are sensitive to high levels of EMI. The accelerators, when they are firing, produce some level of EMI which is to be determined, but could be comparatively high. The diagnostics produce no appreciable EMI. Since one of the purposes of the experiment is to determine if the Shuttle orbiter can be charged up by the ejection of charged particles, it should be expected that high voltages between the Shuttle and the ambient plasma could be produced. It is possible that high voltage arcing could occur which would produce very high levels of EMI.

The investigator would like to refly this experiment as often as possible in the early Shuttle flights in as many different orbital inclinations and altitude regimes as possible. The instrumentation has been flown on sounding rockets.

4.0 ASSESSMENT FOR STS FLIGHT

This experiment is well qualified to fly on the Shuttle. Since the object of the experiment is to measure the properties of the Shuttle orbiter itself, it is necessary to perform the experiment on an STS flight. The accelerators and the diagnostic package can be mounted in the cargo bay with the necessary controls mounted in the aft flight deck.

Possible problem areas are: (1) The accelerator will generate some amount of heat that will have to be dissipated; (2) there is some possibility of high voltage arcing at some point on the Orbiter or in the payload if the Orbiter charges up to high voltage with respect to the ambient ionosphere, (3) the production of EMI by the accelerator during its firing may be a problem, (4) there is very minor danger of radiation exposure for the crew during accelerator firings.

4.1 Experiment Considerations

4.1.1 Design Suggestions

It is suggested that the investigator study the long-term thermal properties of the electron and ion accelerators to determine if they will need active cooling. This cooling could be effected by connecting the accelerator package to a heat exchanger or placing it on a Spacelab cold plate.

The investigator should study the character of the EMI radiation produced by the accelerator when it is fired and take design or operational steps to reduce unacceptable levels if they appear. This would mean testing the unit in a large vacuum system.

There should be no problem in mounting either one of the packages on the Orbiter. They could fit on Spacelab pallets, standard test racks or any other mounting points capable of taking their very modest loads. The diagnostic package should be mounted in a position above the midplane of the cargo bay so that it has a clear view of the ambient plasma. It would be desirable if there were some exposed electrical conducting surfaces near the diagnostic package.

The command and data handling lines could be either hardwired, or the instruments could be designed to be compatible with the Spacelab CDMS and connected to a RAU. The data rates and operation times are fully compatible with either the Orbiter or Spacelab capabilities. There would be need for a small control panel on the aft flight deck if Spacelab CDMS is not used.

The electrical power usage is consistent with Orbiter capabilities. There should be no problem meeting even the 2.5 kW demand by planning so that no large power using experiment is operated at the same time. The power could be hardwired from the Orbiter or delivered through the Spacelab power system.

With the instrumentation hardmounted to either the Orbiter or to Spacelab pallets, there would be no difficulty in meeting the pointing requirements of the experiment. It is suggested that the experimenter supply his own magnetometer. The added weight and power would be less than 1 kg and 1 W. The relative separation between the accelerators and the diagnostic package should be about 10 feet or more, although the experiment is operational even if the packages are contiguous. There is no requirement that packages be aligned accurately with respect to each other. The 2° accuracy holds also for relative alignment between the two packages. If the diagnostic package contains the magnetometer, then the alignment between the accelerator and the diagnostic package should be 1° or better.

4.1.2 Operation Restrictions

The major operating restriction on firing the full power electron beam is to have adequate power available when the Orbiter is pointed so that the beam will travel parallel to the earth's magnetic field lines. For safety, it should be required that there be a gradual buildup in the intensity of the beams so that the charging behavior of the Shuttle could be studied. Present studies indicate that the Orbiter should not begin to charge up for ejected currents less than about one ampere. Nevertheless, the increase in the charged particle beam power should proceed as follows:

> The beams should be fired at low intensity, with the resulting spacecraft potential relative to the ambient plasma determined, the strength of the beam then increased in gradual steps until the potential did begin to differ significantly (greater than about 100 volts) from the plasma potential; after that it should be increased in even smaller increments so that at no time is the Orbiter charged to kilovolt potentials. The reason for this care is that high voltage breakdown between conducting and non-conducting surfaces of the Orbiter could occur if the potential of the conducting surfaces were raised too high.

There should be no problems associated with the experiment operations as stated above. The Orbiter can be oriented with respect to the geomagnetic field so that the beams can be fired along the field lines -- either up the field line or down it. The Orbiter can easily be held in this orientation for the one-hour experimental period within the required accuracy.

The crew involvement with the experiment could be minimized if it were flown on a Spacelab flight. The stepping up of the beam current and the analysis of the Orbiter potential could all be done automatically from a program stored in the CDMS computer. The function of the crew would be to align the Orbiter, check the status of the instrumentation, load the automatic program, initiate the sequence, and terminate the experiment. If the experiment does not fly with the Spacelab, these functions could perhaps be done by the Orbiter computer, however, it would be more likely that the crew would have to control the step by step procedures from a control panel on the aft flight deck. Either one of these operational methods is feasible.

The experiment cannot be operated on the ground so that tests during integration would be restricted to checking the electronic functioning of the diagnostics and of the accelerator firing circuitry. This is another reason for gradually raising the power levels when the experiment is performed during the flight.

4.1.3 Experiment Support Equipment

The experiment needs power, data handling, control, crew interaction and Orbiter orientation support during the flight. The needs of the experiment are entirely consistent with the abilities of either the Orbiter or the Spacelab system. No other support equipment would be needed during the flight.

4.1.4 Experiment Cost Considerations

This experiment has already been flown in rocket configurations, and the transference to STS flights should involve little reworking of the existing instruments.

4.2 STP Integration Considerations

The only problem that is foreseen in integrating this experiment is the possible EMI during the firing of the accelerator. Some procedures must be developed to assure that during the firing of the accelerator in flight, there are no unacceptable levels of EMI. For this experiment, there is no need to switch the beam on or off rapidly and therefore, it should be possible to keep EMI within acceptable limits.

Figure 4-1 shows a conceptual layout of the experiment packages on Spacelab pallets. It should be noted that there is no a priori reason why pallets are required. The data, power and cooling could be hardwired directly from the Orbiter.

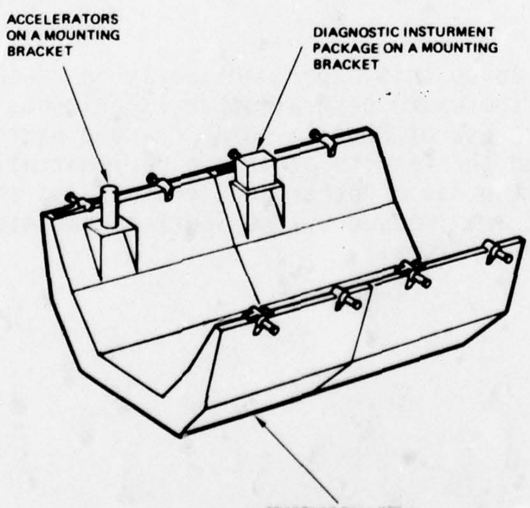

Figure 4-1. Conceptual Layout of the Spacecraft Charging Experiment Instrumentation on Two Spacelab Pallets

The following items must be purchased as optional STS services: (1) Mounting fixtures in the cargo bay are required. Power, data and cooling connections are required; (2) there may be the need for unique payload/Orbiter EMI integration tests, (3) more than one day of flight operations would be desired but not definitely required and (4) there is need for some data processing onboard during the flight to determine the actual Shuttle orbiter potentials from the data output of the diagnostic instruments.

5.0 RECOMMENDATIONS AND REMARKS

1. This experiment is compatible with the STS/Spacelab and has to be flown on the STS Orbiter. Its science objectives are important to other experiments, therefore, it should fly on early STS sortie missions.

2. Care should be taken with the design of the experiment to minimize EMI from the accelerator firings.

3. The operation of the accelerator in orbit should be planned for a show progression from low current, low voltage beams, to high current high voltage beams. During this progression, the potential of the Shuttle orbiter relative to the ambient ionospheric plasma should be continually monitored and determined to be low enough so that no danger of uncontrolled high voltage breakdown exists.

4. The investigator should determine if the accelerator can operate without using the Orbiter fluid cooling lines for thermal control. The costs of integration would be significantly lowered if this were possible.

5. The accelerator should not be fired at 90° pitch angle because of the possible radiation hazard to other payload elements. There should be no EVA when the accelerator is firing.

6. It is important to do this experiment early in the Shuttle program because there are several other experiments that will want to make use of Shuttle-borne charged particle accelerators, and the results of this experiment will directly impact the use of other accelerators and the current limits that are imposed by the Shuttle characteristic in the ionospheric plasma.

PASSIVE ENERGETIC PARTICLE DETECTORS

1.0 EXPERIMENT IDENTIFICATION

Robert Filz, Principal Investigator
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2.0 REQUIREMENT BACKGROUND

Response to STS Utilization Presentation (Response Sheet No. 33)

3.0 EXPERIMENT APPROACH

The object of this experiment is to determine the time history of the trapped proton flux in the earth's radiation belt. This will be accomplished by flying small nuclear emulsion packages on numerous Shuttle flights. Analysis of the emulsions is made after recovery. The data obtained in this manner will be compared with data taken on AF recoverable satellites flown through the 1960's.

In its simplest form, the experiment is completely passive. Several samples varying in size and shape from about 2" in diameter, 1/4" thick, to a rectangular sample 4"x4"x1" thick are mounted in the orbiter bay so as to expose the emulsion to the radiation environment with a nearly 2π clear field of view. The various samples are designed to be sensitive over different energy ranges. The total sample weight is less than one pound.

The integrated proton flux to which each emulsion will be exposed throughout the mission will depend on the characteristics of the orbit and also on the history of the orientation of each sample relative to the geomagnetic field. This orientation will be known as soon as each mission trajectory and orbit are known and flights will be chosen to maximize the amount of information obtained.

A conceptual layout of the Passive Energetic Particle experiment is shown in Figure 4-1. In this concept, various size nuclear emulsions are bonded to the cargo bay door.

Because of the simplicity of this experiment, it can be ready for flight within a week after the actual go-ahead is given.

4.0 ASSESSMENT FOR STP FLIGHT

Clearly, this experiment can readily be accommodated on any STP Shuttle flight. The emulsions require relatively short exposure times (approximately one week) and therefore could not use the LDEF. Since it also requires recovery, its flight on attached orbiter payloads is recommended.

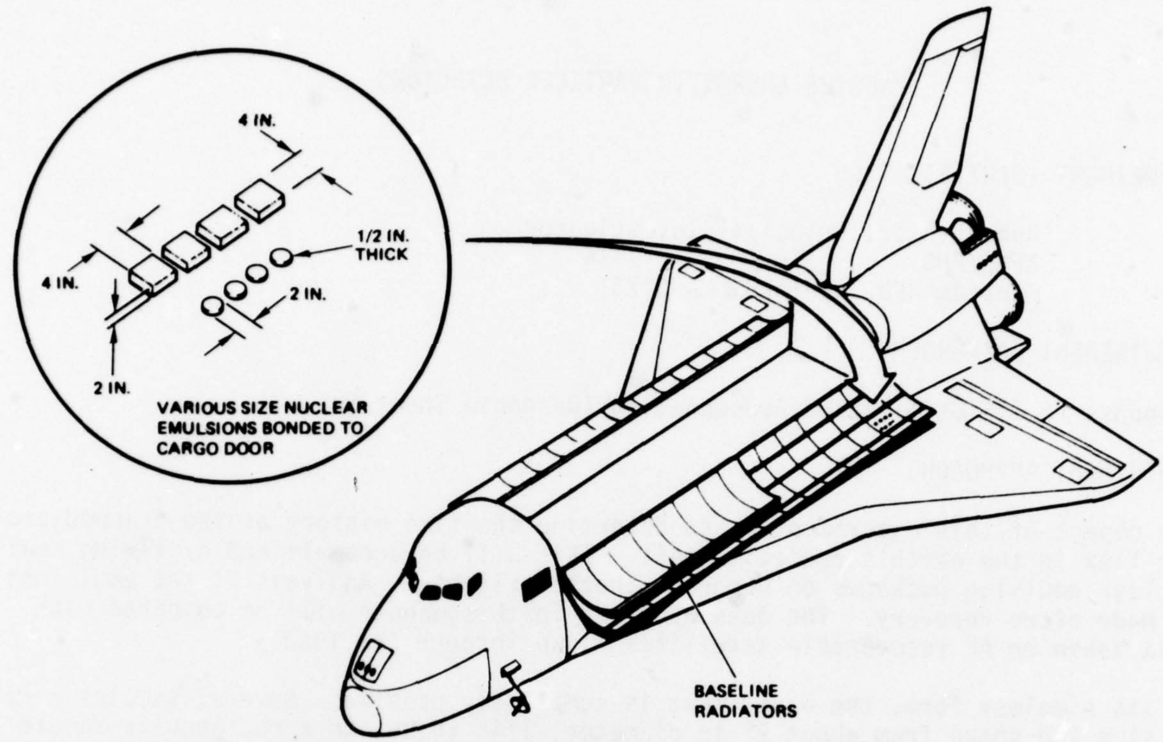

Figure 4-1. Conceptual Layout of the Passive Energetic Particle Experiment on the Shuttle

4.1 Experiment Considerations

4.1.1 Design Suggestions

The mounting of the experiment can be on any surface that permits exposure to the environment with a large ($\sim 2\pi$) clear field of view. The concept of attaching the emulsion to the cargo doors (as shown in Figure 4-1) may not always be feasible since orbiter heat rejection is provided by means of radiators on the cargo bay doors. In the baseline orbiter configuration, these radiators mount forward on the cargo doors as shown in Figure 4-1. However, on missions where additional heat rejection is required, optional payload radiators can be provided which extend the radiators across the entire length of the doors.

In any case, there should be no problem finding adequate locations for the samples on most flights.

The temperature of the emulsions must not exceed 100°F. This will require that the experiment provide passive thermal control with thermal blankets biased toward the cold temperatures. The surface temperature in the cargo bay exposed to the sun can reach temperatures as high as 150°C. Thermal shielding of the emulsions will place a limit on the lower energy of the trapped particles which can be observed.

4.1.2 Flight and Ground Restrictions

The experiment can and should be flown in a variety of different Shuttle orbits. The higher Shuttle altitudes (> 300 km) are preferable since they permit greater exposure of the emulsions to the Van Allen belt radiation. As pointed out previously, the emulsions require only a few days exposure. Long term exposure such as that provided by LDEF will produce overexposure. Thus, long duration orbiter flights are not required.

In order to fully evaluate the results of this experiment, the time history of the orientation of the emulsions relative to the geomagnetic field throughout the flight, will be required. For this purpose, the experimenter will probably require the attitude as well as the ephemeris of the orbiter for each mission. The emulsion should not be exposed to radioactive materials either in flight or on the ground.

These materials are often used for in-ground and in-flight calibration sources for cosmic ray and trapped particle experiments. The actual separation distance from radioactive sources should be precisely defined by the experimenters.

4.2 STP Integration Considerations

The only lead time required for delivery of this experiment is approximately one week's notice in order to obtain fresh emulsions. No preparation, testing of simulations for the flight are required. The experiment can be ready on short notice to participate in flights of opportunity.

In the simple experiment discussed in this assessment, no flight support equipment is required. An advanced version of the experiment may desire a magnetically stabilized platform to mount the emulsions so that the orientation of the emulsion with respect to the geomagnetic field can be selected and defined.

4.2.1 Cost Considerations

The Passive Energetic Particle experiment is a minimal cost experiment. It requires no Shuttle services (power, deployment, etc.) and can be flown on a space available basis. For this reason, the experiment should qualify as a "getaway special" for small self-contained payloads. The STS user charge for the small self-contained research and development payload, provided it weighs less than 200 lbs. and occupies a volume of less than 5 cubic feet, will be between $3,000 and $10,000.

5.0 RECOMMENDATION AND REMARKS

If this experiment is selected for flight, it is recommended that it be flown as an attached payload to a variety of orbiter flights, as the opportunities permit. The experiment should qualify for the small self-contained payload "getaway special" and therefore should have minimal cost. No problems are anticipated in accommodating this experiment to the STS.

SATELLITE MEASUREMENTS OF INFRARED AIRGLOW

1.0 EXPERIMENT IDENTIFICATION

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2.0 REQUIREMENT BACKGROUND

Response to STS Utilization Presentation (No. 34)

Identifying Number ILIR-7D

3.0 EXPERIMENT APPROACH

Perform limb scan using radiometer. Target altitude is 90 km with scan of ±40 km at 0.05 degrees per second. Continuous operation in sun synchronous polar orbit is desired but not required. The equipment consists of a high spatial resolution multi-channeled scanning radiometer mated to a 12-inch diameter high off-axis rejection cassegrain telescope with a 24-inch sun baffle assembly. A filter wheel assembly will provide several selectable spectral bandpasses.

4.0 ASSESSMENT FOR STS FLIGHT

No serious problems are expected in combining this experiment with other STS payloads. The principal points of interest are:

1) Acquisition of a tape recorder could be eliminated by using the STS.
2) High inclination orbit requirement constrains launch window to post 1982 when VAFB becomes operational.
3) Experiment operation is basically continuous; this will limit mixed payload co-passengers to experiments with compatible attitude and pointing requirements.

4.1 Experiment Considerations

4.1.1 Mechanical

The size and weight of the equipment required for this experiment, i.e., 2 ft. dia. x 5 ft. long and 150 pounds, should pose no mechanical problems. Possible accommodation alternatives are Orbiter bay pointing platform or free flyer. The use of a free flyer will require a deployment mechanism.

4.1.2 Thermal

The operating temperature ranges of each of the major assemblies can be readily accommodated using the Orbiter thermal control system, instrument heaters and thermal blankets. The sensor temperature requirement of -78° C would require thermal control within the instrument design.

4.1.3 Attitude Control and Pointing

The stability requirements of ± 0.1 degree in each axis are within the capability of the Orbiter. However, the accuracy of the pointing vector is approximately 0.5 degrees. It seems that the accuracy required for this experiment would be greater than that provided by the Orbiter. The instruments should be mounted on a pointing platform, SIPS or POINTS.

4.1.4 Communication

The use of the STS communication system could eliminate the need for the onboard recorder necessary for free flyer operations. Direct data transmission is available approximately 95% of the time using TDRSS; and if necessary, the Orbiter recorder could be used. The projected data rates of 12,000 bps are easily accommodated.

4.1.5 Command and Data Handling

All projected requirements are easily accommodated by the STS. Uplinked commands for instrument operations and down link data are minimal. No onboard data display or processing are required.

4.1.6 Power

Operating power requirements of 300 watts are well within STS power constraints. Electrical energy consumption is fairly high since the instruments have a 90% duty cycle. For example, for a seven-day Spacelab flight, approximately 39 kw hr. is required; this is about 10% of the total available energy for experiment operation.

4.1.7 Controls and Displays

None required.

4.1.8 Contamination

The environment in the Orbiter bay is generally not sufficiently clean for sensitive optical sensors early in the mission. Sensor covers should be provided for ground and early orbital operations.

4.2 Integration Considerations

The drawing shows a possible installation on the small instrument pointing platform in turn mounted on a Spacelab pallet. The SIPS has two instrument-carrying canisters, each supported at its center by a yoke which can rotate independently of the other canister in an up-down direction (120 degrees freedom). Each canister in turn is connected to the yoke so as to provide a limited ($\pm$10 degrees) left-right rotational degree of freedom. Both yokes are attached to a common ±180-degree azimuth gimbal drive at the base. An optional roll gimbal about the instrument line of sight can be added internally to each canister. The instrument package for this experiment could use the standard SIPS canister depending on the sensor cooling design selected to maintain the -78° C requirement.

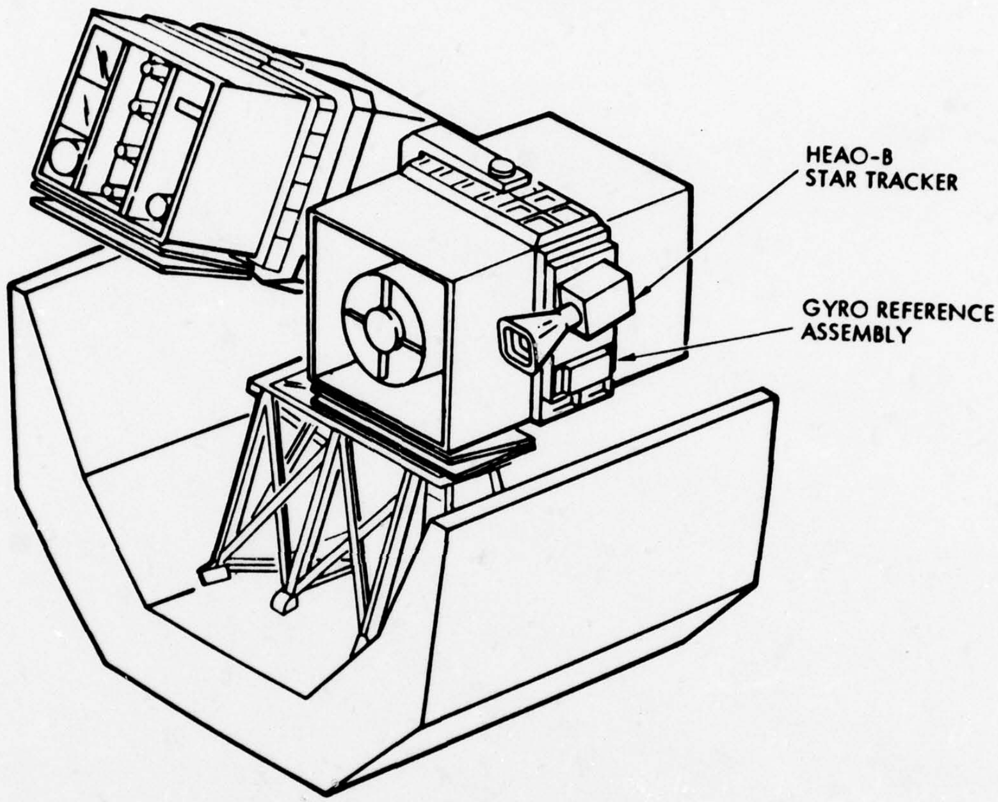

Figure 4-1. Small Instrument Pointing System

5.0 RECOMMENDATIONS AND REMARKS

1) Sensor thermal control should be provided by the instrument equipment.

2) Equipment should be mounted on a pointing platform such as SIPS.

3) Use of the STS communication system should eliminate the need for a tape recorder.

4) Sensor covers should be provided.

INFRARED BACKGROUND SENSOR

1.0 EXPERIMENT IDENTIFICATION

Drs. B. Schurin, S.D. Price and T.J. Murdock,
Principal Investigators
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Hanscom AFB, Bedford, MA 01731

2.0 REQUIREMENT BACKGROUND

Response to STS Utilization Study Presentation (Response Sheet #35)

Infrared Background Sensor Interface Control Document, Honeywell, Control No. F07401-74-C-0009.

An Infrared Deep Sky Survey in Four Infrared Wavelengths, Response to NASA AO OPPI-76-1, S.D. Price, T.L. Murdock.

An Absolute Infrared Calibration of Two Hundred Stars at Wavelengths Between Two and Fifty Microns, Response to NASA AO, 055-2-76, T.L. Murdock, S.D. Price.

3.0 EXPERIMENT APPROACH

The purpose of the Infrared Background Sensor Experiment (IRBS) is to obtain upper atmospheric long-wave infrared emission measurements using a highly sensitive cryogenically cooled IR detector. The measurements are made by scanning the earth's limbs in two different modes. In one mode, the limb is examined at fixed tangent height from the earth as the vehicle travels through its orbit and in a second mode, the instrument is scanned through various tangent heights over a short period of time to study the altitude dependence of the emissions.

The IRBS consists of a sensor, a liquid helium supply system, and associated electronics and power supplies. A pointing system is required and could be experiment provided if necessary. The experiment configuration is shown in a Spacelab installation in Section 4.2, Figure 4.2-1.

4.0 ASSESSMENT FOR STS FLIGHT

The susceptibility of the experiment to contamination of its optical surface is the primary experimental concern. This and the pointing requirements of the experiment are the principal design drivers for this experiment's accommodations. The size of the experiment and its pointing requirements make it an excellent candidate for a Spacelab flight or for installation in the Shuttle Oribter bay. However, the contamination requirements imposed by the instrument could be a problem. These considerations are discussed below.

4.1 Experiment Considerations

4.1.1 Contamination

The stated contamination requirements for this experiment are relatively severe. The conditions that are imposed on the instrument carrier in flight are as follows:

- Level 300A (MIL-STD-1246A) will be maintained for all surfaces around the exposed aperture.

- All gases expelled from the spacecraft's in-orbit thrusters and vents shall be filtered before expulsion and shall be directed away from sensors field of view.

- The spacecraft shall utilize as much as practical only those polymeric materials which produce $\leq$ 0.1% volatile condensible material (VCM) when tested, as discussed in NASA Report CR-89557, "Polymers for Spacecraft Applications."

It is not clear that these requirements can be met in the Orbiter bay. The first requirement can be assured as part of the experiment itself. Contamination from the Orbiter Reactive Control System (RCS) can be a problem when the system is being used for on-orbit operations. The thrusters are fixed in position. Filtering of their effluent is not practical. Another possible source of contamination is the flash evaporator which will dump water vapor overboard each day. The dumping occurs through two nozzles located at X_0 = 1507 and Z_0 = 291 on each side of the fuselage. The plume extends along the Orbiter Y_0 axis. Mounting of the instrument can be remote from these sources; and the sensitive instrument components can be covered during these events.

The actual outgassing properties of the Orbiter and Spacelab have not yet been determined, but the Spacelab Payload Accommodation Handbook does specify that material used in pallet mounted experiments meet the VCM < 0.1% requirement when tested by NASA specification J.S.C. SPR-0022. So one might assume that this requirement will also be met by the Spacelab components in the bay.

If this experiment is to be successful in the Orbiter bay, material control of experiments and components flying with the experiment will be required. NASA and ESA require a material control program for all Spacelab experiments. Furthermore, it will probably also be necessary to have covers capable of in-flight closure over sensitive instrument components. The instrument covers will be used during RCS operation, and flash evaporator dumping as required. The covers should not be open for about the first two days in orbit to ensure cooling down of the sensor and permit sufficient outgassing of on-board components.

4.1.2 Mechanical

The layout of this instrument will be similar to that shown in Figure 4.2-1. This figure shows a somewhat similar experiment from the same P.I.'s Spacelab 2 proposal. The sensor can fit in a cylindrical shroud 26" in diameter and 64" long. The He dewar is 46" in diameter and 10" long.

4.1.3 Pointing System

The requirements for pointing stems from the objective to examine infrared emissions from the upper atmosphere. In order to scan the earth's limb at fixed tangent height as the vehicle travels through its orbit or to scan through various tangent heights over a short period of time and study the altitude dependence of the emissions, a 3-axis gimbal system is required. Furthermore, the gimbal system must have the capability for field rotation (i.e., rotation about the line of sight [LOS]).

Four pointing systems are presently planned for Shuttle operation: the European instrument pointing system (IPS); the NASA/GSFC small instrument pointing system (SIPS); the NASA/LeRC annular suspension pointing system (ASPS): and the Air Force Payload Orientation and Instrument Tracker for Shuttle (POINTS). These pointing systems are described in Appendix A5. All systems, except POINTS, have a planned capability for field rotation; but since the IPS does not utilize roll around the LOS as the last gimbal, some care must be taken to prevent gimbal lock when pointing at targets close to the gunwhales of the Orbiter. Further, the IPS is very large and by itself, occupies approximately one-half of a pallet.

A roll gimbal option is planned for the GSFC SIPS as shown in Appendix A5. The SIPS comes in two different versions. One has a single yoke and the other a double yoke for accommodating two instruments or assemblies.

A comparison of the IRBS requirements and capability of various pointing systems is shown in Table 4.1-2. An examination of this table shows that all the IRBS requirements can be met by a single yoke SIPS if a 10 percent reduction can be made in the experiment and He dewar mass. In this case, the dewar would be mounted behind the sensor on the gimbal system but behind the yoke. This will require a special thermal canister. In order to minimize the heat leak to the instrument, a passive canister having low absorbtivity and high emissivity will be required.

The NASA/LeRC annular suspension pointing system (ASPS) should also be considered as a candidate for the pointing system. It has a 600 Kg weight capability and is more compact than the IPS. A picture of the ASPS is also shown in Appendix A5. In this design, a three-axis magnetically suspended platform is mounted on top of a two-axis (elevation and cross-elevation) mechanical gimbal set. In operation, the suspended platform is only optically coupled to the base for data transfer, and the payload power is supplied by batteries. This should not be a problem for IRBS since power consumption in the instrument must be minimized to conserve cryogenic fluid.

The POINTS system should also be considered. POINTS has an advantage in that the gimbal assembly has been built and tested. The POINTS system can accommodate the IRBS in all aspects except that POINTS does not have a roll gimbal. Capability to rotate about the line of site would have to be provided within the instrument, however, commands to this axis must be generated in conjunction with pitch and yaw commands within the POINTS control loop.

4.1.4 Power

The power required by the experiment is 138 watts with 150 watts peak. This included power is estimated by the experimenters for the gimbal system. No problem in providing this power from Spacelab or the Orbiter is anticipated. It is assumed that a major fraction of the power dissipated will be outside of the thermal canister.

TABLE 4.1-2

POINTING SYSTEM REQUIREMENTS VS. POINTING SYSTEM CAPABILITY

	IRBS REQUIREMENT	SIPS CAPABILITY	ASPS CAPABILITY	POINTS CAPABILITY
Envelope	Sensor: .66 x 1.62M He Dewar: 1.17 dia x .25M	.9 x .9 x 3.0M	Sensor size relatively unconstrained. 1 Meter dia. interface ring	2 M Width
Weight	675 kg	600 kg/yoke	600 kg CM < 1.5 meters from the gimbal	680 KG or two x 340 KG
Pointing Angle Access	Hemispherical	Hemispherical	$\pm 100°$ elevation $\pm 60°$ cross-elevation	-60°, +90° elevation $\pm 180°$ azimuth
Scan Rates	$\sim$ 0.1 deg/sec	Up to 2 deg/sec	TBD	2 deg/sec.
Pointing Stability	Fine tracking to within 20 arc sec.	± 0.3 arc sec ($1\ \sigma$)	± 0.01 arc sec ($1\ \sigma$)	Can be accommodated, stability depends on pointing sensor used.

4.1.5 Flight Restrictions

The orbit for the IRBS can have any nominal Shuttle altitude, but the inclination should be chosen so that the auroral zones and at least one earth's pole can be seen. At the low shuttle orbits, this implies that the inclination of the orbit should be about 70°. Operational flights with orbital inclinations greater than 56° will be launched from VAFB starting in December 1982. If this experiment must be done before that time, the only alternative would be on a spacecraft using an expendable booster.

Major crew support anticipated will be to orient the Shuttle conjunction with the limb scanning. The crew will also be required to turn the instruments on and off and to open and close the cleanliness covers.

4.1.6 Ground Restrictions

The major ground operation problem will be concerned with the prevention of contamination and frosting. Cleanliness facilities and clean handling procedures will be required whenever testing must be performed with cleanliness covers removed. The instrument will have to be in vacuum when cooled to prevent frosting.

It will take about 20 hours before launch to cool down the system after which it will be required to reload the liquid helium. This total operation will take about 30 hours. This can be a serious problem since very little time will be available before launch for payload access.

The approach proposed by TRW for integration of a similar experiment on AMPS can be used to solve this problem. Two on-board tanks are used. A large tank mounted on the pallet replenishes the cryogen supply of the gimbal mounted tank. Once in orbit and just prior to the deployment of the SIPS, the feed line between the tanks is severed. In this manner, sufficient cryogen can be provided without excessive tank weight on gimbal and without flexible cryogen lines. This also alleviates the need for pre-launch instrument access.

2 STP Integration Considerations

A Spacelab pallet installation of an instrument similar to the IRBS is shown in Figure 4.2-1. The layout shown is for a similar experiment proposed for Spacelab 2 by the same investigators. The installation for IRBS would be the same except that a SIPS may be used with the dewar mounted behind the yoke and behind the instrument. A second He tank would be mounted directly to the pallet hard points. As discussed in Section 4.1.6, this tank would be coupled to the primary dewar by a severable cryogenic line.

If this instrument is flown in the Orbiter bay, some additional costs over the basic Shuttle charges will accrue due to the requirements for the following optional Shuttle services:

(a) Use of Spacelab or other special equipment, i.e., the SIPS pointing mount.
(b) Payload data processing.
(c) Launch from Western Test Range for high inclination orbits.

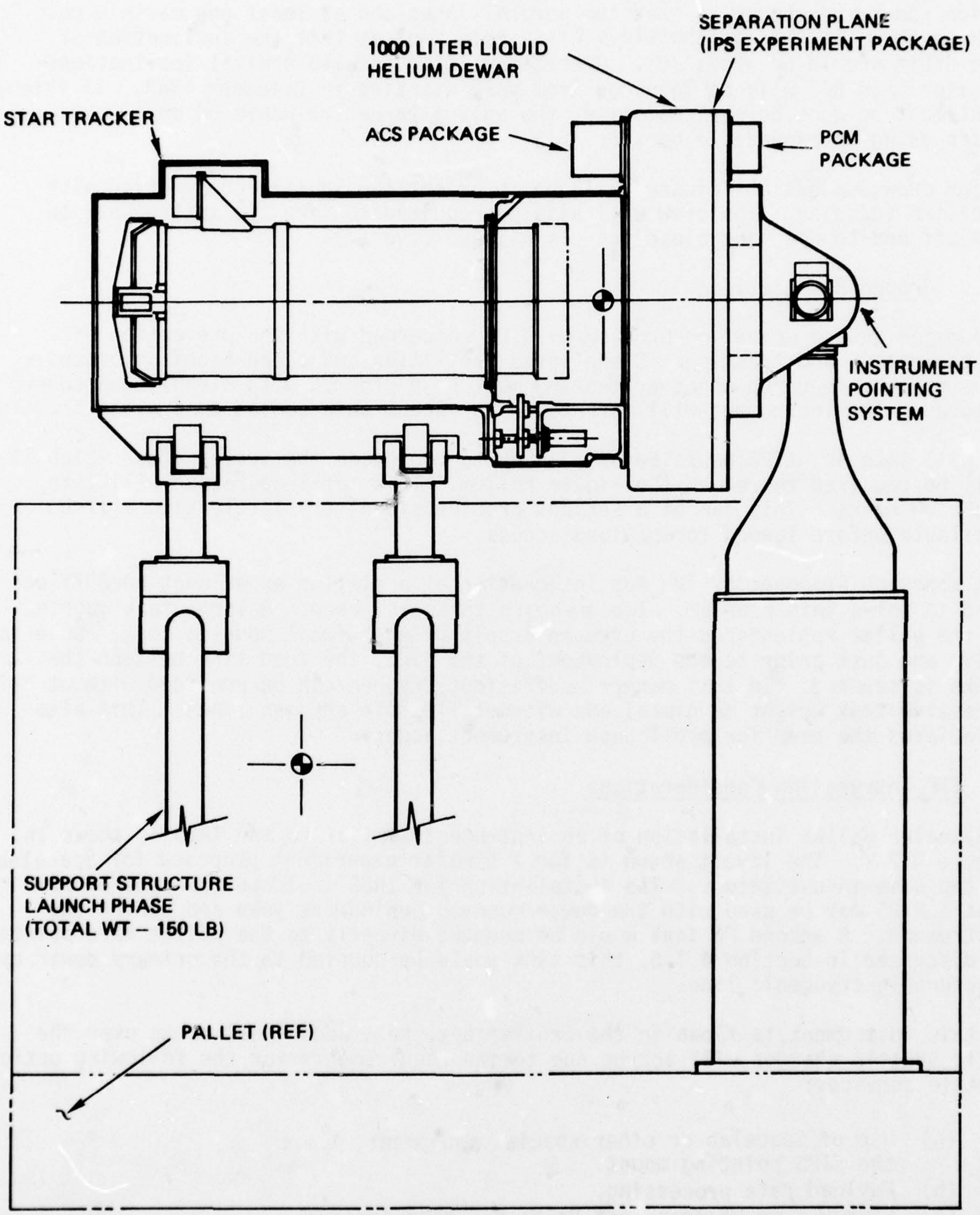

Figure 4.2-1. Layout of IRBS as Proposed for Spacelab II

The IRBS could benefit from being on the same mission with the VM Cryo Cooler and Radiator (R. E. Harris and W. L. Haskins AFFDL/FEL Response Sheets #28 and 29) and the Heat Pipe/Thermal Energy Storage experiments (T. Mahefky AFAPL/POE, Response Sheet #27) assessed in this study. If this joint mission were performed, the IRBS could use the cooling provided by the refrigerator, thus eliminating the need for the liquid helium supply system.

5.0 RECOMMENDATIONS

It is recommended that the IRBS be flown on a Spacelab type pallet using a SIPS. if it can be shown that the contamination requirements can be met. The details of the installation are discussed in Section 4.1. The requirements that would be imposed by this experiment on a free flyer would be very costly.

It is not clear that the contamination requirements imposed by the experiment can be met in the orbiter bay. The experiment will, of course, have to provide protective devices, covers, shrouds, etc., to protect highly sensitive components. At least a couple of days in orbit may be necessary for cooling and outgassing. Appropriate measurements of the environment should be made before the covers are removed. Payloads selected to fly with this experiment will have to be chosen with cleanliness in mind. Inherently "dirty" experiments, such as gas lasers or propulsion experiments, must be excluded. Some material control will be imposed on Spacelab experiments by NASA. Similar control should be imposed by STP.

On the other hand, if the orbiter bay can be used for this experiment, the advantages in both cost and accommodations over a free flyer are great. The SIPS pointing system already being developed can be utilized. Advantage can be taken of the crew to change experiment modes and assist in pointing operations. Furthermore, cost reductions should accrue as a result of simplification in experiment design because it can be returned and reused. The requirement for viewing the auroral zones and at least one pole will require a VAFB launch and thus will move the launch of this experiment out past 1982.

ENERGETIC PARTICLES AND FIELDS EXPERIMENT

(CHARGED PARTICLES)

1.0 EXPERIMENT IDENTIFICATION

Paul Rothwell, Principal Investigator
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2.0 REQUIREMENT BACKGROUND

Response to STS Utilization Presentation (Response Sheet No. 36)

Task 2311G1

Energetic Particles and Fields

3.0 EXPERIMENT APPROACH

The purpose of this experiment is to determine the impact of high energy cosmic rays on spacecraft components. This portion of the effort consists of the measurement of cosmic rays with energies greater than 100 MeV. The measurements are performed with an array of Cerenkov and plastic scintillation counters. Cosmic rays are analyzed in energy, specie and flux. The experimenter plans to store the data on tape for evaluation on the ground. Details of this experiment have not yet been completed and planning is in a preliminary stage.

4.0 ASSESSMENT FOR STS FLIGHT

There will apparently be no significant problems in accommodating this experiment in the orbiter bay as part of an attached STS payload. It is not particularly sensitive to, nor will it unduly interfere with, most Orbiter bay experiments.

4.1 Experiment Considerations

4.1.1 Mechanical

The envelope of the Energetic Particle and Fields experiment is a 3' (91.4 cm) cube. The total instrument weighs 310 lbs. (680 kg). There should be no problem in designing this experiment to withstand shuttle vibrational requirements since devices used in this type of experiment have been previously designed for levels as severe as those imposed by the orbiter.

The orbiter cargo bay acoustical environment reaches a maximum of 145 db near lift-off. The maximum vibrational levels occur between 2 and 8 Hz and therefore the experiment to hardpoint structure should be designed to increase the natural experiment/hardpoint structure frequencies above 12 Hz. The mounting of this experiment in the bay can readily be accommodated using a standard test rack bridge.

4.1.2 Power and Thermal

The experiment requires a total power of about 20 watts. There are no significant power variations. This can easily be accommodated through the orbiter electrical interface.

Power at 28 V is available at several stations in the cargo bay. This power is provided to the payload from fuel cells. The power quality is not very good, i.e., a voltage range of 27 to 32 volts can be anticipated. Peak voltage can be as high as 39 volts for loads less than 2 KW on a dedicated fuel cell. Therefore it will be necessary for the experiment to provide its own power regulation unless such regulation is provided to the experiment as part of an STP shared payload.

The temperature requirements for this experiment are typical of that required for electronic equipment, i.e., -30°C to 40°C. These temperatures can be met either by using a cold plate or providing passive control for the hot case and internal heaters for the cold case. The cold plate will maintain the temperature of the instrument between 0 and 40°C. The second approach requires a detailed thermal analysis of the instrument for heater design and α/ε selection. Use of a cold plate will probably be the more cost effective approach.

4.1.3 Data and Commands

The data from this experiment is digital and the rate should be well below 10 kilobits/sec. It is not required that the data be received in real time. The data can be collected by the PCM master unit and can be transmitted via the KU band link and TDRSS. Commands from the orbiter to the experiment can be provided to meet the ground to experiment command requirements of less than 10 commands.

4.1.4 Safety

No problems are anticipated in meeting the safety requirements for STS payloads. The Payload Safety Guidelines Handbook (JSC-11123) should be used as a guide in designing the experiment.

4.1.5 Viewing and Pointing

The experiment should not view in the direction of the earth when data is being taken. The instrument need not be pointed accurately and a knowledge of the pointing direction of about 2° is required. The Orbiter Guidance, Navigation and Control System can point any vector defined in the Orbiter Navigation Base Axis System at any desired inertial direction or in the direction of the local vertical to within $\pm 0.5°(3\ \sigma)$ half-cone angle. This does not include the alignment between the Navigation Base and the instrument. It is anticipated that there will be a structural relaxation effect when in orbit which will add an uncertainty of 2° to alignment between the Nav Base and the instrument. This combined with the accuracy of the navigation system itself provides approximately the 2° accuracy specified. If it is determined that the alignments cannot be made with sufficient accuracy, the system is designed to compute and process error signals from a payload mounted sensor. For this experiment it is not anticipated that an additional sensor will be needed. In any case Orbiter Inertial Measurement Unit ([illegible]) updates will be required roughly each hour.

4.1.6 "Smart" Standard Test Rack (STR)

A "smart" Standard Test Rack (STR) is being studied for STP. The plan is for STR to provide structure power, thermal control, telemetry and pointing independent of Shuttle services. This type of STR will be able to provide all the services required by the Energetic Particle and Fields Experiment free of the orbiter. At the present time, however, this rack is only in the study phase and is not part of the available facilities for STP payload use.

4.1.7 Flight and Ground Restrictions

The principal concern in this experiment is the effect of background counts from radiation belt charged particles. The signal levels in Cerenkov counters are small and background counts can be produced by:

(a) Scintillations in photomultiplier tube faceplate.
(b) Cerenkov radiation in photomultiplier faceplate.
(c) Direct bombardment of photocathode by penetrating particles.

To prevent background effects due to penetrating particles, the orbit should be as low as possible. At altitudes of about 200-300 km and at low inclinations penetrating charged particles are only encountered in a small region of space about 20° in latitude and 20° in longitude in the South Atlantic anomaly. At higher altitudes, this region becomes increasingly extensive. Furthermore, at low altitudes, we do not anticipate any significant dead time in the plastic scintillators since the pulsewidth in those devices is very fast (nanoseconds).

There will be no problem with radiation induced into the vehicles by the radiation belt protons since the products of that kind of reaction have energies well below the 100 MeV threshold of this experiment.

The experiment should remain in orbit for at least one week to permit accumulation of a statistically significant number of events.

No support is required by the flight crew - except for vehicle orientation and perhaps instrument turn-on/off and mode change.

No unusual ground support is required.

4.1.8 Cost Considerations

A number of optional services is provided to experiments at a cost over the basic charge. It does not appear that the Energetic Particles and Fields experiment will require many of these services. Those optional shuttle services that have been identified are:

- o Additional time on-orbit (more than one day) and
- o Payload Data Processing
- o Payload mission planning other than launch, deployment and entry.

The second of these may not be necessary, particularly if the "smart" STR is available. Payload mission planning will be required to insure that the instrument has adequate operating time while viewing in the anti-earth direction. This will require coordination of time line activities with the orbiter.

4.2 STP Integration Considerations

A conceptual drawing of the Energetic Particles and Fields experiment mounted in a Standard Test Rack Bridge is shown in Figure 4.2-1. The bridge mounts to hard points in the orbiter bay and provides adequate volume and viewing for the experiment. The rack, as shown, can accommodate several other small experiments and can conveniently fit into other orbiter payload groups.

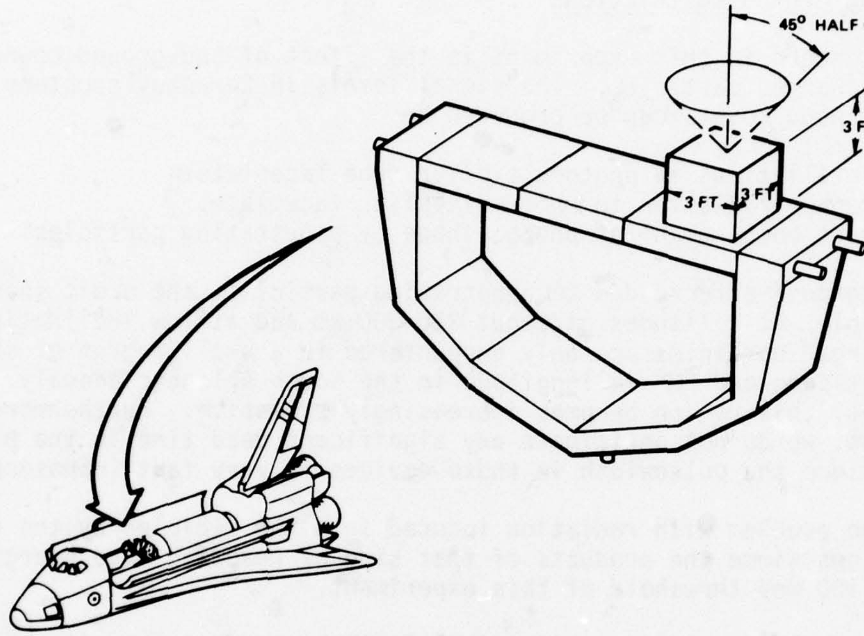

Figure 4.2-1. Energetic Particle Detector on Standard Test Rack Bridge

The Energetic Particle and Fields experiment has requirements very similar to many other high energy astrophysics experiments discussed by NASA for Spacelab flights. This is particularly true for the radiation contamination, the anti-earth viewing, and the desire to remain in orbit for more than seven days when possible. For this reason, this experiment should be considered as part of a payload made up of high energy astrophysics instruments.

If a "smart" STR is not available, cabling will be required between the instrument and the appropriate orbiter stations. Clearly planning for these interfaces cannot be made until the experiment has been included as part of a payload.

This experiment by itself need not require any flight support but when considered as part of a larger payload, it may be more cost-effective to combine several common functions into common flight support elements. For example, pointing accuracy can be improved with a common star sensor. Common data handling tape recording and power processing functions can be incorporated for cost-effectiveness.

A similar argument holds for the ground support equipment (GSE). Experiment unique GSE should be provided by the experimenter. This should include:

Unit Testers
Environmental Test Fixtures
Special Experiment Handling Fixtures
Alignment Fixtures

In the case where the experiment is part of an integrated payload, common GSE such as handling and checkout equipment for common payload elements, will also be required.

5.0 RECOMMENDATIONS AND REMARKS

If this experiment is selected for an STP flight, it is recommended that this experiment be flown in a Standard Test Rack in the orbiter bay. A large amount of simplification would occur in the instrument/orbiter interface if the "smart" Standard Test Rack being planned by STP were available for this experiment.

The requirements of the Energetic Particles and Fields experiment are compatible with those of a large number of high energy astrophysics experiments. Therefore, it is suggested that this experiment might be incorporated into a shuttle mission carrying experiments in that discipline. Two astrophysics missions (APP) are planned by NASA. These are tentatively scheduled for late 1981 and late 1982.

In conclusion, no difficult problems are foreseen in the performance of this experiment using the STS.

1.0 <u>1. MEV ALPHA PARTICLES TRAPPED IN THE MAGNETOSPHERE</u>

<u>2. MATERIALS EFFECTS ON SPACECRAFT CHARGING</u>

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2.0 REQUIREMENT BACKGROUND

Response to STS Utilization Presentation (Response Sheet No. 37)

Projects:

1. Mev Alpha Particles Trapped in the Magnetosphere - 2311G102
2. Materials Effects on Spacecraft Charging - 766107
3. Mev Range Energy Telescope flown on Spacecraft S3-3

3.0 EXPERIMENT APPROACH

The dual objectives of this experiment are achieved with an instrument of the type that has been developed and flown previously by the experimenter. The instrument is a two-element Mev range energy telescope using scintillation detectors.

The first objective, that of characterizing the flux of the trapped Mev alpha particles is a continuation of current experiments in mapping the spatial and temporal extent of these high energy particles. The second objective, that of evaluating the effects of these particles on spacecraft charging is related to the long term effects that these high energy particles may have on materials characteristics. Spacecraft charging may affect the quality of data in low energy particle detection experiments, and also has been shown to present a hazard to spacecraft survivability at geosynchronous altitudes because of the resulting arc discharges. Spacecraft materials characteristics such as bulk and surface resistivity and photoemission and secondary emission are intimately related to how the spacecraft responds to the environmental plasma. Information on how these materials properties are changed by long term exposure to high energy particles is unavailable at the present time, and this experiment will provide some of this type of data. Designing spacecraft which are to be operational for as long as ten years in geosynchronous orbit requires that materials parameters over this time span be known. The Shuttle launched long duration exposure facility (LDEF) would be an ideal STP mission for this experiment. Figure 3-1 shows the two parts of this experiment on the Shuttle and the LDEF.

For the materials effects study portion of this experiment on the LDEF, the dosage of the characterized high energy particles will be determined and the effects on the material parameters will be studied after recovery. To minimize the experiment data recording capacity requirement, the instruments will be turned on only periodically. The data obtained will be correlated with that obtained on the Shuttle, and the dosage of environmental high energy particles will be extrapolated.

The experiment characteristics are summarized in Table 3-1.

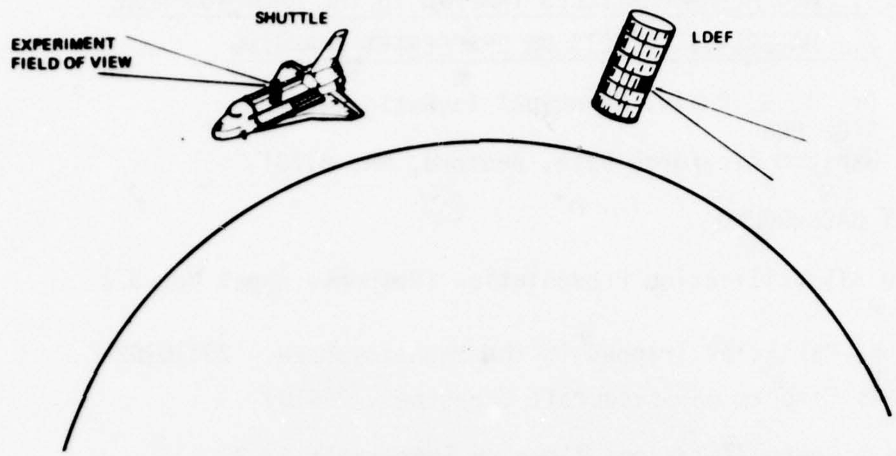

Figure 3-1. Experiment on Shuttle Defines NEV Alpha Particle Environment LDEF Experiment Defines Materials Effect of Environment

Table 3-1. Experiment Characteristics

SHUTTLE PORTION	
Instrument:	1' x 1' x 1'; 25 pounds; 20 watts
Energy Range:	18 Mev to >70 Mev
View Angle:	± 10° clear field of view: preferably perpendicular to the local geomagnetic field
Telemetry:	1 kbps, digital; 4 analog channels for housekeeping data
Commands:	One for on-off
Orbit:	Equatorial ±40° inclination; altitude greater than 200 km; as high as possible up to 4000 hours
Crew Time:	Essentially none
Operational Time:	7 days; as many launches as possible
Additional Data:	Requires spacecraft location and attitude information; also local magnetic field data
LDEF PORTION (ONE TRAY)	
Instruments:	Mev range energy telescope as for Shuttle kev range energy particle detector Magnetometer Spacecraft materials exposed to environment
Auxiliary Equipment:	Experiment programmer/timer Power supply Tape recorder
Weight:	150 pounds
Tray Size:	50" x 38" x 12"

4.0 ASSESSMENT FOR STP FLIGHT

4.1 Experiment Considerations

4.1.1 Design Suggestions

The instrumental portion of this experiment has been developed to the point where it has already been flown on unmanned satellites. For STP flights the following design suggestions are appropriate:

- Design for low cost in view of increased weight and power availability, while eliminating hazards to crew and to the orbiter.
- Design of the LDEF portion of the experiment should be such that the flux of high energy particles is only checked periodically with on-board sensors to save on power and data acquisition capacity. These readings would verify the environment as inferred from the Shuttle borne part of the experiment.
- The influence of other factors such as UV irradiation and other species of high energy particles on materials parameters should be evaluated and taken into consideration.

4.1.2 Operation Restrictions

For flight, the requirement for higher altitudes, greater than 200 km and less than 400 km and orbit inclinations less than $\pm 40°$ is not particularly restrictive and only requires that this experiment be flown with the appropriate Shuttle payloads. Although view angles perpendicular to the local geomagnetic field are desired, the experimenter is willing to live with all angles as long as they are determined. A sensor pointing capability is not required. In flight, only minimal attention is required from the crew, and therefore no restrictions arise from this consideration. On the ground the main requirements are the capability to observe the data being taken and to execute commands such as on-off and operating mode changes.

4.1.3 Experiment Support Equipment

Since this instrument has been flown previously, support equipment is already available. Some modifications to accommodate Shuttle and LDEF requirements on accessibility will have to be worked out. Primarily, this involves pretesting and precalibration at the bench level and foregoing of integrated systems testing to a major extent.

4.2 STP Integration Considerations

This instrument requires a clear field of view of $\pm 10°$. Other integration interfaces such as power, thermal, commands, and telemetry should present no special problems. Having been flown previously, the delivery lead time should be minimal since there is no development involved.

Normally, a radioactive calibration source is used for prelaunch testing. It would be desirable to perform this step at the experimenter's facility at the unit level and eliminate it for all subsequent procedures. Re-calibration after the flight is possible.

For the LDEF portion of this experiment, the equipment will be self-contained and have its own programmer timer as well as power source and recording equipment. Equipment of common design for the power supply and data recorder are being proposed at low cost.

4.2.1 Conceptual Layouts

The conceptual layouts of the Shuttle mounted instrument is shown in Fig. 4-1. The entire unit is self-contained and only requires power, command and telemetry interfaces. A 10° clear field of view is required. Figure 4-2 shows the LDEF tray configuration. It is completely self-contained with its internal power source, batteries and data tape recorder. These are sufficient for a one-year mission. Except for a section of the external surface reserved for instrumental apertures, the major portion is reserved for various surface materials whose environmental exposure degradation characteristics are to be measured.

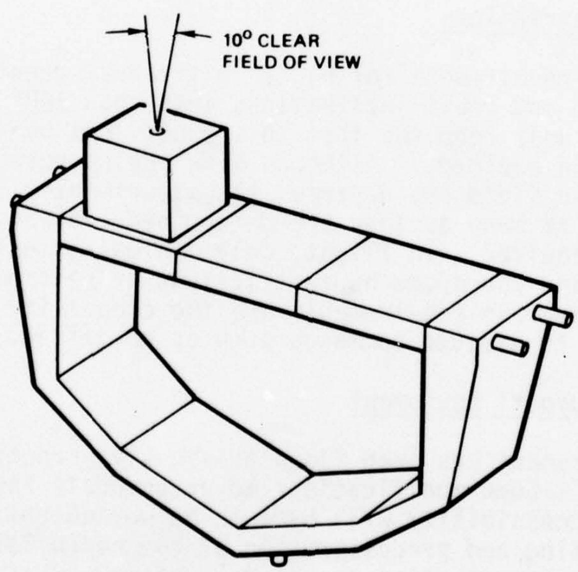

Figure 4-1. MEV Energy Telescope Mounted on Standard Test Rack

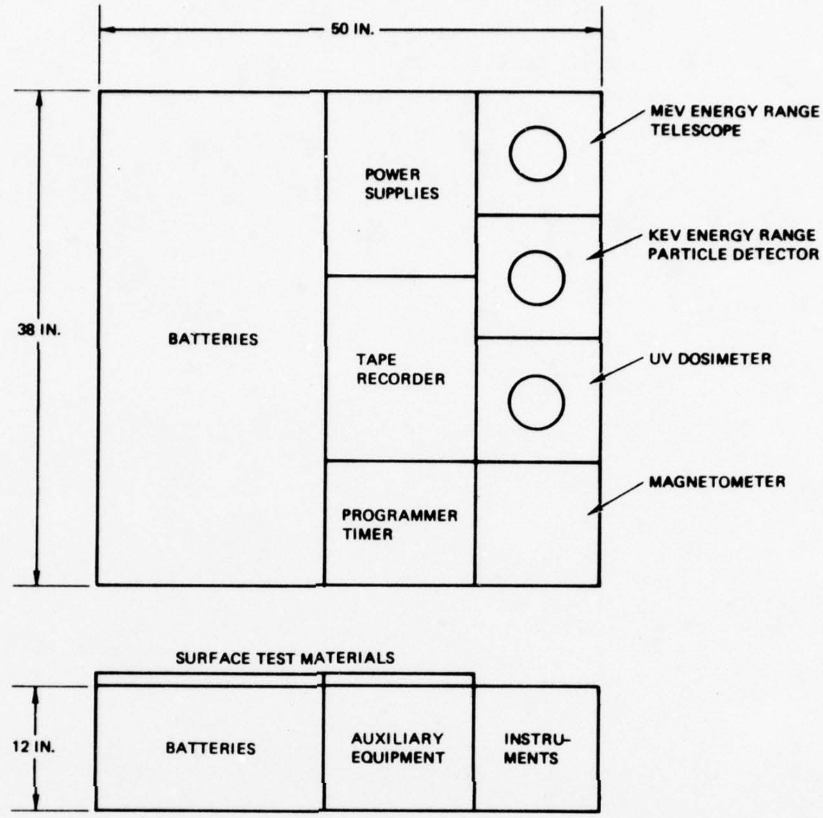

Figure 4-2. LDEF Tray Configuration

4.2.2 Flight Verification

The flight worthiness of the proposed instruments have already been established and should pose no problems for the Shuttle or the LDEF. For the LDEF tray configuration, the LDEF launch environment specifications must be verified at the tray level before integration into the LDEF.

5.0 RECOMMENDATIONS

This experiment is recommended for an early STP flight because of its prior development and minimal associated integration costs. The data to be gathered is of scientific interest and provides valuable engineering data as well, for the design of geosynchronous spacecraft. The expansion of the scope of the LDEF portion of the experiment should be considered to cover other factors in the environment such as UV irradiation, and other species of high energy particles such as electrons and protons.

CONTROLLED ARTIFICIAL DEPLETION OF THE IONOSPHERE

1.0 EXPERIMENT IDENTIFICATION

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2.0 REQUIREMENT BACKGROUND

Response to STS Utilization Presentation (Response Sheet No. 38)

3.0 EXPERIMENT APPROACH

This experiment will be performed in order to measure the response of the ionosphere to the injection of a large mass of gaseous material at high velocity. The materials to be used would be either water vapor or ammonia. A canister containing the liquid phase of the material would be orbited, then the liquid would be allowed to evaporate to fill a large volume balloon. Subsequently, the balloon would be torn open and the gaseous material would be allowed to interact with the ionosphere. Once the material in the balloon has slowed down by collisions with the ambient atmosphere, it would interact chemically with the ionospheric electrons in such a fashion as to deplete the electron content. The effects of this interaction would be studied both by instruments passing through the cloud shortly after the release of the gas and by ground-based incoherent backscatter radar. The radar measurements are a necessary part of the experiment because they will be able to continue a long term observation of the behavior of the cloud.

This experiment is intended to be performed from the Shuttle. The canister, together with the uninflated balloon, would be orbited on the Orbiter and then released. A subsatellite which would be equipped with diagnostic instruments would also be launched from the Orbiter so that it would be able to fly through the expanding gas cloud several seconds after the release. The flight experiment sequence must be well coordinated with the ground operation of the radar.

The canister and the uninflated balloon would weigh about 1200 kg and before inflation, would have a volume of about 3 cubic meters. This weight includes the weight of 1000 kg of water. After the canister has been launched from the Shuttle, the water vapor will begin filling the balloon. The final size of the inflated balloon will be about 20 meters in diameter and 120 to 150 meters long in a roughly cylindrical shape. The diagnostic instruments mounted on the subsatellite would consist of thermal electron and ion Langmuir probes and an ion mass spectrometer. The size of the subsatellite would be roughly 50 cm in diameter by 20 cm high and and would weigh about 50 kg total. There is also the possibility that a set of diagnostic instruments identical to those on the subsatellite would be mounted on the Shuttle. The size of this package would be about 30x30x15 cm and would weigh about 15 to 20 kg.

The operations involved in this experiment relate to the proper timing of the release so that it will be over the chosen ground station, Millstone Hill, at the time the gas is released from the balloon. One of the requirements is that the balloon be in sunlight for at least about one-half hour so that the water vapor can be heated above the freezing point. Further, there is a requirement that the diagnostic package be located so that it will pass through the release cloud shortly after the release. The details of these operational requirements will be discussed in section 4 below. The desired orbital altitude is in the range from 200 to 500 km where the dominant ion is oxygen. The time of the release should be about twilight in the evening sector, 1700 to 2100 LT.

The only pointing requirements for this experiment involve the orientation of the balloon-canister package at the time that it is ejected from the Shuttle. The same is true for the diagnostic subsatellite. 2° is sufficient accuracy for these functions. The subsatellite would be spin-stabilized and its spin axis would have to be known to an accuracy of 0.2° during the period of observation.

The electrical power requirements of this experiment are minimal. The diagnostic package on the subsatellite would require a total average power of about 20 watts during the observation period. This period would be no longer than fifteen minutes including the time required to check out the instruments. The energy required would therefore be about 5 watt-hours which could be easily supplied by the batteries on the subsatellites. The balloon-canister package would require no more than 10 watts with an energy need of about 3-5 watt-hours. All power can be delivered at 28 volts.

The experiment requires that a command signal be sent to the balloon-canister to start the filling of the balloon with water vapor, and another command to tear the balloon and release the gas. The subsatellite will output 16 kbps of serial digital data.

Before the time that the packages are ejected from the Shuttle, the canister filled with water should be kept within the temperature range from 0° to 100°C. If the canister is filled with ammonia, the temperature range would be from -40° to 60°C. The diagnostic instrument can be designed to operate in the temperature range from -40° to 40°C. The diagnostic instruments must be kept relatively clean before they are ejected from the Shuttle. The subsatellite will radiate S-band telemetry to the Shuttle. When the experiment is performed properly with the gas release occurring ahead of the Shuttle, there is no danger of the Shuttle contaminating the experiment. If the Shuttle were to fly through the release cloud, it would be exposed to water vaper at a very low pressure, or to ammonia at a similarly low pressure.

There are no special requirements for this experiment during the ground operations before the launch of the Shuttle. The subsatellite instrumentation should be kept in class 100,000 environment, and the temperature ranges for the balloon-canister package should be observed.

The balloon and the canister system is a relatively straightforward design problem and there are no foreseeable difficulties in being able to fly it at any date. The diagnostic instrumentation is of standard spacecraft design and similar instruments have flown often before. The investigators would like to recover the subsatellite in order to fly it again with another balloon gas release package. They have considered deploying a tethered diagnostic package instead of a free flying subsatellite. This will be discussed further below.

4.0 ASSESSMENT FOR STS FLIGHT

The balloon-canister package and the diagnostic package can be accommodated on the STS, and it appears feasible to launch them into the required experimental configuration.

The possible problem areas are the following: (1) The timing of the launching of the two packages from the STS in coordination with the observations that must be made from the ground by the radar; (2) there may be a problem in flying the Shuttle orbiter through the cloud that is generated by the release; (3) Maintaining continuous communication with both the balloon-canister package and the subsatellite might be difficult, and (4) the question of whether to fly a free flying subsatellite or a tethered one must be settled.

4.1 Experiment Considerations

4.1.1 Design Suggestions

At the present time, there does not exist a standard subsatellite of the size suitable for this experiment and the investigators would have to provide the subsatellite together with the instrumentation. However, for the purpose of this experiment, the required subsatellite could be very simple. It would be to use the standard STS S-band detached payload telemetry link and would require little in the way of power or attitude control systems. The launching of similar sized subsatellites has been studied in the phase B NASA AMPS studies and has been found to require little more than a spring-load ejection mechanism. The thermal control of the subsatellite would be relatively easy because of the short design lifetime of less than six days. The question of recovering the satellite is more difficult to answer. For this reason, the investigators are also considering the use of a tethered subsatellite that would be reeled out to a distance on the order of one kilometer from the Shuttle for the duration of the experiment. There are several problems associated with the use of a tethered system. One of these is that studies show that dynamic instabilities may arise during the reeling in of the tether. In the extreme, this could lead to wrapping the tether lines around the Shuttle. Another of the problems associated with the tether is the length of time that the Shuttle must remain in a single attitude position relative to the local vertical. This would impact the mission operations for other experiments. However, at the present time, since both free flying and tethered subsatellites have problems associated with them, a choice of which system to use cannot be made without further study.

In either case, the telemetry link between the subsatellite and the Shuttle would be carried out by S-band to the Shuttle's detached payload receiver. Since this link is limited to 16 kbps of science data, and during the traversal of the gas cloud the subsatellite might be gathering data at a much higher rate, there will be a need for a data recorder on the subsatellite. The total time involved in the traversal of the gas cloud would be less than five minutes so that the total amount of data that need be recorded should be less than a few megabits. The alternative would be to limit the data rate out of the instrumentation to the 16 kbps rate.

It is suggested that the water (or ammonia) canister be thermally connected to the Shuttle liquid cooling system in order to maintain proper temperatures. Alternatively, a study might indicate that passive thermal control will suffice.

It is suggested that the balloon-canister package be equipped with a timer system so that the commands to fill and tear balloon could be given some time before the actual events. In this way, the Shuttle detached payload S-band transmitter could lock onto the balloon-canister package to deliver the commands and then have time to lock onto the diagnostic subsatellite during the entire time that the subsatellite is passing through the gas cloud.

Finally, there is the question of the advisability of flying the Shuttle orbiter itself through the gas cloud. If the release gas is water vapor, there would be no possibility of chemical contamination of the Orbiter. Whereas, if it it were ammonia, there is a possible contamination problem, however, the possibility is low because the densities will be extremely low. There is another possible problem which is that as the gas release cloud hits the ambient atmosphere, its orbital velocity kenetic energy is converted into thermal energy and the cloud becomes relatively hot (on the order of a few thousand degrees C). However, the gas would be extremely tenuous and have low specific heat capacity. The additional heating to the Shuttle should be negligible. The only remaining hazard would be that the Shuttle could hit pieces of the fragmented balloon. But the relative velocity would be low and their mass small. The conclusion is that, at least on first analysis, there would be no danger of flying the Shuttle through the gas cloud in order to make further measurements of its properties. Further study of the Shuttle-cloud interaction should be performed, however, before it is planned to fly the Shuttle through the expanding gas cloud.

4.1.2 Operation Restrictions

The total operational time of this experiment would be less than two hours including the checkout of instrumentation. There would be some time between the launching of the balloon-canister package, the diagnostic subsatellite and the actual period of observation, but no experimental operations would be required during this waiting period. The balloon-canister package would be launched, probably from a spring-loaded launching platform, so that it would drift ahead of the Shuttle. The subsatellite would be launched with a similar velocity after a distance of 20 to 100 kilometers has opened up between the Shuttle and the balloon-canister package. The balloon would begin filling about 30 minutes before it was due to pass over the ground-based radar site. As it filled, its drag would slow it down and increase the distance between it and the subsatellite and Shuttle. The command for release would be transmitted to a timer on the balloon, and the transmitter would be returned to the satellite. At the proper time, the balloon would automatically release its gas over the ground radar and several seconds later, the subsatellite would pass through the cloud-making measurements. The Shuttle could either pass through the cloud or lower its orbit so that it would pass under the cloud. (The cloud should expand most rapidly upwards, bouncing off the higher density atmosphere below it.) This entire sequence of operations is quite straightforward and poses no problem for an STS flight so long as proper communication can be maintained with the ground radar station.

The ground operations in the integration of the payload before launch would be straightforward and make no special impact on the STS.

4.1.3 Experiment Support Equipment

The experiment would need some power if the subsatellite and balloon command system batteries were to be charged on the STS before launching. The monitoring of the status of the diagnostics and of the balloon-canister would require either hardwiring into the Orbiter aft flight deck or using a RAU connection if the experiment flies on a Spacelab flight. The STS detached payload telemetry system would be needed and would operate at two differing frequencies to be selected in sequence. The water canister would probably be connected to the fluid cooling loop through a heat exchanger. The data from the diagnostic subsatellite could either be recorded and recovered after the flight or be transmitted directly to the ground.

4.1.4 Experiment Cost Considerations

The cost of the balloon and the water canister system would not be great. The diagnostic instruments already exist. Flights with other balloons filled with differing chemicals would be planned if this first experiment were successful.

4.2 STP Integration Considerations

These instruments could be delivered on a short lead time. The basic experimental concept can only be tested in space, but the tearing of the balloon could be simulated in a large vacuum chamber. The experiment could be mounted at any hard point, on a Spacelab pallet or any standard test rack. The system which ejects the packages from the Shuttle could be either the responsibility of the investigator or of the integrator. Since there may be a need in several experimental areas to eject packages at relatively low velocities from the Shuttle, it might be reasonable to assume that STP will provide such a device as flight support equipment.

Figure 4-1 shows a sketch of the experiment instrumentation excluding the mechanism used to launch the packages from the Shuttle. Figure 4-2 shows the arrangement of the Shuttle, subsatellite, and balloon at the time of the release of the gas from the balloon over the ground radar station.

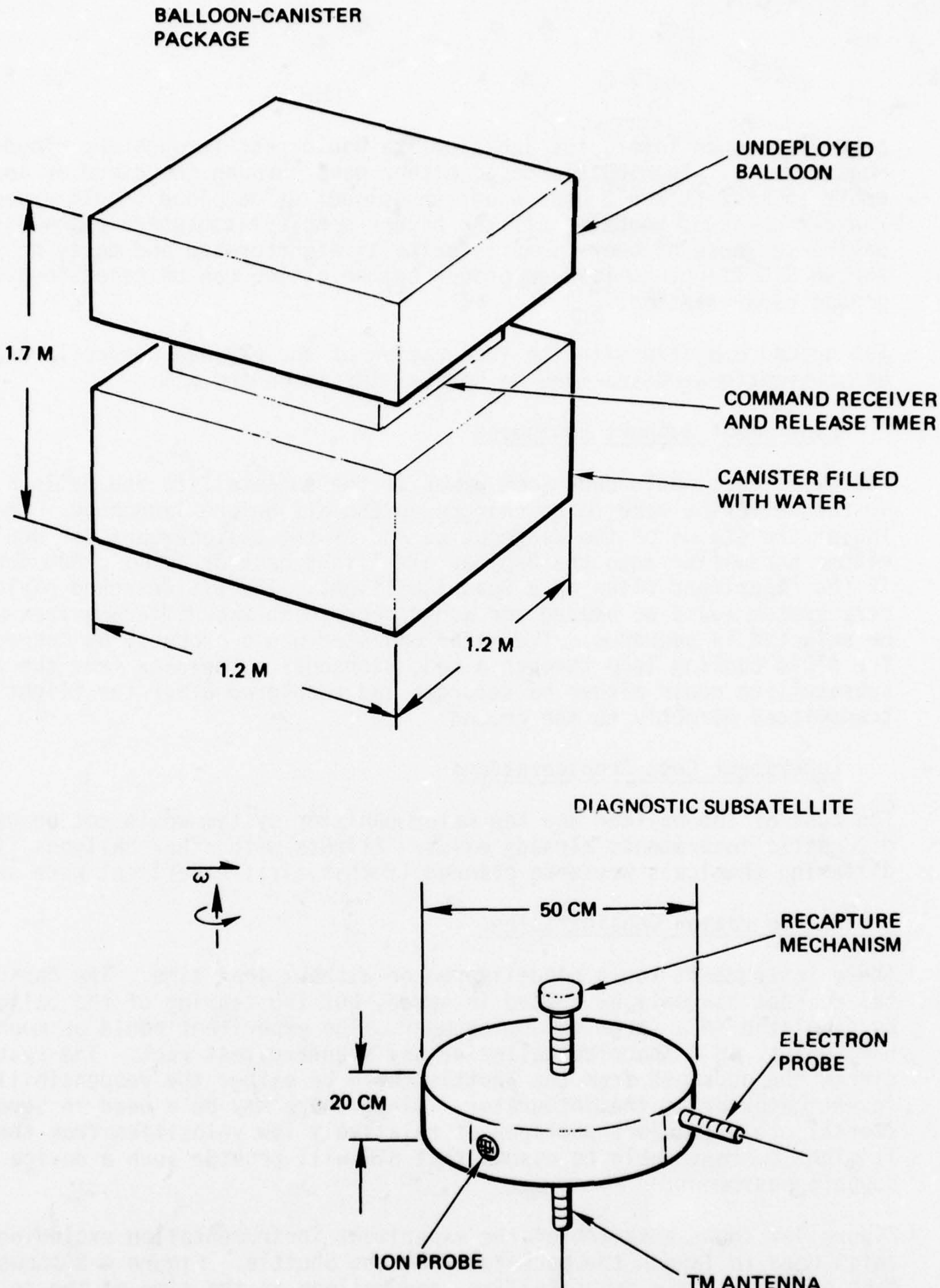

Figure 4-1. Sketch of the Instrumentation for the Controlled Artificial Depletion of the Ionosphere Experiment

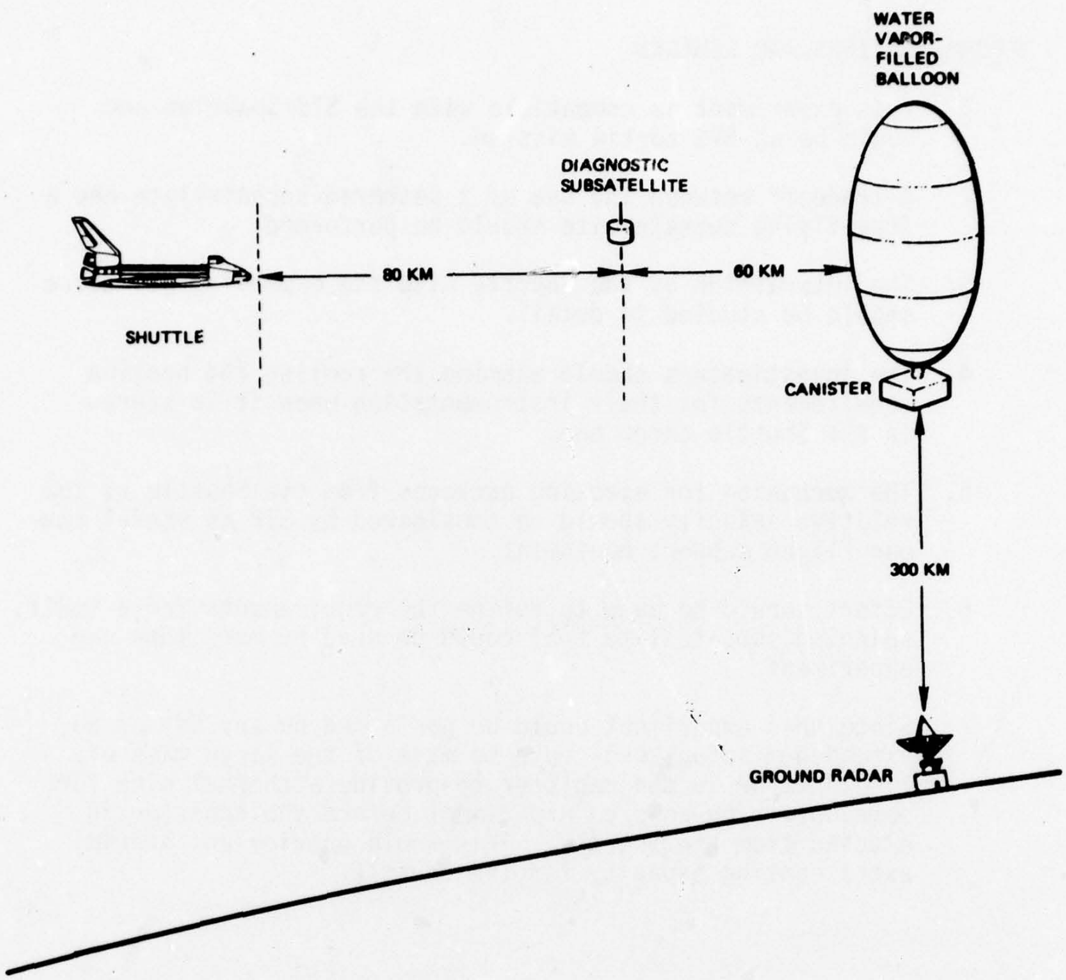

Figure 4-2. Conceptual Configuration of the Instruments just before Release of Gas in the Artificial Depletion Experiment

The following items must be purchased as optional STS services:

1. Either the tether system for the subsatellite or the retrieval of the free flying subsatellite. Cost tradeoff should be done to determine if it might not be advisable simply to leave the subsatellite since its recovery is not necessary for the performance of the experiment.

2. A heat exchanger may be needed and the mechanism for ejecting the packages from the Shuttle is needed.

3. There may be a need for additional mission planning services in order to coordinate the release with the ground station at the desired time of day.

5.0 RECOMMENDATIONS AND REMARKS

1. This experiment is compatible with the STS/Spacelab and could be an STS sortie mission.

2. A tradeoff between the use of a tethered subsatellite and a free flying subsatellite should be performed.

3. The interaction of the Shuttle with the expanding gas cloud should be studied in detail.

4. The investigators should examine the cooling and heating requirements for their instrumentation when it is stored in the Shuttle cargo bay.

5. The mechanism for ejecting packages from the Shuttle at low relative velocity should be considered by STP as useful common flight support equipment.

6. Effort should be made to define the requirements for a small, spinning subsatellite that could be used by more than one experiment.

7. Since this experiment could be performed on any day of an extended mission, use could be made of the large mass of liquid water in the canister to provide a thermal sink for some hot, high-powered experiment before the canister is ejected from the Shuttle. This would provide an interim, extra cooling capacity for the Shuttle.

SHEATH AND WAKE STUDIES

1.0 EXPERIMENT IDENTIFICATION

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2.0 REQUIREMENT BACKGROUND

Response to STS Utilization Presentation (Response Sheet No. 40).

This experiment is part of the Task: Electrical Structure of Aerospace, Task No. 2311G2 at the Air Force Geophysical Laboratory.

3.0 EXPERIMENT APPROACH

The intention of the investigator is to measure the wake and sheath properties of simple geometrical shapes as they pass through the ionosphere at orbital speeds. The instrumentation consists of a wake producing body and a set of ionospheric plasma measuring devices.

The wake producing body will be a 30 cm diameter inflatable sphere which will be mounted on a 10 meter long rigid mast attached to the orbiter at some point in the cargo bay. The diagnostic devices will consist of: (1) a spherical electron probe sensitive to electrons in the energy range from 0 to 40 ev.; (2) a planar ion trap, (3) an ion mass spectrometer, and (4) an electrostatic analyzer for electrons in the energy range from 1 to 100 ev. These diagnostics will be mounted in a single package that must be moved with respect to the wake body in order to map out the properties of the wake at various angles and distances from the center of the body. The diagnostics must be oriented so that during the measurements they are pointed into the orbital ram velocity direction in order to measure the ion properties of the wake.

The plan of operation is to erect the test body mast, orient the orbiter so that the test body lies outside of the effects of the orbiter's wake, and then move the diagnostic package through a set of positions that have been planned before the flight. Each observation of the wake would take one orbit period to accomplish, and the investigator would like to repeat the observations about twice a day throughout the flight. The crew involvement would consist only of commanding erection of the test body mast, preparing the diagnostics, initiating the diagnostic mapping, and shutting down the whole experiment before reentry. There is also a need for monitoring the position of the diagnostic package so that no safety hazards are generated during its mapping operations.

Figure 3-1 shows a sketch of the instrumentation. There is a 30 cm diameter inflatable wake body mounted to a rigid mast 10 meters long. The diagnostic package is roughly 45 cm by 40 cm by 35 cm, and is a self-contained unit with a telemetry system. The wake body needs no data or power connections to the orbiter. It is planned that the diagnostics transmit their data on the Orbiter's detached payload S-band communication link. The electrical power requirements are to be determined. The weight of the test body and mast is 13 kg. The weight of the diagnostic package is 21 kg.

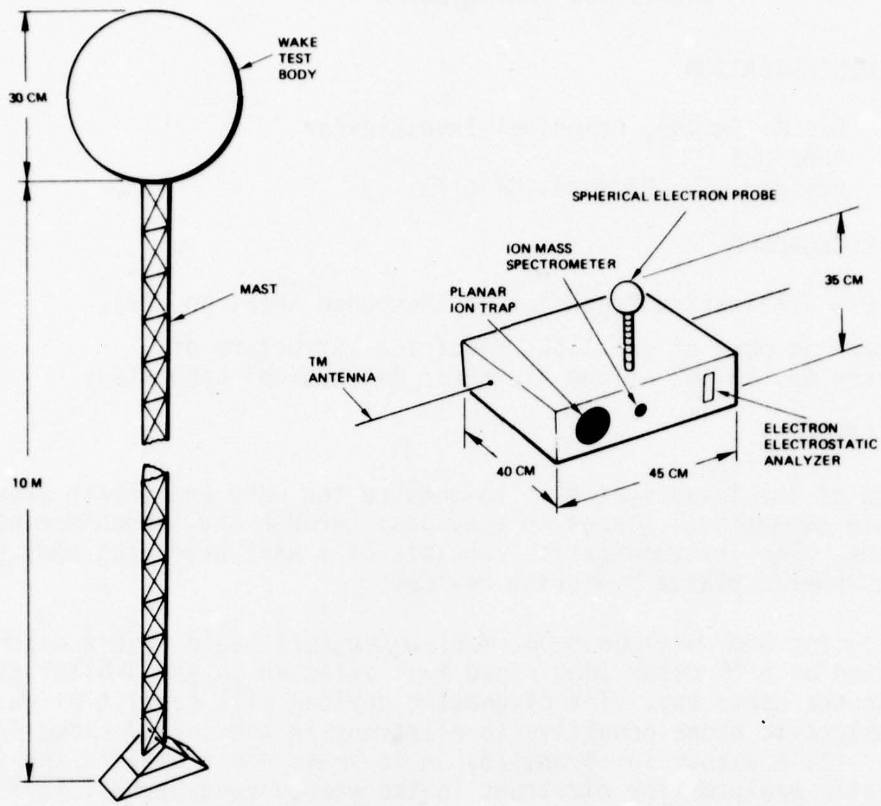

Figure 3-1. Sketch of Sheath and Wake Studies Instrumentation

The pointing requirements of the experiment involve the relative angles and displacements of the test body and the diagnostic package instruments. Angular resolutions on the order of 1°, and spatial resolutions on the order of a few centimeters are required. The mapping should be carried out in the region of space up to 10 meters from the center of the test body. However, it is only necessary to map one-half of the region because the wake is mainly symmetrical about any plane containing the center of the sphere and the ram velocity vector.

The test body will not need any electrical power. The diagnostic package instruments use an average power of 33 watts and a peak power of 35 watts. Less than 5 minutes of every observation period would be at the peak power level, and the remainder of the 90-minute period would be at the average power level. The package takes 28 volt power.

The test body would have only minimal data to monitor the erection of the mast and the inflation of the balloon, and even these could be done away with by simply requiring that one of the crew members visually checks that the mast is erect and the balloon inflated. The diagnostic package has its own data handling system whose output is an S-band telemetry signal to the orbiter's receiver. The telemetered data are in a serial bit stream with a bit rate of 12.4 kilobits per second during the period of observation.

Instrument commands consist of turning on the diagnostic instruments, maneuvering the relative position of the test body and the diagnostic package, erecting the mast and inflating the balloon. Before reentry, the balloon would be either ejected or deflated, the mast restowed and the diagnostic package would be stowed.

The instruments are designed to operate in the temperature range from 10° to 40°C and can be stored in the temperature range from -10° to 70°C. The diagnostic instruments are designed to operate at pressures of less than a few times 10^{-5} Torr. During the time the diagnostic package is exploring the wake of the test body, there is a requirement that the non-ambient magnetic fields be less than about 10 milli gauss. Also, care should be taken that dust and condensation outgassing products do not collect on the diagnostic package either during its operational or storage periods.

There is a definite reflight potential for this experiment. The diagnostic package can be used for other flights. The wake test body would be changed either by changing its shape, e.g., cylinders or cones, or an electrostatic potential could be applied to it. The instruments that comprise the diagnostic package have been flown several times previously on sounding rocket flights.

4.0 ASSESSMENT FOR STS FLIGHT

This experiment is well qualified for flight on the Shuttle. It must be performed in the space environnent, and it requires the use of a fixed mounting platform and maneuverable system to carry out the mapping of the wake.

There are 3 areas of possible concern. The first is the arrangement for the erecting of the mast. The second is the method that will be used to map the wake. The third is the arrangement for the transmission of the data. These concerns will be discussed below under Experiment Considerations.

4.1 Experiment Considerations

4.1.1 Design Suggestions

The investigator mentions use of EVA to erect the mast. EVA is very time-consuming, using a minimum of 1/2 day, most of which is directly removed from available experiment time. The experiment should use one of the several motor driven, self-erecting, masts that are available and space qualified. This type of mast could easily extend the body for more than 10 meters. Assessment #16, "Optical Countermeasures Demonstration," describes such a mast.

The investigator plans to use the Remote Manipulator System (RMS) to hold the diagnostic package and perform mapping of the wake of the test body. The RMS does not have leads to the end effector, thus the diagnostic package will require a battery pack. Such a pack would require about 700 watt hours in order to carry out the operational requirements of the experiment. This would make a significant weight contribution to the package, but the RMS is capable of manipulating weights up to 29,000 kg so that the added weight would be of little significance in terms of accommodation.

It is suggested that the investigator study the possibility of fixed mounting the diagnostic package on a stronger version of his mast and afixing the test body to the RMS. In this configuration, wires could easily be attached to his mast for power and data transmission. One disadvantage of such a revised configuration would be that the investigator would not be able to explore the wake properties of the Orbiter itself, which was one of his secondary objectives.

Whether the balloon or the diagnostic package is fixed, the relative distance between them will have to be read from the RMS position sensors. At the present time, the system is not well enough defined to know the accuracy of its positional determinations with respect to the Orbiter coordinate system, however numbers on the order of 2° or 10 cm have been mentioned. This accuracy is not sufficient for the measurement of the wake of a 30 cm diameter body. Therefore, the experimenter should at least consider some active system for determining the relative position of the diagnostics and the test body, since the RMS position measurements may not be able to meet his needs.

There are no problems in accommodating the experiment data rate either through the Orbiter's detached payload S-band receiver, or through an RAU into the Spacelab CDMS if the diagnostic package can be fitted with data wires. There are no serious environmental constraints that cannot be met by the STS or by the instrumentation. All other aspects of the instrument design can be accommodated by the Shuttle Transportation System or by the Spacelab system, and there should be no problem in placing this experiment on some STS payload which is equipped with the RMS.

4.1.2 Operation Restrictions

At all times during the measurement period, the diagnostic package and the test body must be positioned so that they are not in the wake of the Orbiter. The purpose of the 10-meter mast is to remove the experiment to such a distance that when the Orbiter is properly oriented, the experiment will not be in the Orbiter wake. There should be some range of Orbiter orientations that are acceptable to the wake experiment in which the cargo bay can view the earth, the sky, or the horizon to accommodate other experiments. The best orientation of the Orbiter is, however, close to YPOP with X vertical or possibly ZPOP with X vertical. Both of these orientations should be within the ability of the Orbiter to maintain attitude. The ZPOP orientation would be preferred because the Orbiter drag would be less.

The experiment requirements for orbital altitudes in the range from 200 to 450 km and an orbital inclination of 57° are compatible with the STS.

The mapping of the wake is planned to be done by programming the RMS to execute the proper motions during the course of the experiment so that the full half wake can be mapped. On the other hand, the accuracy of the RMS position measurement may not be great enough so that a crew member would be required to periodically determine the distance between the diagnostic package and the center of the test body.

If the diagnostic probe is to be mounted on the RMS, there would be a necessity to provide a stowage platform for the package during the launch and landing periods of the flight and to provide a special mounting bracket that can be grasped by the RMS end effector.

The instruments in the diagnostic package are susceptible to contamination by dust. At least 10,000 class cleanliness as a matter of course to experiments so that this experiment poses no problem. The instruments for the most part can only make measurements in a vacuum so that they cannot be fully tested once they have been installed in the payload. This fact will make it desirable for the investigator to have access to the instruments as close to the launch date as possible.

4.1.3 Experiment Support Equipment

The experiment requires the use of the Remote Maneuvering System. It needs power for the diagnostic package and data handling capabilities for a serial bit stream of 12.4 Kbps. Depending on the experiment design, these data will be either transmitted from the diagnostic package to the detached payload receiver in the Orbiter, or could be wired into a RAU and put on the CDMS data bus. There is a need for minimal display of the instrument's status to the crew member so that he can check on the operation of the experiment. This status data can be displayed on the CRT in the aft flight deck or on the CDMS CRT.

There is a possibility that other payloads will be doing similar experiments (specifically the AMPS payload) and that the investigator would be able to make use of the mast that must be developed for that payload.

4.1.4 Experiment Cost Considerations

The following cost considerations should be made by the investigator: (1) the use of a motor to both erect and stow the mast; (2) placing the balloon instead of the diagnostic package on the mast, so that he would be able to use the Spacelab RAU connection instead of the telemetry link, (3) consider the use of a mast designed for another flight such as the AMPS, and (4) a spatial determination instrument.

4.2 STP Integration Considerations

The only problem with integrating this experiment involves the use of a ten-meter rigid mast that must be stowed along the cargo bay. Strong consideration should be given to use of an extendable mast. The diagnostic package, if it is carried by the RMS, will require special mounting brackets and a bracket that can be grasped by the RMS end effector. The diagnostic package will require a battery system which might have to be charged before it is deployed on the RMS. Power lines would be required to the point where the diagnostic package is stowed during launch and landing. Figure 4-1 shows the positioning of the instruments on the Orbiter in the investigator's preferred configuration with the balloon on the mast and the diagnostic package on the end of the RMS.

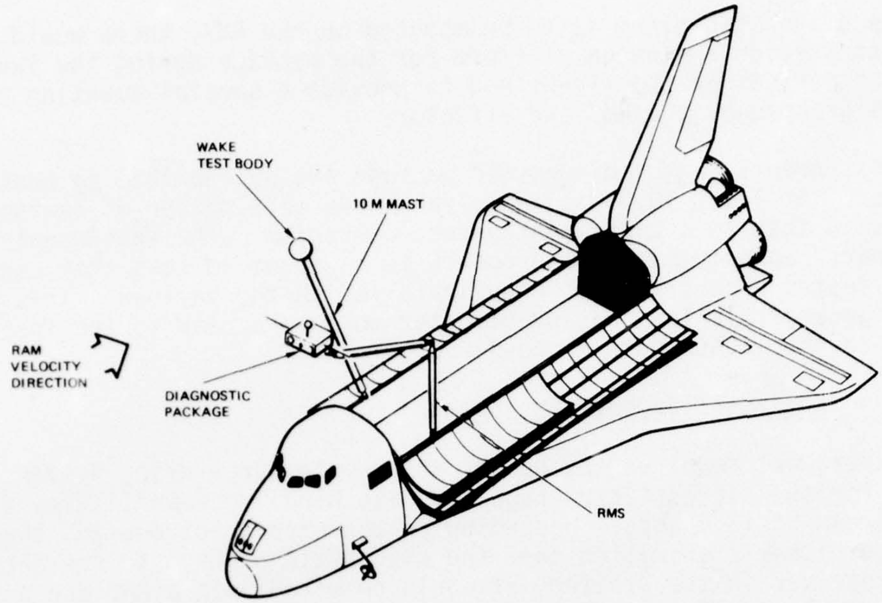

Figure 4-1. Conceptual Layout of the Sheath and Wake Studies Experiment on the Shuttle

This experiment can be performed with any other experiment that does not produce a significant change in the ambient plasma environment of the orbiter during the period that the wake and sheath are being measured. The instruments in the diagnostic package have been flown repeatedly on rocket flights and on spacecraft and there should be no great problems encountered in integrating them onto the STS once the most cost effective mechanical design of the experiment has been determined. As was mentioned before, a similar experiment was proposed for the AMPS flights of Spacelab, and designs of the experimental configuration were fully compatible with the STS/Spacelab system.

5.0 RECOMMENDATIONS AND REMARKS

(1) This experiment is compatible with the STS and could fly on an STS sortie mission.

(2) The investigator should consider using the RMS to hold the wake test body and mounting the diagnostic package on his own mast with Spacelab power and RAU cabling.

(3) The accuracy of the RMS position determination may not be great enough for the experiment and some scheme for better position determination will have to be devised.

(4) During the performance of this experiment, the instruments must point into the orbital ram direction and they must be outside of the Orbiter wake. This requires that some special Orbiter attitude be maintained for periods of 90 minutes twice a day throughout the flight.

(5) All masts and packages that extend beyond the cargo bay must be able to be jettisoned in case they fail to stow. The investigator is aware of this requirement and plans to incorporate the needed mechanisms.

NEUTRAL ATMOSPHERE/PLASMA INTERACTION AT LOW LATITUDE

1.0 EXPERIMENT IDENTIFICATION

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2.0 REQUIREMENT BACKGROUND

Response to STS Utilization Presentation (Response Sheet No. 41).

This experiment is part of the Task: Electrical Structure of Aerospace, Task No. 2311G2 at the Air Force Geophysical Laboratory.

3.0 EXPERIMENT APPROACH

The intention of the investigator is to measure the plasma and neutral properties of the ionosphere in the altitude range from about 200 to 900 km for the range of latitudes from about 45° N to 45° S latitude. He wishes to characterize the interaction of the ionosphere with the neutral atmosphere in this latitude region where a large number of interactive phenomena are known to occur. The instrumentation consists of a set of five interrelated devices which measure the particle and wave properties of the plasma and the neutral mass composition of the atmosphere. If possible, the investigator wishes to carry out simultaneous correlative measurements with ground based incoherent backscatter radars. The entire experiment is invisioned as occupying a free flying satellite which must be well separated from the EMI and particulate contamination of the Shuttle orbiter.

The following instrumentation is to be mounted on the free flyer: (1) A neutral mass spectrometer sensitive to the mass range from 1 to 64 AMU; (2) an array of five ion sensors, (3) a three-axis electric field measuring device sensitive from 0 to 30 Hz, (4) a thermal electron probe and (5) a spacecraft-provided magnetometer.

Several of these instruments must be mounted so that they are facing into the orbital ram velocity direction while simultaneously the electric field meter must be spinning at a few revolutions per minute. This requires that the free flying spacecraft have some means of de-spinning some of the instruments.

The electric field meter consists of an electronics box and six dipole antennas that mounted in the three orthogonal axes of the spacecraft. The antennas have two lengths, four of them are 60 feet long and two of them are 30 feet long. The two 30-foot antennas are intended to be mounted along the spin axis of the spacecraft and would be semi-rigid, while the 60 foot antennas would be mounted perpendicular to the spin axis and would be of the STEM type and would be kept straight by centrifugal forces. The neutral mass spectrometer and two of the 5 ion detector units would be mounted on a de-spun platform which would always be pointed into the orbital ram velocity direction. The neutral mass spectrometer weighs 20 pounds and measures 8 inches by 20 inches on the base and is 8 inches high. The two ion devices would weigh 12 to 15 pounds and would be 10 by 15 inches on the base and be 6 inches high. The electron probe active element would be mounted on a rigid 4-foot long boom. The remainder of the detectors would be mounted on the body of the spacecraft and their apertures would have unrestricted viewing of the ambient plasma. The total weight of the instruments including the antennas would be about 90 pounds. The size of the electronics boxes mounted on the spacecraft body would total about 20 by 30 by 10 inches (the exact size remains TBD, but is unlikely to be significantly larger than given).

The pointing requirements of the instruments are that the spin axis of the spacecraft be from 85° to 95° to the orbit plane and that the despun platform holding the neutral mass spectrometer and the two ion devices should be able to be pointed along any one of three inertial axes with an accuracy of 5°. Relative to the spacecraft coordinate system, the angles should be known to within 0.1°. The angles should be known to 0.4° with respect to some fixed inertial coordinate system.

The electrical power requirements of the instrumentation are 42 watts average with a 33% operational duty cycle so that the total energy consumed during one orbit would be about 70 watt hours. The peak power during the deployment of the various antennas and the boom will be 56 watts and would last no more than 20 minutes. There might be occasion later in the flight when the length on some of the antennas might be changed, however, the contribution to the total power use would be very small. The instruments would take 28 volt power.

The experiment needs a full telemetry system capable of handling up to 40 channels of data with an overall data rate of about 20 Kbps. There is a need for a data storage capability to maintain a continuous record of experiment performance when the spacecraft is unable to transmit. There will be a need for about 40 bilevel commands which would be issued one at a time from the ground. As each command is issued, its effect would be observed and then another command would be issued. During the period when there are simultaneous ground radar observations, there would be a possible need to go through a command sequence. The uplink command rate could be rather slow at one bilevel command every 5 to 30 seconds.

The instruments require a spacecraft that will be able to maintain them within the temperature range from -10° to 50°C during their operational period and -30° to 80°C during non-operation. The instruments are sensitive to dust and outgassing products and the spacecraft should produce a minimal interaction with the ambient plasma. There is a requirement that the surface of the spacecraft be a conductor at least near the entrance apertures of the various ion detectors. The conducting area of the spacecraft should also be large enough so that the spacecraft does not become charged during the periods of observation. There are TBD requirements on the RFI of the spacecraft because of the electric field meters which would be very sensitive to electrostatic and electromagnetic radiation by the spacecraft subsystems. There would be a milliguass limit on the magnetic fields produced by the spacecraft.

The neutral mass spectrometer will be vacuum sealed until after the spacecraft has been launched, and there are other requirements for cleaning of surfaces as near to the launch time as possible.

These instruments are standard spacecraft instruments that have been flown repeatedly. The investigator has no need to recover the payload, however, he would like it recovered if possible for refurbishment and modification for another flight.

4.0 ASSESSMENT FOR STS FLIGHT

The spacecraft that carries these instruments should be well qualified to be launched from the STS. However, at the present time there does not appear to be a suitable spacecraft for carrying the instruments within the STS provided hardware. The multimission modular spacecraft does not appear to be suitable for this experiment.

There are three areas of concern where this experiment does not seem compatible with the MMS system: (1) the required orbital altitudes lie beyond the design capabilities of the MMS; (2) the MMS does not as a matter of course provide a despun platform, (3) the weight and power of the experiment seriously underutilize the capabilities of the MMS system. These concerns will be discussed below.

4.1 Experiment Considerations

4.1.1 Design Suggestions

The design of the instruments and their requirements are straightforward. There should be no problem in accommodating the spacecraft that contains them on the STS orbiter or in launching that spacecraft into the desired orbit with the spinning solid upper stage or even some propulsion stage of much lower capability since the entire spacecraft with instruments and subsystems would probably weigh less than 300 pounds. The boost required to obtain the required elliptical orbit of 200 km by 900 km at 45° inclination would not be very large.

The major problem in accommodating this set of instruments on any spacecraft is the necessity for the despun platform and the number of long antennas that are deployed. The required spin rate of 2 to 3 revolutions per minute is needed primarily to keep the antennas extended and to assure that there are no instruments that find themselves making measurements only in the wake of the spacecraft. It is suggested that the investigator consider strengthening the electric field meter antennas so that they would remain rigid without needing the spacecraft spin. In this case, the spacecraft would then be able to spin at the rate of once per orbit and therefore be able to keep a set of the instruments always pointed into the orbital ram velocity. At some increased cost in data reduction effort, it is possible to reduce the number of antennas required for a 3-axis measurement of the field to four, however, the complications of the data analysis probably outweigh the simplification in the spacecraft support requirements. The four-antenna configuration requires that the antennas be rigid for even a slowly spinning spacecraft.

4.1.2 Operation Restrictions

There are no foreseeable restrictions on the spacecraft once it has been launched so long as it is equipped with a tape recorder to store data when transmission to the ground is not available. The coordination of the spacecraft activities with the ground based radar observations would have to be planned for a time when the spacecraft was in real time contact with the ground so that command functions could be used.

During the time that the spacecraft is being carried by the Shuttle orbiter, there are some restrictions on its operations. The electric field meter antennas cannot, of course, be deployed, and the mass spectrometer would remain covered. Otherwise, the status of the various instruments can be checked out. During this checkout period, it would be desirable to be able to transfer the spacecraft telemetry bit stream to the ground so that the investigator can check on the status of the spacecraft and instruments. Given the telemetry rate of 20 Kbps, there should be no problem in obtaining the data in real time on the ground. It is assumed that by the time this spacecraft is launched, upper stage propulsion systems and the methods of launching them will be standardized for the STS, and that the launch would be a routine procedure.

4.1.3 Experiment Support Equipment

The experiment would need STS communication links during the checkout before being launched from the orbiter. It would need some upper stage propulsion system to place it in the proper orbit. It would need STDN/TDRSS telemetry coverage during its periods of operation.

Finally, consider the use of the MMS system as the spacecraft to carry this experiment. The MMS has thermal design specifications for the altitude range from 500 km to 1600 km and for geosynchronous orbits. This range does not include the lower altitudes required by the experiment, 200 to 300 km. It is not clear if this altitude would be totally inaccessible to the MMS, but the spacecraft is not designed to operate at such low altitudes. Drag and prolonged solar eclipse would be two of the possible problems encountered at lower altitude.

The MMS has no provision for supplying a despun mounting platform to any of the instrumentation packages as part of the spacecraft system. If a platform is required, it must be supplied by this experimenter. The spin rates required by the long antennas pose no problem to the MMS, nor do the attitude and pointing accuracy requirements. In fact, the MMS abilities far exceed the needs of this experiment.

If MMS is to be used, other complementary experiments should be added to form a payload exploiting the full capabilities of the MMS. This approach would provide this experimenter with an excellent carrier while affording other opportunities to orbit additional experiments on a free flyer.

If the MMS is to be used, the investigator would be responsible for designing all of the instrument mechanical support equipment which would then be mounted on the MMS Transition Adaptor. He would be responsible for designing all antenna and boom deployment mechanisms and would be responsible for all electrical and data interconnections within the instrumentation section of the payload. He would be responsible for the despun platform. There may be some difficulty deploying one of the dipole antennas through the MMS Module Support Structure along the spin axis of the spacecraft.

4.1.4 Experiment Cost Consideration

Given the above argument concerning the use of the MMS, it appears that if the investigator wishes to use the standardized STS/Spacelab equipment, there will be a need for a considerable design and construction effort involved in flying his experiment. This would lead to rather high costs, probably considerably higher than the cost of the instrument themselves.

4.2 STP Integration Considerations

The integration of the spacecraft carrying this experiment onto the STS would be easy under the assumption that the launching of free flying payloads with the use of upper stage propulsion systems is one of the standard operating modes of the STS. There is no definite requirement to recover the payload so that aspects of the integration would be simplified. The experiment itself would cause no integration complications, since it operates only in the free flying mode.

On the other hand, the integration of the instruments onto a suitable spacecraft would be considerably more complicated. It appears to us that this experiment would require a specially designed spacecraft or considerable redesign of existing spacecraft. Therefore, the integration effort would be comparable to that required for the Atmospheric Explorer series of NASA spacecraft.

Figure 4-1 shows a very conceptual layout of the instrumentation on a spacecraft similar to the Atmospheric Explorer spacecraft. Figure 4-2 shows the same instruments mounted on the MMS.

5.0 RECOMMENDATIONS AND REMARKS

(1) This experiment requires a free flying spacecraft upon which to mount its instruments. Once such a spacecraft exists, there would be no problems foreseen in using the STS to launch the spacecraft into the required orbit with an upper stage propulsion system.

(2) The Multimission Modular Spacecraft associated with the STS does not appear to be the ideal vehicle to use for the free flyer.

The MSS does not have the required despun system. Its power and weight capabilities far exceed the needs of this experiment, and its lowest design altitude, 500 km, is higher than the required perigee, 200-300 km.

(3) The investigator should study the cost tradeoff between flying rigid electric field meter booms and having to provide a despun platform for the neutral mass spectrometer.

(4) The investigator should determine if some spacecraft system, other than the MMS, would meet his requirements. The requirement for constant pointing into the ram direction would rule out any spacecraft which depended solely upon spin stabilization. The NASA Atmospheric Explorer series of spacecraft provides some of the needed facilities, however, it does not have a despun platform. The spacecraft with despun platforms tends to be designed for much larger instrument compliments than those of this experiment and consequently, would be expensive to acquire.

(5) The major integration effort involved with experiment would be the integration onto the free flying spacecraft. Integration of that spacecraft onto the STS would not be complicated.

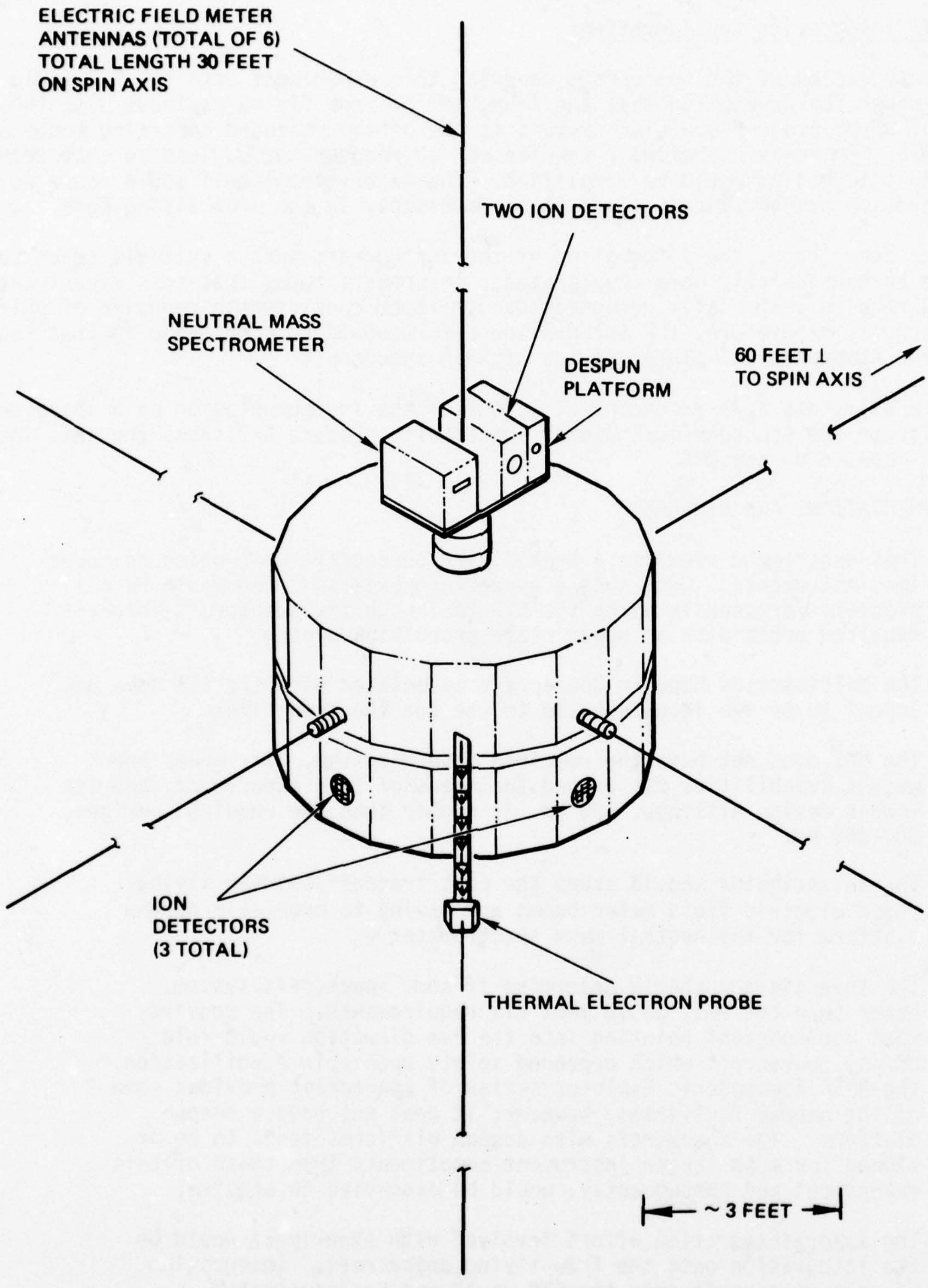

Figure 4-1. Conceptual Layout of a Small Free Flying Spacecraft Fitted with the Neutral Atmosphere/Plasma Interaction Experiment

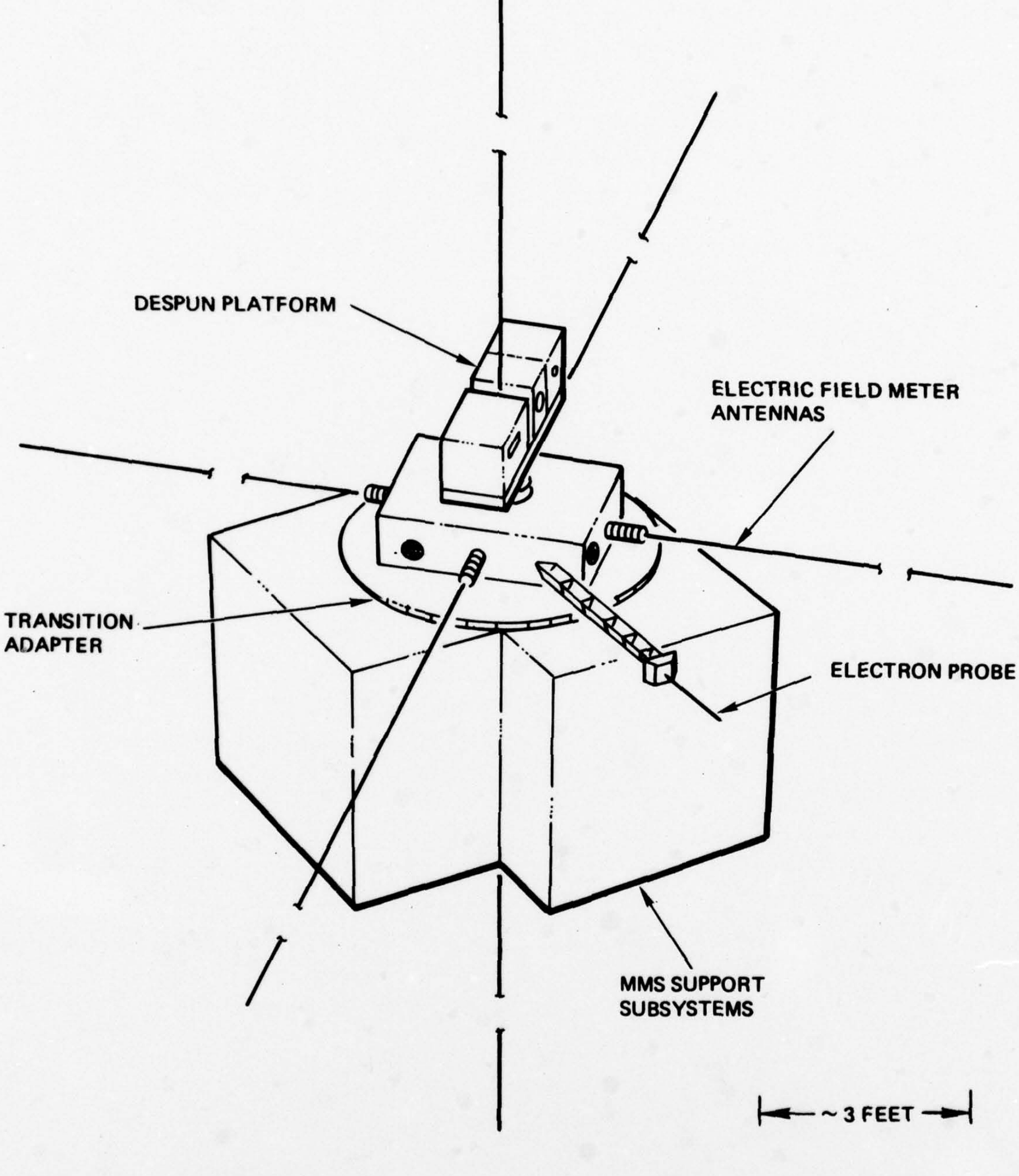

Figure 4-2. Conceptual Layout of the Neutral Atmosphere/Plasma Interaction Experiment on the Multimission Modular Spacecraft

HORIZON ULTRAVIOLET EXPERIMENT

1.0 Source

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2.0 REQUIREMENTS BACKGROUND

Response to STS Utilization Presentation (No. 44)

Identifying Numbers: FY 77, Work Unit 66880701
FY 78, Work Unit 66901702, UV Horizon Measurements

Form 1721 is in preparation

3.0 EXPERIMENT APPROACH

3.1 Objectives

This experiment is to provide detailed quantitative data on brightness of the earth's atmosphere, and in particular, that of the earth's limb, at ultraviolet wavelengths ranging from 500 to 4000°A. This information is needed to aid in the development of UV horizon sensors and of sensors applicable to missile surveillance and tracking. Lack of sufficient UV atmospheric and earth limb profile data is hampering progress toward development of such sensors at present.

Data from the proposed experiment will permit evaluation of the potential of UV horizon sensors in comparison with existing IR horizon sensors. In addition, sensors with improved characteristics, including greater accuracy, reduced complexity and cost, and lower susceptibility to geophysical variations, cloud interference, etc., are needed for systems engaged in missile surveillance and tracking, communications, navigation, and in most earth-oriented observations.

The proposed limb observation experiment will provide the needed data on UV radiance along slant paths. This is the background against which missile exhaust plumes are to be detected.

3.2 Background

Several rocket and satellite flight programs are currently being planned by the Air Force Geophysics Laboratory to perform related UV atmospheric measurements but on a less comprehensive scale. In these measurements, the sensors will make observations primarily in the nadir direction. The flight programs include the following:

1) VUV Backgrounds, CRL-246, STP Mission S77-2. This experiment is being integrated into a pallet payload at the present time under the guidance of the Space Test Program Office at SAMSO. Spectral and spatial data will be obtained in the nadir direction. Some limb scans may be possible at the conclusion of the flight, depending on resources, but the detailed, global coverage needed will not be obtained. Data are expected during CY 1978.

2) Multispectral Measurements Program (MSMP). This program will obtain missile exhaust plume intensities in a wide wavelength region from the infrared through the ultraviolet. It is associated with SAMSO (SZ). The UV sensors will provide spatial and spectral target data that will be used with background data in order to develop the most suitable applications for ultraviolet missile detection. The program involves a series of Aries rocket launches carrying both a separable target engine and a sensor module. Flights are planned over the next several years with the initial launch during 1977.

This proposed Shuttle-based experiment series will implement the earlier programs by systematically mapping the brightness of the near-earth atmosphere as a function of pointing direction, or altitude, and ultimately provide global coverage. Although the emphasis is on limb profile measurements, a sufficient number of scans from nadir to horizon will be conducted for correlation with results from the earlier experiment series.

3.3 Experiment Equipment and Procedure

3.3.1 Equipment

The instrumentation is composed of six 3x4x12 inch Faste-Ebert UV spectrometers that are independently set to a wavelength band of interest. Together, they cover the wave length range from 500 to 4000°A. Motor driven mirrors are used to scan the incident ultraviolet light across variable-geometry diffraction gratings. The instruments' clear field of view is 0.1 to 0.5 degrees. The external configuration of the spectrometer is illustrated in Figure 3.1. A gimballed mounting platform capable of pointing the spectrometers at various points of the horizon and of scanning the limb is to be provided as flight support equipment. This gimballed platform also isolates the precision pointing spectrometer package from Orbiter altitude changes.

The required pointing accuracy is between 0.1 and 1 degree and the required pointing stability 0.1 degrees per 5 millisec (the exposure duration per measurement). These accuracy requirements are preliminary and can possibly be alleviated. Knowledge of the pointing direction is more important than exact control.

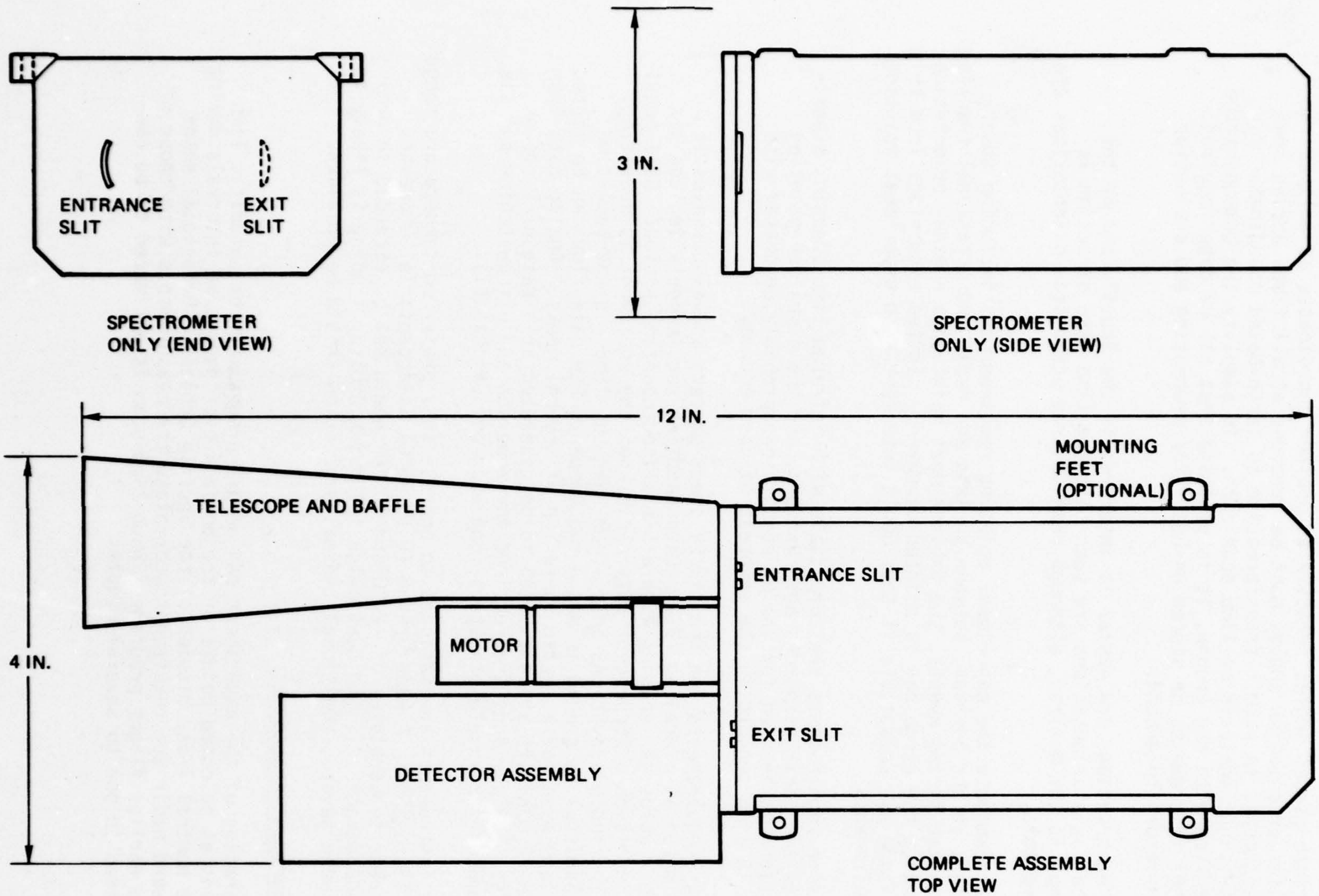

Figure 3-1. Configuration of Horizon UV Spectrometer (Preliminary Sketch)

Platform gimbal angles relative to Orbiter coordinates as well as the Orbiter orientation angles must be recorded at all times during the measurements to permit transformation to earth-based coordinates for post-flight analysis of limb scan data. To simplify the communication interface with the ground, it is proposed that all UV data acquired by the instruments be stored on-board for processing and evaluation upon return to ground.

The observations are easier to interpret if the Shuttle and/or the gimbal axis orientations are such that the limb scan direction is normal to the horizon, although results from other scan orientations are acceptable.

To accommodate the experiment pointing requirements including observations at the horizon, between horizon and nadir and occasional tracking of rocket firing events, the Orbiter must maintain a nominal orientation in which the cargo bay is pointed downward. Limited excursions from this attitude are acceptable if they don't interfere with experiment pointing requirements.

UV scan requirements are compatible with infrared limb scanning experiments and earth resources observations, and the gimballed pointing platform to be used for the UV sensors can probably be shared with the IR limb sensors in the interest of cost economy.

Power requirements for the six UV spectrometers and electronics is estimated as averaging 12 W. Data handling requirements include six 16 bit words per channel with a 5-millisec counting period. Data acquisition is on a 25 percent duty cycle when the equipment is operating, reflecting observations only at or near the limb. Since real-time transmittal to ground is not a requirement, the data flow can be stored on tape even for a sortie operation of several weeks. Analog data from 8 to 12 monitors are estimated to be generated at a rate of 1 cps or less. Six commands are required for power on and off switching and six commands for wavelength steps (one each per UV channel).

The instruments are designed to operate in a preferred temperature range of 15 to 25°C. A wider range (0 to 35°C) is acceptable. However, extreme temperatures of -20°C and 100°C should not be exceeded to preclude damage. If the heat pulse following Orbiter landing is likely to be more severe, additional thermal protection should be provided.

3.3.2 Experiment Procedure

Operation of the experiment can be pre-programmed for automatic limb scans at selected points of the horizon over some time intervals during the nominal 7-day mission of the Shuttle Orbiter. Occasional scans toward nadir are required to correlate the measurements with those of the earlier flight programs. Each limb scan is estimated to be completed in one to several minutes.

In addition to horizon scanning, the mission plan will define opportunities for viewing rocket firings at launch sites such as ETR, WTR and Wallops Island, if the orbiter pass is within observation distance. The timing of the mission and the rocket launch schedule require careful advance consideration as well as confirmation and program adjustment while the mission is in progress. Although trajectory data of the target rocket and relative position data between the Orbiter and the target will be provided by mission control to the Orbiter on a real-time basis to control instrument pointing, it is anticipated that visual tracking and manual pointing control override may be necessary by one of the crew members to assure successful observation of the event.

Other than this specific task, participation of the crew in the conduct of the UV experiments is minimal. These crew activities are restricted to:

- Initially deploying the pointing platform from the stowed condition (see below) when the Orbiter is ready for orbital operations.
- Readying the experiment for measurement initiation which is commanded from the ground.
- Monitoring the status of the experiments.
- Effecting secure retraction of the platform prior to closing the cargo bay in preparation for reentry.

3.4 Shuttle Orbits

The program requires acquisition of UV atmospheric data at all latitudes. Initial flights launched from ETR will permit coverage of low and intermediate latitudes only. Shuttle flights launched off WTR will permit measurements in polar orbit at a later time. This will extend geographical coverage to higher latitudes and permit observation of auroral UV phenomena, considered important to this program. Ultimately, complete global coverage of UV atmospheric phenomena is desired.

Since orbital altitudes are not critical to the experiment, (altitudes from 100 to 400 n.m. are acceptable), there will be many flight opportunities. However, with increasing altitude the slant range to the horizon increases rapidly, and consequently, resolution and accuracy of the UV limb measurements decrease. On the other hand, higher orbital altitudes will provide more frequent opportunities for rocket firing observation (see below).

3.5 Program Evolution

Work toward UV horizon sensors will involve a series of missions. Initially, it is necessary to acquire the needed limb profiles to evaluate the suitability of the UV limb for this purpose. Global coverage is required which will require a number of flights. In addition, various developmental ideas will be evaluated in space.

Operational, or near-operational, sensors may use detectors that are different from those used to gather the limb data. Alternate techniques for using the UV limb and alternate sensor designs, multicolor systems and on-board processing approaches require in-flight testing in subsequent phases. AFGL therefore foresees a continuing evolutionary UV experiment program in the development of operational sensors for use on spacecraft.

4.0 ASSESSMENT FOR STS FLIGHT

The horizon UV experiment will provide data that are essential to the development of UV detectors for horizon sensing and for missile surveillance and tracking. Such sensors will be used to complement the capabilities of existing IR sensors. To cover the spectral range from near-UV through VUV and XUV, the measurements must be conducted from above the earth atmosphere.

Utilization of the Shuttle Orbiter for this experiment is primarily a matter of cost effectiveness in view of the following considerations:

a) Repeated flights are required to obtain the necessary atmospheric UV data base and to support sensor technology evolution.

b) Measuring equipment and flight support equipment can be reused in successive flights.

c) Some of this equipment can be shared with similar IR experiments (e.g., the pointing platform) being carried in the same mission.

d) The Shuttle Orbiter provides most of the engineering support and housekeeping functions required by the experiment.

e) The experiment has modest weight, volume and power requirements (except for the pointing platform) and can be accommodated on Shuttle flights that are shared by several other users.

The experiment is largely self-contained and can be conducted automatically in a pre-programmed sequence. Atmospheric and target observation data, acquired by the experiment, can be recorded and stored for post-flight analysis, along with data on relevant Shuttle operating conditions, e.g., orientation angles and orbit positions.

Experiment support onboard the Orbiter requires a precision pointing platform with two (or preferably three) gimbal drives to provide sufficient line-of-sight pointing accuracy and to decouple the sensors from Orbiter rotations. Otherwise, the electrical and mechanical interfaces with the Orbiter system are of modest complexity.

Crew tasks and ground communication requirements are minimal except during observation of rocket firings. The objective of rocket plume observations requires careful coordination with launch site activities prior to and during the mission and will constrain Orbiter mission timing and mission profile selection.

Orbit characteristics required for the experiment are compatible with many other Shuttle sortie missions. This facilitates experiment accommodation. Also, the nominal Orbiter flight attitude with the cargo bay pointing downward is compatible with other Shuttle earth observation and atmospheric research objectives, especially since limited pitch and roll excursions from the nominal orientation do not interfere with the experiment and are acceptable.

4.1 Experiment Considerations

4.1.1 Scan Patterns

Limb scan patterns that may be used in the experiment include:

- A squarewave pattern with measurements taken during the upward and/or downward strokes.
- Sinusoidal or triangular wave patterns.
- A sawtooth pattern scanning in nearly vertical direction downward.

The sawtooth pattern seems best suited for purposes of this experiment since it scans nearly normal to the horizon and always in the same direction.

Azimuth sweeps may be conducted around the entire horizon or within some limited azimuth angle. The circular azimuth sweep tends to produce overlapping coverage in successive orbital passes. For example, with 200 n.m. orbital altitude and 50 n.m. horizon altitude, the horizon radius is 1035 n.m. The distance between adjacent ground tracks at 30 degree orbit inclination is only 675 n.m. The overlap beyond adjacent ground tracks, therefore, is 360 n.m. A limited azimuth sweep on one side of the Orbiter, e.g., between 30 and 150 degrees from the velocity vector, avoids this overlap. It also precludes field-of-view obstruction and reflected light interference by the Orbiter's front and tail structures.

4.1.2 Day and Night Observations

Both day and night observations of the atmosphere are desirable. Fluorescence and sun light scattering effects are observable only in daylight, but sun interference at angles up to 90 degrees from the instrument center line must be avoided. This implies some azimuth restrictions during daylight observation and near the terminator. Eclipse durations depend on orbital altitude, inclination, equator crossing times and season. For low inclination orbits, the eclipse duration is typically one-third of the orbit period. Thus, the available observation times in sunlight and eclipse tend to match observational priorities indicated by the experimenter.

4.1.3 Rocket Plume Observation

Careful advance and in-flight coordination with rocket launch schedules is required in order to make rocket plume observation from the Orbiter possible. A first such observation was conducted successfully during the SKYLAB program during a passage of WTR although the observatory's 51-degree orbital inclination was not optimal for this purpose. Crew participation in target acquisition and tracking as well as instrument pointing proved essential in conducting that experiment.

"Observation windows" occur when one of several Orbiter passes of the launch site fall within the daily launch window of a missile or satellite launch event. With a four-hour launch window and several successive orbiter passages near the launch site at about 90-minute intervals, as many as three observation windows may occur under favorable conditions, as illustrated in Figure 4.1-1.

Figure 4.1-2 shows successive ground tracks of a 30-degree inclined orbit in the vicinity of ETR. The tight ground track pattern that forms near the maximum latitude permits five or more successive target observation opportunities. Three concentric circles indicate launch site distance of the Orbiter passes, with the largest circle of 1200 n.m. radius representing a typical horizon distance.

Target observability actually depends on many factors including the UV instrument detection range and rocket plume intensity, the amount of background interference, and on relative Orbiter, target and sun positions. Because of the geometrically sensitive nature of the encounter, detailed analysis of target observability is necessary in each case. However, it is apparent that orbit inclinations between 30 and 35 degrees are more favorable than higher or lower ones because of latitude compatibility with different U.S. launch sites (ETR, WTR and Wallops Island).

4.2 STS Integration Considerations

The UV sensors are sufficiently well developed and compatible with the Orbiter so that integration should cause no major problems. Since the experiment is a continuation of rocket and satellite flight programs, there should be no need for extensive testing or simulation. Assuming that the sensors be mounted on a pointing system such SIPS (see below), which will be available as part of the Spacelab system, there should be no problem integrating the instruments with the flight support system. The experiment can therefore be accommodated early in the STS program on a "space-available" basis.

4.2.1 Configuration Concept

Use of the Small Instrument Pointing System (SIPS), is suggested as a support platform for the package of six UV spectrometers used in this experiment (See Figure 4.2. The SIPS, being developed under NASA/GSFC direction for the Spacelab program, consists of a deployment/retraction pedestal and a pointing section which includes an azimuth rotation drive and a pair of instrument canisters supported and gimballed separately in elevation. Each canister can be rotated independently inside its elevation yoke over a small range of angles. An optional roll gimbal assembly can be added to support the instruments inside the instrument canister. The angular freedom of these gimbal drives is as follows:

Azimuth	$\pm$200°
Elevation	$\pm$120°
Right Left (in the elevation yoke)	$\pm$10°
Roll (about instrument line of sight)	$\pm$125°

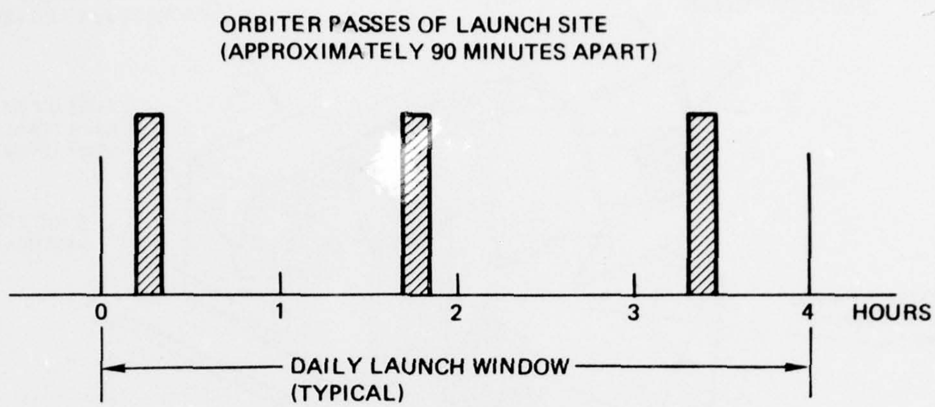

Figure 4.1-1. Observation Windows of Rocket Firing

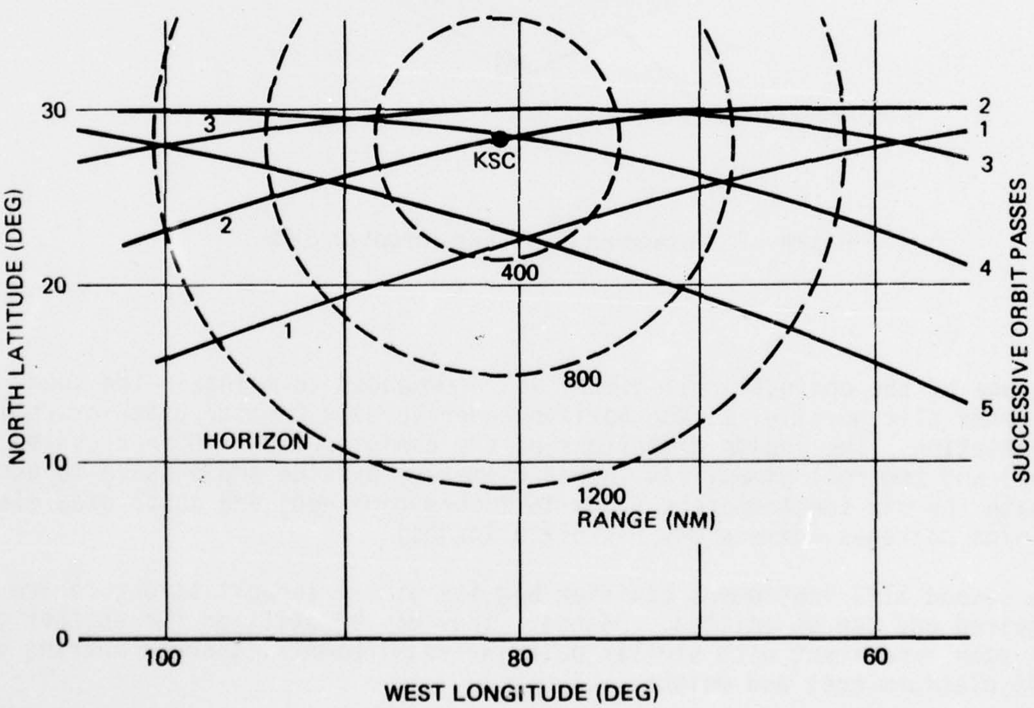

Figure 4.1-2. Successive Orbiter Passes in Vicinity of ETR

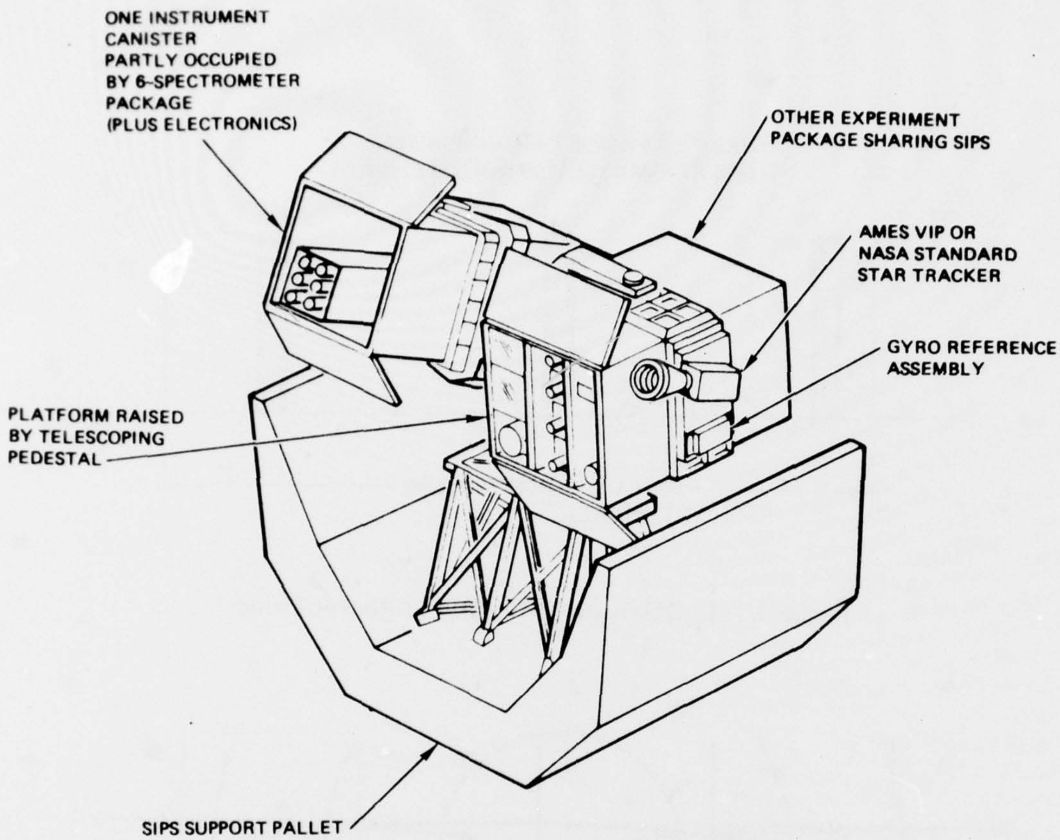

Figure 4.2. Installation of UV Experiment on SIPS

The use of the optional roll gimbal is recommended to maintain the spectrometer slit parallel to the horizon under varying Orbiter pitch or roll orientation. The inside dimensions of the canister (40 x 40 in cross-section) and the roll gimbal (34 inches diameter) provide ample space to accommodate the six spectrometers (16 x 16 inches combined) and associated electronics packages (dimensions 5 x 5 x 6 inches).

The second SIPS instrument canister and its gimbal support structure are not required and can be omitted. However, they may be utilized for another optical scan experiment with similar pointing requirements, thereby sharing the SIPS platform cost and weight.

The platform, designed for astronomical instrument pointing, provides pointing accuracies and stability exceeding those specified for the UV experiment. Pointing sensors suitable for the accuracy requirements of the experiment can be selected from a "stockpile" of standard units. Slewing rates (2 degrees per sec maximum) are sufficient for rapid horizon scanning, target acquisition and tracking. In addition, the instrument canister provides complete environmental protection including:

- thermal control
- contamination protection (purge capability and a high level of cleanliness)
- acoustic protection

The protective covers are opened by spring action. Addition of a motor drive will permit opening and closing the covers as desired during the mission for added instrument protection.

The deploy/retract pedestal raises the platform to a maximum height of 4.3ft. above the stowed position on the support pallet. This serves to improve the instrument field of view over the cargo bay sides and structures fore and aft of the platform. Pyrotechnically actuated emergency separation and jettison provisions are included in the SIPS design to assure Orbiter safety in the event the system should fail to retract on command at the end of the mission.

4.2.2 Operation Restrictions

During the flight the UV sensors should not be exposed and operated until the pressure and dust contamination of the orbiter cargo bay have subsided to their nominal flight values. A crew member will have the task of checking the status of the experiment and to initiate the exposure sequences. The crew will also be responsible for the initial platform deployment and the final retraction and stowage sequence.

Possible interference with UV atmospheric observations by the exhaust from the Orbiter's RCS thrusters must be avoided. Contamination of optical surfaces by rocket exhaust particles is probably of no concern during firing of the small (25 lb) vernier thrusters but could be more significant during operation of the 900 lb primary thrusters. During these events, it may be necessary to close the protective covers on the SIPS instrument canisters. Instrument protection during any major orbital maneuvers in which the large 6000 lb OMS engines are fired, is a primary concern. However, such maneuvers probably would be performed with the cargo bay doors closed and thus would interrupt any other orbital experiment as well.

4.2.3 Preflight Preparations

Principal preflight preparations include:

- Evacuation and sealing of instrument.
- Optical alignment of the sensor package.
- Checkout.
- UV sensitivity checks and calibration.
- Recalibration between flights and recleaning, if necessary.

The design of the SIPS platform and instrument support canisters facilitates late access during ground integration and delivery of a fully aligned, checked out and sealed instrument package.

4.2.4 Cost Considerations

Low cost of STS services for this experiment can be realized because of its small instrument weight (estimated as 40 lb. including electronics) and size, because of its compatibility with mission profiles and orbit characteristics common to other earth observation missions, and because of modest demands made on crew activities. Special mission timing and coordination requirements with rocket launch schedules do not necessarily increase the STS service cost but primarily restrict the number of flight opportunities that may be utilized.

The cost of using the SIPS can be greatly reduced by sharing this platform with other experiments, perhaps even the same instrument canister since the spectrometer package occupies only one-third to one-fourth of the canister viewing area. Since the total required observation time is probably less than one day, time-shared SIPS operations during a seven-day mission will be acceptable.

5.0 RECOMMENDATIONS AND REMARKS

The experiment is compatible with the STS and can take advantage of the frequent flight opportunities offered for earth and atmospheric observation payloads. An available pointing platform such as SIPS can accommodate the UV instruments readily, having the required pointing accuracy and stability as well as environmental protection provisions. Sharing of the SIPS with other experiments is feasible and will considerably reduce cost.

An area requiring more detailed analysis is the requirement for, and feasibility of, coordination with rocket launch schedules, the availability of "observation windows," and the degree of crew involvement in accomplishing rocket plume observation.

ATMOSPHERIC TOPSIDE LASER SOUNDER

1.0 EXPERIMENT IDENTIFICATION

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2.0 REQUIREMENT BACKGROUND

Response to STS Utilization Presentation (Response Sheet No. 45)

3.0 EXPERIMENT APPROACH

This experiment will be performed to measure the density of the atmosphere in the altitude range from 8 to 30 kilometers above the surface of the earth. A laser beam is directed at the atmosphere from above and scattered by atmospheric molecules and dust particles. By using two different laser frequencies, the contribution of the molecules and the dust particles can be separated and a determination of the in situ density of the atmosphere can be made as a function of height above the earth. Because the fraction of photons scattered back toward the source is small, a light gathering system is required. The altitude distribution of returning photon flux is calculated by measuring the time of flight of the photons as they travel from the laser to the scattering point and back to the receiver. For this experiment, a ruby laser operating at its fundamental and at the first harmonic would be used for the two different wavelength sources.

The laser unit would be colinear with the telescopic receiver. The unit would be 80x80x100 cm in size and would weigh about 160 kg. The processing of the return signals would be done in a special purpose electronics package. This package would be about 30x30x30 cm in size and weigh about 23 kg. The laser and electronics would be mounted in the cargo bay in such a fashion that the laser would be able to fire vertically downward into the atmosphere.

The normal operational mode of the laser would be to carry out a sequence of firings into the atmosphere during the night portion of the orbit. The laser would fire every 10 seconds for periods of time on the order of ten minutes. The lower the orbital altitude the better, but the experiment would be designed to operate at a 200 km orbital altitude. Any orbital inclination is acceptable, but eventually 90° inclinations would be preferred.

There is only a minimal requirement for crew operations. At the beginning of an experimental period, the experiment would be turned on and the status monitored. The laser and the electronics would operate automatically during the performance of the experiment.

During the period when the experiment is in operation, the laser and receiver must be kept pointing at the nadir with an accuracy on the order of 1/2°. A stability of 1/2° is required for a time period on the order of one second. Slightly poorer pointing accuracy may be acceptable.

The power required by the laser system depends upon the rate at which the laser is fired. For a pulse every 10 seconds, the average power required would be 175 watts for both the laser and the electronics. A standby power of 5 watts would be required during periods of non-operation. The power profile of the laser itself is, of course, a set of spikes every 10 seconds. However, the power required from the STS would be a smooth function with only a slight ripple averaging 175 watts. Of this, the electronics unit requires 130 watts and the laser 45 watts. If the repetition rate of the firings were increased to once per second, the laser would require a power of 450 watts. The electronics package power would still be 130 watts. All power can be delivered at 28 volts.

The data rate out of the electronics package would be about 300 bits per second at the 10 second firing interval, and 3 Kbps at a 1 second firing interval. These data could either be recorded or transmitted to the ground in real time. There is an occasional need for the investigators to examine the data during the flight. Commands to the instrument would come either from the flight crew or from the ground at the beginning of an expermental period. The investigators desire to issue about 15 bilevel commands to the experiment from the ground during the operational period.

The experimental instrumentation can operate in the temperature range from 0° to 40°C, and can be stored in the temperature range from -20° to 40°C. During operations on the ground, class 10000 cleanliness must be maintained. The laser system would be sealed and filled with dry nitrogen before flight. When the pressure and dust levels in the cargo bay reduce to their nominal flight values, the system can be unsealed and exposed to the vacuum. The receiver system cannot operate during the daylight and would be sensitive to light scattered from other sources in the cargo bay. The experiment is not too sensitive to EMI, however, when the flashlamps fire a large amount of EMI is produced. The investigators intend to take spacial care in shielding this source of EMI.

On the ground, the experiment should be purged with dry nitrogen. There is a requirement to carry out the alignment of the laser with the receiver optics after the unit has been mounted in the cargo bay, or on a pallet or STR assembly. When the unit is uncovered, it should be in a class 10000 cleanliness area.

4.0 ASSESSMENT FOR STS FLIGHT

The requirements of this experiment are easily met by the STS. The experiment must be carried into space so that it can do topside probing of the atmosphere.

The following possible problems might exist: (1) the EMI levels of the flash lamp firings could be objectionable; (2) heat buildup within the laser could be damaging to the instrument in a vacuum environment, and (3) it may be difficult to meet the 1/2° accuracy requirement for a package that is mounted in the cargo bay without adding some active alignment system to the package.

4.1 Experiment Considerations

4.1.1 Design Suggestions

The size, weight and power of the experiment pose no design problems for the STS. The requirement to have a pointing accuracy of about 1/2° while mounted fixed in the cargo bay is within the capability of the Orbiter inertial measurement unit if the pointing direction of the laser can be referenced to the unit. This means that some means would have to be provided to measure the mechanical and thermal strains and to correct for them. Either a boresighting arrangement with light, mirror and photodetector or a separate inertial reference system on the laser would be able to correct for the misalignments in the cargo bay. Alternatively, the investigators could determine if an accuracy error of 2° corresponding to a ground track error of 7 km at 200 km altitude would be acceptable. The stability would be about 0.1° regardless of the accuracy and this is well within the required 0.5°.

The investigators realize that the thermal problem caused by the flash lamps will require that they take active means to cool the interior of the instrument. The amount of heat that is generated, 45 watts for one firing per 10 seconds, is small and should be no problem for the STS liquid cooling loop or even for passive radiation. It is suggested that the experiment be connected to the cooling loop in order to simplify the problem of thermal design. It should be remembered that the experimenters are responsible for their own thermal design. The use of passive radiation requires considerably more analysis of the thermal load than would the use of a heat exchanger connected to the Orbiter's liquid cooling loop.

The generation of EMI during the firing of the flash lamps should be controllable by the using of shielding materials and filtering of all power and data lines. The fact that the investigators have been able to fly a laser system on a rocket flight seems to indicate that the EMI problems can be handled.

4.1.2 Operation Restrictions

The investigators should consider firing the laser more often than once every ten seconds. They should investigate firing at a rate of once every two or three seconds. This would improve their mapping capability and the science yield. The increase of power consumption to levels of 1 kilowatt during the operation of the experiment still would leave an excess capability of 4.3 kilowatts for the rest of the payload in the case of a pallet only Spacelab flight or 1.7 kilowatts for a module plus pallet flight.

Orienting the Orbiter toward the nadir as required by this experiment would pose no problem especially since the experiment can be performed only in the night portion of the orbit. There are also no problems associated with the orbital requirements of the experiment. The Orbiter can fly at the 200 km level in the Y-POP Z-Nadir mode for a full seven-day mission and lose only about 14 kilometers in altitude. By using some of the OMS capability of the

Shuttle, it might even be possible to maintain the orbit in the 180 to 190 km altitude range. The maximum inclination orbit for the first Shuttle flights will be 57°, however, beginning in December 1982, launches from the Western Test Range will allow 90° inclination orbits to be obtained.

During the operation of the laser both on the ground and in orbit, care must be taken that the beam does not hit any photo-sensitive materials or humans. The flash lamps use a high voltage capacitor discharge and could pose some danger if care is not taken in handling the instrument when it is powered. Reflections from surfaces could also have enough energy to cause damage to unprotected eyes. In orbit, the only danger would be from reflection and this danger could be avoided easily by requiring a clear field of view for the laser at all times of operation.

During ground integration, the laser would be a sealed unit purged with dry nitrogen. During periods between performance of the experiment in orbit, the laser would be covered by its own lid in order to avoid accidental contamination and maintain proper storage temperatures.

4.1.3 Experiment Support Equipment

The experiment needs to be connected to the Orbiter or Spacelab power bus, data bus and probably cooling lines. Either the Orbiter or STS/Spacelab systems are adequate to supply the needs of the experiment. Some equipment will be needed in order to reference the optical axis of the laser/receiver to the Shuttle inertial measurement unit if the investigators wish to meet their 0.5° accuracy pointing requirement. Only a minimal number of status displays would be needed on the aft flight deck if the experiment is flown on a non-Spacelab mission. For Spacelab flights, the CDMS would provide all needed displays and data handling equipment.

4.1.4 Experiment Cost Considerations

In its present configuration, this experiment would impose no extraordinary added costs to an STS mission. If the repetition rate were changed from once per 10 seconds to once per 1 second, additional effort would have to be made in the thermal analysis and the design of the flash lamp power system. Once the instrument is built, it could be reflown repeatedly with little additional cost for equipment.

4.2 STP Integration Considerations

The instrumentation for this experiment would be the result of a continuing design effort which is presently in the sounding rocket stage. Delivery of flight equipment meeting the specifications outlined above could be in the early 1980's, provided the design effort were begun in the next two years.

The flight support equipment need for the experiment would include some status displays on the aft flight deck or the use of the Spacelab CDMS displays. A heat exchanger would probably be needed. Power and data connections would also be needed. Finally, if the 0.5° accuracy requirement for pointing the laser is retained, some method must be devised to reference the laser pointing direction to the Shuttle inertial measurement unit direction.

Figure 4-1 shows a sketch of the instrument. and Figure 4-2 shows a conceptual layout of the experiment on a Standard Test Rack. The experiment does not require the use of a Spacelab pallet since it could be mounted either directly to the cargo bay hard points or to a Standard Test Rack.

The following items must be purchased as optional STS services:

1. Mounting Hardware, a heat exchanger, data and power lines, and some display capability.
2. This experiment can make use of more than one day in orbit.

   The instrumentation could be designed to operate for up to 30 days if that were feasible, but a seven-day mission would be nominally the best length of time to carry out the observations.
3. Eventually, on reflights of the experiment, a launch from the Western Test Range would be required.

5.0 RECOMMENDATIONS AND REMARKS

1. This experiment is compatible with the STS/Spacelab or Standard Test Rack. It could fly on an STS sortie mission.
2. The investigators should examine the possibility of increasing the experiment repetition rate.
3. Either the accuracy requirement for pointing be increased to 2° or some method be designed to reference the laser axis to the Shuttle inertial measurement unit.
4. A test of the laser-generated EMI be performed before flight to assure that no problem arises in the Shuttle data handling systems or avionics units.
5. Care should be taken during all laser firings that the light is not shone directly or indirectly so that it causes eye damage.

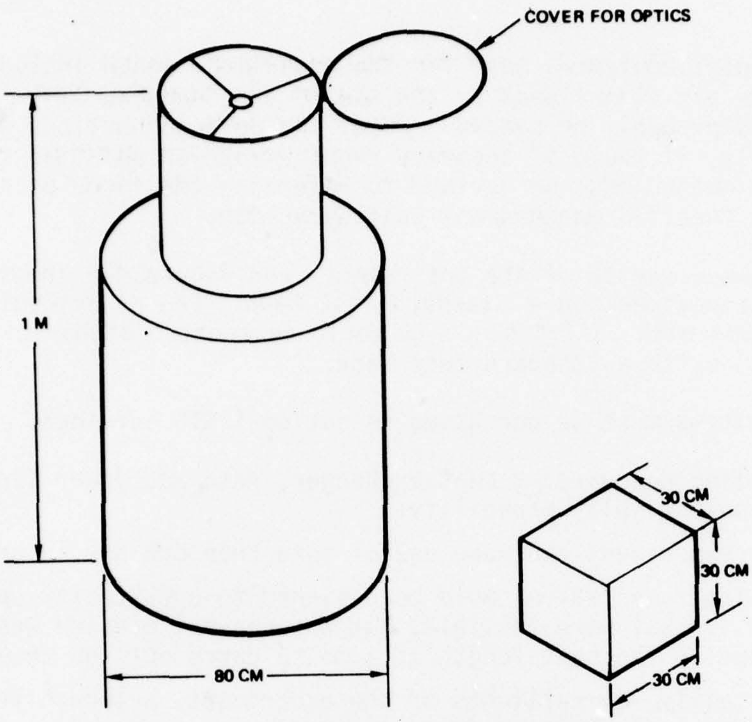

Figure 4–1. Sketch of the Instrumentation for the Atmospheric Topside Sounder Experiment

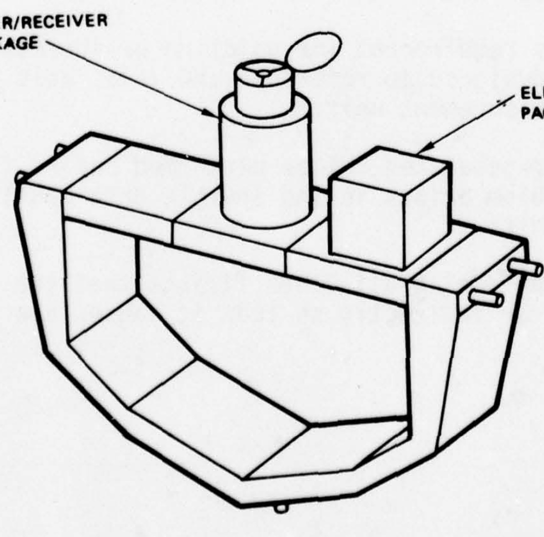

Figure 4–2. Conceptual Layout of the Atmospheric Topside Sounder Experiment on a Standard Test Rack

ENHANCED INFRARED EMISSIONS

1.0 EXPERIMENT IDENTIFICATION

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AFGL/OPI
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2.0 REQUIREMENT BACKGROUND

Response to STS Utilization Presentation (No. 65)

3.0 EXPERIMENT APPROACH

A telescope version of the cryogenically coded HIRIS interferometer will be operated in two modes: limb scans of the auroral oval and nadir measurements of the earth background. The auroral zone will be scanned from 90 to 150 km altitude. The target vector is 3.2° above the horizon (Z = 107°). The Orbiter will perform a roll maneuver to accomplish the horizon scan at 0.05 deg/sec. A rapid horizon scan will also be performed from tangent heights of 40-200 km at 0.05 deg/second. Nadir measurements are to be accomplished by swinging a gimballed mirror assembly into the field-of-view. Stabilization will be obtained with a control sensor and centroid tracker. The target is to be manually acquired using the Orbiter attitude control system and an astronaut "joystick." Once the target has been acquired, control is maintained by the centroid tracking system.

Problems will be encountered in combining this experiment with other STS payloads. The principal points of concern are:

1) Use of the Orbiter as a pointing platform requires continuous flight crew participation, provides a pointing accuracy of 0.50°, increases contamination environment and constrains other payload operations during EIRE duty cycle (67%).

2) Manual acquisition of targets for nadir pointing and targets of opportunity will require scientific as well as operational training for at least two crewmen. A payload specialist may be required.

3) Use of a gimballed pointing platform such as SIPS eliminates many of the concerns. In addition, it could eliminate the complication of the gimballed mirror in the instrument. In this case, a cryogenic canister similar to those proposed for the NASA/AMPS scanning instruments would be required. The cryogenic canister eliminates the requirements to transfer coolant across the gimbal systems.

4.1 Experiment Considerations

4.1.1 Mechanical

The size and weight of the equipment required for this experiment, i.e., 49.4 $ft.^3$ long and 1052 pounds, should pose no mechanical problems. Possible accommodation alternatives are Orbiter bay pointing platform or free flyer. The use of a free flyer will require a deployment mechanism.

4.1.2 Thermal

The operating temperature ranges of each of the major assemblies can be readily accommodated using the Orbiter thermal control system, instrument heaters and thermal blankets. The sensor temperature requirement makes necessary thermal control within the instrument design.

4.1.3 Attitude Control and Pointing

The stability requirements of $\pm$ 0.1 degree in each axis are within the capability of the Orbiter. However, the accuracy of the pointing vector is approximately 0.5 degrees. It seems that the accuracy required for this experiment would be greater than that provided by the Orbiter. The instruments should be mounted on a pointing platform, probably the Small Instrument Pointing System (SIPS).

4.1.4 Communication

Direct data transmission is available approximately 95% of the time using TDRSS; and if necessary, the Orbiter recorder could be used. The projected data rates of 500 bps are easily accommodated.

4.1.5 Command and Data Handling

All projected requirements are easily accommodated by the STS. Uplinked commands for instrument operations and down link data are minimal. Onboard data display and processing can be accomplished by hardwire or through the Orbiter/Spacelab computers.

4.1.6 Power

Operating power requirements of 450 watts are well within STS power constraints. Electrical energy consumption is fairly high since the instruments have a 67% duty cycle. For example, for a seven-day Spacelab flight, approximately 48 kw hr. is required; this is about 10% of the total available energy of experiment operation.

4.1.7 Controls and Displays

Astronaut joystick and target displays must be provided by experimenter.

4.1.8 Contamination

The environment in the Orbiter bay is generally not sufficiently clean for sensitive optical sensors at low orbital altitudes. Sensor covers should be provided for ground and early orbital operations. A more complete discussion on contamination is included in Response #35, "Infrared Background Sensor."

4.2 Integration Considerations

The drawing shows a possible installation on the small instrument pointing platform in turn mounted on a Spacelab pallet. The SIPS has two instrument-carrying canisters, each supported at its center by a yoke which can rotate independently of the other canister in an up-down direction (120 degrees freedom). Each canister in turn is connected to the yoke so as to provide a limited (± 10 degrees) left-right rotational degree of freedom. Both yokes are attached to a common ± 180-degree azimuth gimbal drive at the base. An optional roll gimbal about the instrument line of sight can be added internally to each canister. The instrument package for this experiment could use the standard SIPS canister depending on the sensor cooling design selected.

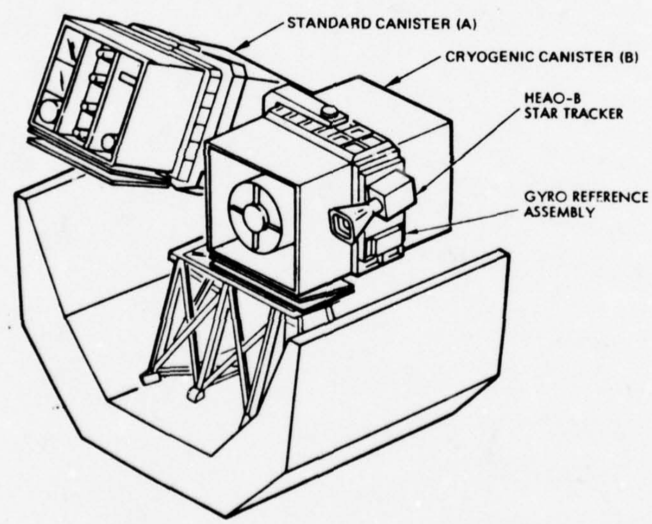

Figure 4-1. Small Instrument Pointing System (SIPS)

5.0 RECOMMENDATIONS AND REMARKS

1) Equipment should be mounted on a pointing system such as SIPS or POINTS.

2) Sensor covers should be provided.

3. Flight crew manual operations should be reviewed. Use of 67% of the crew time to perform repetitive operations does not seem appropriate.

DUCTED IONOSPHERIC RADIO PROPAGATION EXPERIMENT

1.0 EXPERIMENT IDENTIFICATION

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2.0 REQUIREMENT BACKGROUND

Response to STS Utilization Presentation (Response Sheet 42)

Work Unit Number: Project 4600 Task 16

3.0 EXPERIMENT APPROACH

Ducted ionosphere radio propagation modes have been observed in which received signals greatly exceed normal ionospheric transmission modes in the 3 to 30 MHz frequency range. The purpose of this experiment is to identify the global distribution of such propagation paths and their temporal pattern. The experimental approach is to use ground based transmitters and to record signal levels on the Shuttle. A second mode of operation is to locate the transmitter on a free flyer released from the Shuttle, and to record the signal levels received on the Shuttle as the transmitter drifts away as far as the other side of the earth. This second approach is, of course, more costly in that a free flyer mounted transmitter is required. The advantage is that more possibly useful transmission paths are investigated.

The "roof" of these ducts are located in the lower portions of the ionospheric F-layer. Thus Shuttle altitudes lower than 200 km, preferably 180 km, are required. Orbital inclinations as high as 70° are desirable, 56° inclinations are acceptable, and 50° is poor. Figure 3-1 shows a conceptual picture of the Ducted Ionospheric Radio Propagation Experiment. Table 3-1 lists the main characteristics of this experiment.

Table 3-1. Experiment Characteristics

SHUTTLE MOUNTED EQUIPMENT	
Receiving Antennas:	Dipoles and loops; ∿2 meter cube volume; folded for storage in 1 meter cube and deployed; provides directional measurements
Receiver:	1m x 1m x 1m; 150 pounds; 150 watts
Commands:	10 operating modes commandable from ground
Telemetry:	Analog 100 Hz bandwidth 24 hours/day (1 tape/day)
Operating Temperature:	0-40°C (commercial specifications)
Crew Time:	Antenna deployment and retrieval or ejection: 1 hour total time. Automatic operation; otherwise with crew override in emergency
FREE FLYER	
Transmitter:	10 to 30 pounds; 10 to 100 watts
Antenna:	2 meter dipole when deployed
Crew Time:	Deployment of free flyer

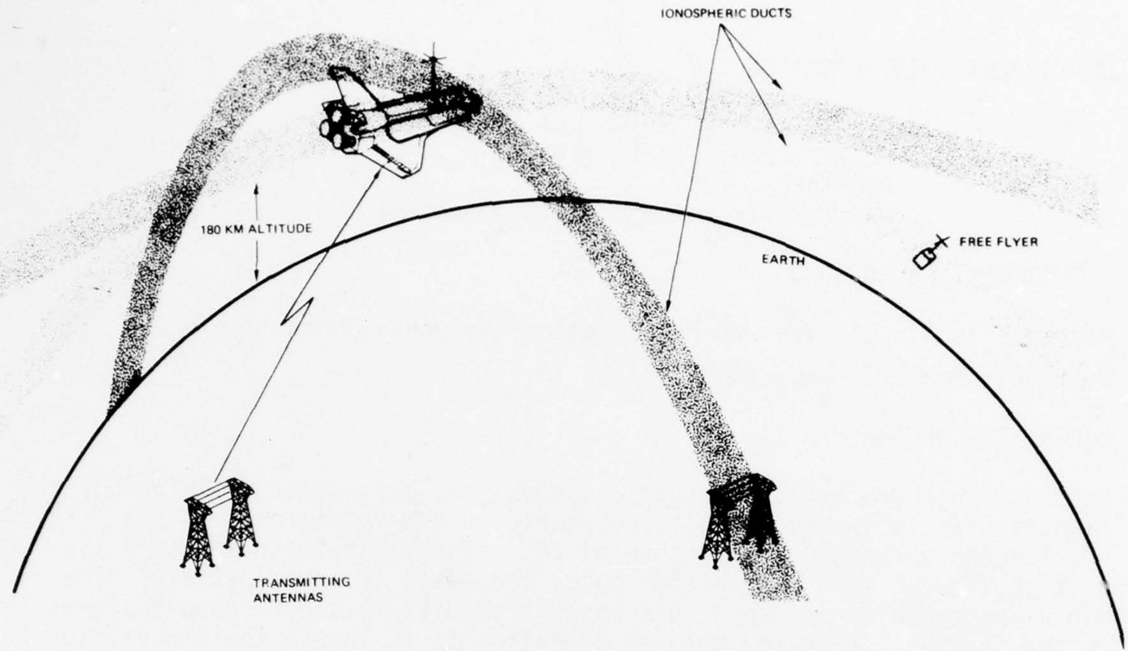

Figure 3-1. Ducted Ionospheric Radio Propagation Experiment

4.0 ASSESSMENT FOR STS FLIGHT

No serious problems are expected in flying this experiment. The techniques of flying automated receivers having a broad frequency range (3-30 MHz) have been employed in many unmanned spacecraft.

- Electromagnetic compatibility (EMC) problems are minimized by narrow bandwidth (100 Hz) tuners which are swept in frequency.
- Signal levels to be detected are high.

4.1 Experiment Considerations

4.1.1 Design Suggestions

The receiving antenna system on the Shuttle should be designed to provide three-axis directional measurements without requiring any particular orientation of the orbiter since maneuvering is very expensive in crew time, flight time, fuel, etc. Shuttle location and altitude data must, of course, be merged with the experiment data since these are important components of the experimental results. With automated operation of the receiving equipment, once the antennas are deployed, crew time and Shuttle orientation requirements are minimal. Ground control, by commands, of receiving equipment operating modes also minimizes crew time requirements.

4.1.2 STS Integration Considerations

Integration of this experiment on STS presents no unusual problems. The electronic equipment box may be pallet mounted or may be located in a rack in Spacelab. The receiving antenna array should be deployed to be about 2 meters away from the orbiter body. Figure 4-1 shows one integration option in which the entire equipment including electronic equipment and stowed antenna are located in a 1 x 1 x 2 meter box on the Standard Test Rack.

4.1.3 Experiment Cost Considerations

The most expensive part of this experiment, the electronics box and boom, are reusable. Restowage of the antenna array may be more costly than ejection. The free flyer portion of this experiment may be postponed until these become readily available. Integration into the free flyer should not entail special costs since many unmanned satellites have been built and flown which incorporate transmitters in this frequency range and operating power.

4.2 STP Integration Considerations

4.2.1 Delivery Lead Time

All of the equipment associated with this experiment are state-of-the-art and similar equipment have already been flown. For STS, consideration should be given to the use of modified commercial equipment to reduce costs.

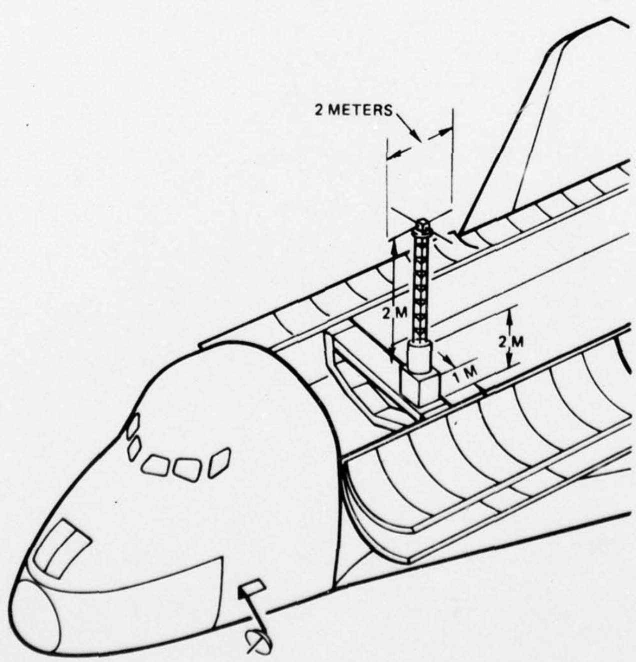

Figure 4-1. One Possible Integration Configuration of Experiment

4.2.2 Preparation

Preparation for this experiment includes items such as analytical and experimental work towards evaluating effects of the Shuttle on expected signal levels, i.e., antenna patterns and electromagnetic interference (EMI).

4.2.3 Flight Support Equipment

A minimal "go-no-go" electrical checkout set is all that is required for STS interfacing. All other checkout and calibrations may be carried out at the experimenter's facility. The other area of flight support equipment that may be required would be in the area of software and possibly hardware for realtime data analysis to assure that good data is being collected.

5.0 RECOMMENDATION

This experiment may be readily accommodated in the STP program and will provide valuable data. The STS interfaces do not present any new problems that have not already been addressed. Flight restrictions, crew time, safety, and costs are not out of scope.

The requirement for higher orbit inclinations and lower orbital altitudes are not extraordinary, and this experiment could be included in those STP missions which satisfy these conditions.

GROWTH OF CINNABAR (α - HgS) IN A LOW GRAVITY ENVIRONMENT

1.0 EXPERIMENT IDENTIFICATION

Mr. Charles S. Sahagian, Principal Investigator
RADC/ESM
Rome Air Development Center
Hanscom AFB, Bedford, ME 01731
Sponsor Agency: RADC/ES

2.0 REQUIREMENT BACKGROUND

Response to STS Utilization Presentation (Response Sheet No. 46)

Work Unit No. 2306J1(6.1)

3.0 EXPERIMENT APPROACH

A premixed mercury chloride-sodium sulfide solution is pressurized with oxygen at constant temperature to force the growth of a cinnabar (mercury sulfide) single crystal in a specialized reaction chamber. The objective is to produce more perfect α - HgS crystals for applications in non-linear electronic and acoustic optical and acoustic surface wave devices. Figure 3.1 schematically describes a reaction vessel and attendant support equipment that could be used to perform this experiment.

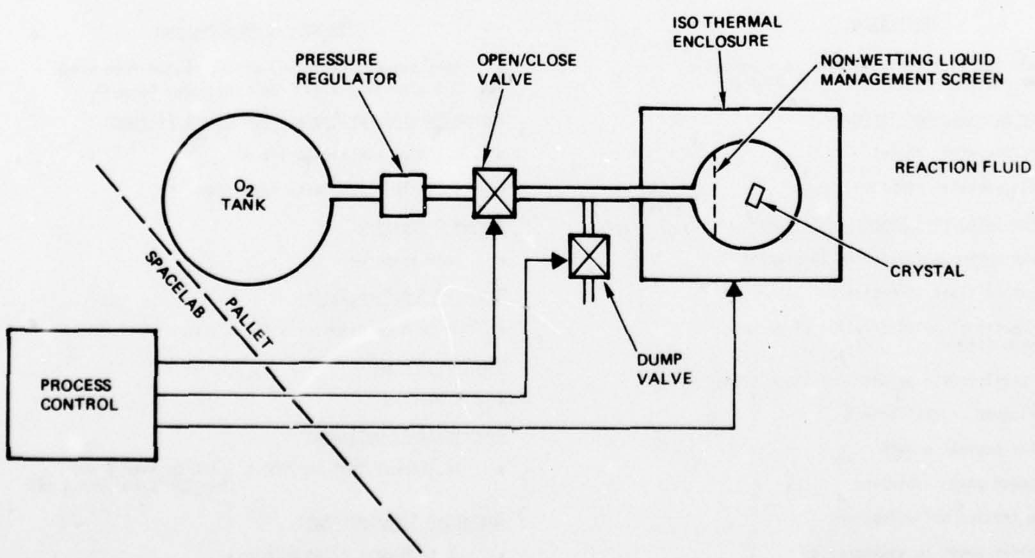

Figure 3.1. Schematic Representation of the Equipment Necessary to Grow α– HgS Single Crystal

4.0 ASSESSMENT FOR STS FLIGHT

No serious problems are expected in integrating this experiment with other experiments requiring isothermal conditions at mild (75°C nominal) temperature during crystallization.

4.1 Experiment Conditions

The experiment is to be performed in a closed reaction container under a nominal oxygen pressure of approximately 10^6 N/m^2. Acceleration level during processing should be 10^{-3} g_e or less, including g-jitter. Operating temperature is 75°C, and temperature fluctuation should be no more than ±3°C. Reaction chamber should be isothermal to at least 1°C/cm.

This experiment could be performed in the multi-purpose furnace that will be developed by NASA for their materials processing program. Characteristics of this furnace are described in Table 4.1. However, the relatively low temperature at which this experiment is performed suggests that a specially developed low temperature furnace might be more appropriate. Additionally, time-lining of the experiment protocol indicates that a processing time of about 5 days is required. This would tie up the multi-purpose furnace for an entire flight and so would put the experiment in competition with NASA's experiment program for use of the facility. The experiment can be designed to be fully automated with only minor crew participation.

TABLE 4.1

NASA MULTI-PURPOSE FURNACE

Description

- This facility contains apparatus suitable for material processing up to 2200 C

Samples Accommodated Include:

- Encapsulated samples
- Sting-mounted samples

Research Categories Supported Include:

- Heterogeneous nucleation in glasses
- Complex glass formation
- Directional solidification of ceramic compositions
- Crystal growth by chemical vapor transport
- Bridgman crystal growth
- Flux crystal growth
- Liquid phase sintering
- Controlled solidifcation
- Molten zones in microgravity
- Directional solidification

Processing Atmospheres:

- Air, oxygen, nitrogen, argon, oxygen-free argon - 0.1 n/m^2 to 4 x 10^5 n/m^2 (Hydrogen later?)

Vacuum Purging for Process Atmosphere Filling:

- 0.1 n/m^2 venting to space
- 1.3 x 10^{-4} n/m^2 with turbopump

Vacuum Processing:

- Same as purge

Range of Specimen Sizes:

- Up to 6 cm diameter x 25 cm long

Electrical Power:

- 1 kw/3 kw

Processing Temperatures:

- Isothermal 700° to 2200°C (.1°C/cm over 6 cm) (max 300°C/cm over 6 cm)

Operating Time/Specimen:

- Up to 5 days (7200 minutes)

Accelerations:

- Depends upon shuttle movement

Number of Specimens/Mission:

- 4 to 160

The specimen(s) might be able to remain in the reaction container to minimize physical damage during landing and subsequent handling. As mentioned previously, the only STS constraint during processing is maintenance of a low acceleration level.

The necessary STS interfaces on power to the isothermal enclosure, the oxygen tank open/close valve, the dump valve to relieve the oxygen pressure after the reaction is completed and a thermal sensor to maintain the proper reaction temperature during process. These interfaces are minimal and thus do not impact cost considerations appreciably.

4.2 STP Integration Considerations

The experiment is not complex and there are no particular requirements which would necessitate abnormally short or long delivery lead times.

In order to obtain verification of the enhancement of material properties due to low gravity processing, ground processed control specimens are necessary. Utilization of equipment identical to that on board the STS will be required in order to control process variables as much as possible. However, the one gravity experiment can be performed at any convenient facility either before or after flight and thus should have no impact on STP integration.

5.0 RECOMMENDATIONS AND REMARKS

This experiment would be located in the unpressurized Spacelab pallet structure in the Shuttle bay, if the NASA Multi-Purpose Furnace is used. If a new low temperature furnace were developed, power requirements might be low enough to make use of the Standard Test Rack. It would be possible to develop a furnace for performing the experiment in the Spacelab pressurized module, however, the additional cost and the potential hazard of releasing mercury compounds into the atmosphere mitigates against this operational mode.

BUBBLE MEMORY EXPERIMENT

1.0 SOURCE

Dr. Brian E. White
The MITRE Corp.
Dept. D91, Mail Stop B230
Box 208, Bedford, MA 01730

2.0 REQUIREMENT BACKGROUND

Response to STS Utilization Presentation (Response Sheet No. 43)

Supplementary information

a) "Orbiting Radiometer Preliminary Calculations" by W. T. Branden, R. L. Jeffcoat, J. C. Maxwell, MITRE Corp. Working paper 20496, 12 Dec. 1975.

b) "Data Courier Satellite System Concept" by W. T. Branden, MITRE Corp. Report MTP-164, Sept. 1975.

3.0 EXPERIMENT APPROACH

3.1 Background

Magnetic domain bubble memories are considered a key element in two proposed low-earth-orbit satellite concepts that require massive data storage capacity at low weight and power consumption:

1. Orbiting radiometer satellites for observation of worldwide frequency spectrum utilization in the UHF range.

2. Data courier satellites designed for transmittal of large quantities of digital data over intercontinental distances, for use by government agencies as well as industry. The satellite would store data it receives on overflight of a transmitting station and dump those data on passing the designated receiving station, one or several orbital revolutions later. Memory storage capacities of 50 megabits and greater are envisioned.

Orbital altitudes of about 900 nautical miles are being contemplated for both mission classes, corresponding to orbit periods of 2 hours with repeating ground tracks every 12th orbit, i.e., once per day.

The technology of bubble memories has advanced to the point where use of these mass data storage devices will be feasible and highly attractive for the two mission classes as early as 1980. However, since the satellites will be operating in an environment of increased trapped radiation compared with that existing at synchronous altitude or at altitudes of only a few hundred nautical miles, the question of possible degradation of the solid-state bubble memories due to this radiation during the satellites' orbital life becomes a matter of concern.

3.2 Objective

The proposed STS experiment has the objective of exposing bubble memory specimens of representative designs and material properties to the orbital environment and to determine radiation-induced degradations, if any, in relation to the length of exposure.

Neither the experimental procedure, equipment requirements, measurement techniques nor the type of bubble memory specimens to be selected for the experiment have been defined since the program is still in an early formative stage. The purpose of the assessment will be a discussion of practical factors, dictated by accommodation on the STS system, that may influence the conduct of this experiment.

4.0 ASSESSMENT FOR STS FLIGHT

Even with no detail of the orbital test of bubble memory specimens defined as yet, it is apparent that prolonged exposure to the orbital radiation environment is the principal requirement. STS utilization for the experiment therefore implies placing the specimens, the measurement and the data acquisition equipment onboard the Long Duration Exposure Facility (LDEF) which will remain in orbit for six to eight months before being retrieved and returned to Earth by the Shuttle Orbiter.

The small size, weight and power requirements of the bubble memory units and the reasonably small and simple experiment support equipment to be used probably will fit readily on one of the support modules, or experiment trays, provided by LDEF for experiments of this class.

Accommodation of the experiment will be facilitated through utilization of small plug-in electrical support units known as Electric Power Data Systems (EPDS) to be made available on some of the LDEF experiment trays. These EPDS's are currently under development by Lewis Research Center, and several units will be acquired by the STP Office to support small size experiments such as this.

4.1 Experiment Considerations

4.1.1 Dependence of Particle Flux on Altitude

A principal question regarding the exposure test onboard the LDEF involves the large difference in energetic particle flux at the LDEF orbital altitude (200 to 250 n.mi.) and the operational altitude (about 900 n.mi.) of the proposed satellites that will make use of bubble memories. The flux of potentially damaging electrons and protons is at least two to three orders of magnitude greater at the 900 n.mi. satellite altitude.

In order to match the fluence to which the test specimen is exposed to that encountered at the higher altitude either a much longer stay time in the low altitude orbit would be necessary (not feasible with LDEF), or the radiation shielding of the test item must be made much less than that of the unit to be carried by the operational satellites. In addition, even if fluences can be adequately matched by this approach, the effects observable on the test specimen will not necessarily be representative because of the difference in the energy spectra of the particles encountered. A subsequent higher altitude test mission by a free-flying satellite may have to be contemplated. Obviously such a test would be much more costly and particularly when it includes retrieval by the Shuttle Orbiter to permit examination of the exposed test item.

4.1.2 Estimated Electron Fluence in High and Low Altitude Orbits

Particles of principal concern in this test will be high energy electrons at energy levels of 3 Mev and greater. Figure 4.1-1 illustrates the 3 Mev electron flux variation with altitude in terms of contours of 10^5 electrons/cm^2 sec at the LDEF orbit altitude (220 n.mi.)intermediate altitudes (380 and 540 n.mi.) and the altitude of the proposed operational satellites (910 n.mi.) These maximum flux contours are located in the area of the South Atlantic magnetic anomaly centered at 35^o W longitude and 30^o S latitude. This is the only area of high flux levels at the altitudes considered here. Two orbital tracks at intermediate inclinations (30 and 60 degrees) are shown in the graph to illustrate differences in exposure frequency and duration. The lower inclination orbit generally encounters the area of high flux more often and for longer periods as it shifts westward in successive orbital passes. A rough estimate of the daily 3 Mev electron fluence is 5×10^7 electrons/cm^2 for the 220 n.mi. orbit and at least 10^{10} electrons/cm^2 for the 910 n.mi. orbit assuming 30 degrees orbit inclination.

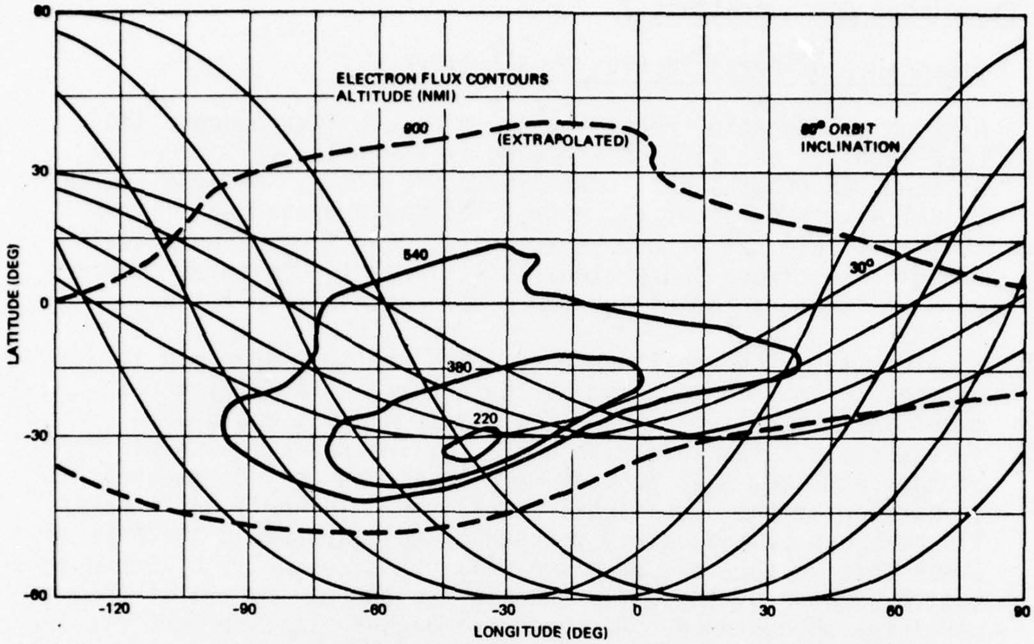

Figure 4.1-1. Flux Contours of > 3 MeV Electrons at Four Altitudes (10^5 Electrons/cm^2 sec) and Satellite Groundtracks (30 and 60 Deg Inclination)

4.1.3 Suggested Orbit Characteristics

From the above data and from the geographical encounter conditions illustrated in Figure 1 it is apparent that an LDEF orbit best suited for this experiment would have an inclination between 30 and 35 degrees and an altitude near the upper limit of the Shuttle performance capability, in the range of 300 to 400 n.mi. The preferred orbit inclination range of 30 to 35 n.mi. is also that which permits the highest orbital altitude and the maximum payload weight (see Figure 4.1-2 for Shuttle flights launched from Kennedy Space Center), a factor very beneficial for purposes of this experiment. However, the determination of orbit characteristics will also depend on the many (up to 72) other experiments to be accommodated on the same LDEF flight. Guidelines for setting priority requirements in a vehicle such as the LDEF that must serve a large number of unrelated experiments still require further definition.

4.2 STS Integration Considerations

Weight, volume and power requirements for the experiment have not been defined as yet. However, it is reasonable to expect that the total weight, including support equipment, will not

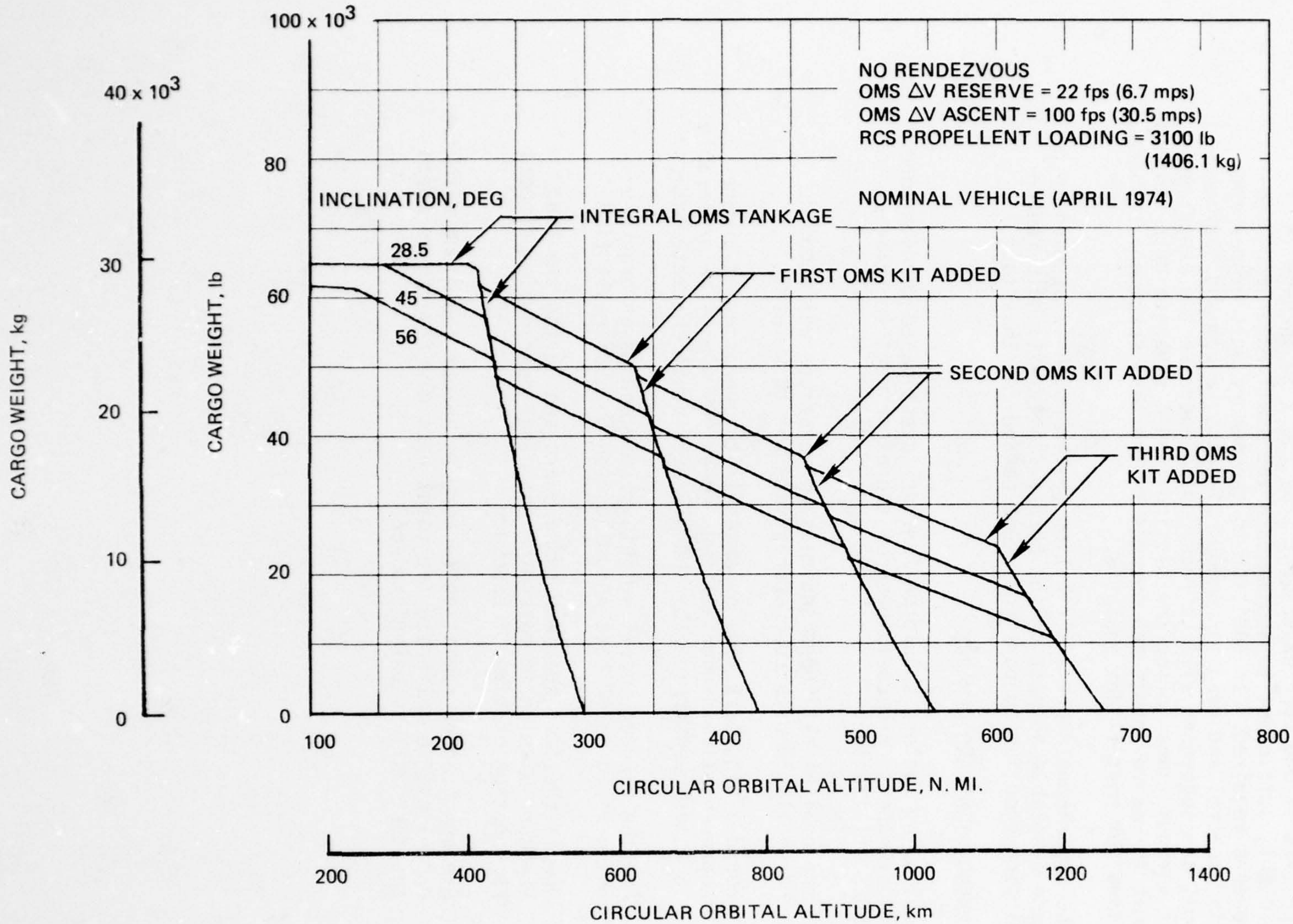

Figure 4.1-2. Shuttle Orbiter Cargo Weight Versus Circular Orbital Altitude – KSC Launch, Delivery Only

exceed 40 lb and that power requirements are less than 20 W. The size of one LDEF experiment tray (50 x 30 inches) is probably sufficient to accommodate the small experiment components required. In addition to the memory unit proper and its control and read in/readout circuits, a command storage unit and sequencer will be required to exercise the experiment at various times during the mission, and a separate data storage unit to record inputs and outputs to the test specimen at these times for post flight analysis.

The Experiment Power Distribution System which is assumed to be made available as part of the LDEF facility will help reduce mission-peculiar support equipment needed for this test.

5.0 RECOMMENDATIONS AND REMARKS

The following factors should be further investigated to determine practical aspects and usefulness of the experiment in relation to STS/LDEF utilization:

- Are orbital altitudes and mission durations which could be available with LDEF appropriate for this test?
- Would a piggyback ride on a satellite operating at a more representative higher altitude (near 900 n.mi.) be preferable to a test constrained by the STS/LDEF orbit characteristics?
- Are the assumed modest weight, volume and power brackets which would facilitate accommodation on LDEF realistic?
- How is the test specimen to be exercised during the long exposure mission to determine potential degradation effects as function of time of exposure?
- Are any trapped radiation sensors to be carried along with the experiment to calibrate the total particle fluence to which the test unit is being exposed?

LOW LEVEL ASSESSMENTS

The assessments contained in this section consist of a brief statement describing the potential accommodation of the subject experiment by the STS and one of the available payload carriers.

The experiments evaluated were those that remained after more than 16,000 active Work Unit Summaries (Form 1498's) were reviewed by TRW's specialists.

The screening initially was performed to determine if the experimenters' objectives would be enhanced by a space experiment. Once this conclusion was reached, the experiment was assessed for compatibility with the STS. Since the information contained in the Form 1498 is, at best sketchy, the ability to determine a viable space experiment using the STS is limited.

However, the combination of the "Medium" level and "low" level assessments covers a wide variety of space experiments and should provide the DoD scientific community with examples of the utility of STS as an experiment carrier.

There are 43 classified experiments included in this section which appear to have a compatibility with the STS. Discussion of these has not been included because of their classification.

INDEX OF LOW LEVEL ASSESSMENTS

AD-A064 766 TRW DEFENSE AND SPACE SYSTEMS GROUP REDONDO BEACH CALIF F/G 22/2
STS UTILIZATION STUDY EXPERIMENT ASSESSMENTS.(U)
DEC 77 F04701-77-C-0112
UNCLASSIFIED TRW-31933-6001-RU-00 SAMSO-TR-77-188 NL

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EXPERIMENT ASSESSMENTS

ID No. (Response Sheet or Assessment #)	Principal Investigator	Agency	Title	STS Facility (Spacelab, LDEF, etc.)	Assessment Comments
DF368286	M. H. Cohen Watkins-Johnson Co. Palo Alto, CA	SAMSO	High Efficiency Ten-Watt Output Power Traveling Wave	Spacelab Pallet or Free Flier	Instrument developme[illegible] contract. The TWT could be flight tested on STS or a recoverable free flier.
DF368429	Charles C. Badcock Aerospace Corp.	SAMSO	Chemistry of Space Power	N/A	Battery study. No apparent need for STS flights. Could provide test for developed batteries.
DF368177	J. Bernard Blake Aerospace Corp.	SAMSO	Trapped Radiation and Hazards	N/A	Data study contract. Could use data generated on STS free flier spacecraft.
DF368436	Joseph F. Fennel Aerospace	SAMSO	Space Plasma Effects	Free Flier	Contract is for SCATHA instrument. Follow-ons could be launched by STS. An IUS booster would be needed to get to geosynchronous orbit.
DF368373	Paul Nosal Gruuman Aerospace Bethpage, NY	SAMSO	Space-Based Surveillance System Study	N/A	Design study contract of radars. Design should be made compatible with STS.
DF368394	Daschan Pecka Lockheed Missiles and Space Company	SAMSO	Teal Ruby Space Experiment	Spacelab Pallet	Current design study. Instrument will be designed to be launched from STS. Some tests may need pallet mounted instruments.
DF368393	William S. Hinds Rockwell Internat'l Downey, CA	SAMSO	Teal Ruby Space Experiment	Pallet or Free Flier	Current design study. Instrument will be designed to be launched by STS. Some tests may need pallet mounted instruments.
DF368372	Albert F. Caprioglio TRW	SAMSO	Space-Based Radar Surveillance System Study	Spacelab Pallet	Current contract for design study of space radar systems. Final system should be made compatible with STS.

EXPERIMENT ASSESSMENTS

ID No. (Response Sheet or Assessment #)	Principal Investigator	Agency	Title	STS Facility (Spacelab, LDEF, etc.)	Assessment Comments
DF368208 DOD Only	F. Nigro AVCO Corporation	SAMSO	Responsive Survival Aid Design		Possible STS Experiment.
DF368378 DOD Only	R. Kiem Eastman Kodak	SAMSO	Laser Beam Expander Technology		Possible STS Experiment.
DF368207 DOD Only	J. Potter General Electric	SAMSO	AFSATCOM II Manufacturing Impact Study		Possible STS Experiment.
DF368325 DOD Only	J. Steffbs Hughes Aircraft	SAMSO	Project II		Possible STS Experiment.
DF368376 DOD Only	Howard Courtney Hughes Aircraft	SAMSO	Advanced Optics Technology		Possible STS Experiment.
DF368374 DOD Only	P. Williamson Lockheed	SAMSO	Advanced Optics Technology		Possible STS Experiment.
DF368381 DOD Only	L. Lunsford Lockheed	SAMSO	Space Laser Experimental Definition		Possible STS Experiment.
DF368379 DOD Only	R. Rowley Perkin-Elmer	SAMSO	Laser Beam Expander Technology II		Possible STS Experiment.
DF368375 DOD Only	Paul Horio Rockwell International	SAMSO	Advanced Optics Technology		Possible STS Experiment.
DF368380 DOD Only	J. Murphy Rockwell International	SAMSO	Space Laser Experimental Definition		Possible STS Experiment
DF368354 DOD Only	D. DeMaio Science Applications	SAMSO	High Energy Laser Cost and Performance Study		Possible STS Experiment.
DF368209 DOD Only	E. Stanszak TRW	SAMSO	Experimental Evaluation of a Shrouding Concept		Possible STS Experiment.
DF368324 DOD Only	W. Cannon Vought Corporation	SAMSO	Project II		Possible STS Experiment.

EXPERIMENT ASSESSMENTS

ID No. (Response Sheet or Assessment #)	Principal Investigator	Agency	Title	STS Facility (Spacelab, LDEF, etc.)	Assessment Comments
DF368200 DOD Only	J. Oliver GTE Sylvania	SAMSO	Radar Radiation Receiver Design Definition		Possible STS Experiment.

EXPERIMENT ASSESSMENTS

ID No. (Response Sheet or Assessment #)	Principal Investigator	Agency	Title	STS Facility (Spacelab, LDEF, etc.)	Assessment Comments
DF105800 DOD Only Trade Sec.	Dr. Warren Ruderman Interactive Radiation Northvale, NJ	AFAL	New Synthesis Methods for CD-A Growth		Possible STS Experiment.
DF102440	Dr. R. Hopkins Westinghouse Pittsburgh, PA	AFAL	Reproducible Growth of High Quality ND-CALASOAP	Spacelab Pallet	Crystal growth experiment. Future laser crystals might be grown in Spacelab materials processing facility.
DF108120	Gary L. McCoy	AFAL	Epitaxial III-V Compound	Spacelab, Pallet (hazards to be determined)	Skylab experiments on similar processes showed beneficial results.
DF106040	Stanley R. Burlage Cleveland Crystals Cleveland, OH	AFAL	Investigation of Growth Inhibition in CD-A	Pallet Only (hazard)	This experiment similar to DF105800, thus crystal perfection is major experiment target.
DF101130 U.S. Gov't Prelim.	B. M. Murphy Raytheon Company Wayland, MA	AFAL	Dual Frequency SATCOM Power Amplifier		Possible STS Experiment.
Response Sheet No. 59	Dr. Robert Q. Fugate	AFAL	Space Radiometer	Pallet or Free Flier	Instruments could be flown on STS Orbiter or free flyer mission.
DF100870 DOD Only	John Gilligan ITTRI	AFAL	EO Materials Studies		Possible STS Experiment.

EXPERIMENT ASSESSMENTS

ID No. (Response Sheet or Assessment #)	Principal Investigator	Agency	Title	STS Facility (Spacelab, LDEF, etc.)	Assessment Comments
Response Sheet No. 60	Dr. Robert Q. Fugate	AFAL	Dual Band Airborne Radiometer	Pallet	Instrument designed for STS Orbiter mission. Mounted on pointing system or to a pallet
Response Sheet No. 61	D. A. Zann Lt. Col. Lindemuth	AFAL/SAMSO	Fiber Optics Communications for Aerospace Systems	Free flyer	Instrument designed for STP satellite. Launched by STS and recovered.
Response Sheet No. 25	R. L. Remski	AFAL	X-Band FET Amplifier for Space	Pallet	Captive flight on Orbiter for space qualification.

EXPERIMENT ASSESSMENTS

ID No. (Response Sheet or Assessment #)	Principal Investigator	Agency	Title	STS Facility (Spacelab, LDEF, etc.)	Assessment Comments
DF457670	H. Rogers Hughes Aircraft El Segundo, CA	AFAPL	Failure Mechanisms in Nickel Hydrogen Cells	Any	Battery hardware development. Batteries could be tested on orbiter or free fliers, or LDEF.
DF456340	Irvin F. Luke	AFAPL	Evaluation of Metal/Gas Cells for Aerospace Power Source	LDEF	Battery hardware development. Batteries could be tested on free flier, orbiter or LDEF.
DF458720	William Harsh Eagle Picher Ind. Joplin, MO	AFAPL	Nickel Hydrogen Space Experiment	Free Flier or LDEF	Battery hardware development. Batteries could be tested on recoverable free flier or LDEF.
DF452550	G. Wolfe Hughes Aircraft El Segundo, CA	AFAPL	Hardened Solar Power Systems (HASPS)	Pallet or Free Flier	Solar power system hardware. Hardware could be tested on STS or free flier. Free flier would need IUS booster.
DF458000	L. Spicer Hughes Aircraft	AFAPL	Nickel Hydrogen Battery	Free Flier	Battery hardware development. Batteries could be tested on low altitude STS-launched free flier.

EXPERIMENTS ASSESSMENTS

ID No. (Response Sheet or Assessment #)	Principal Investigator	Agency	Title	STS Facility (Spacelab, LDEF, etc.)	Assessment Comments
Response Sheet No. 30	M. Duhl	AFML	Thermal Fatigue Behavior of Composite Materials	LDEF	Needs a position for maximum cyclic solar exposure.
Response Sheet No. 30	M. Duhl	AFML	Adhesive/Structural Bonding in a Space Environment	Pallet or STR	Active experiment requires sample return.
Response Sheet No. 30	M. Duhl	AFML	Outgassing of Composite Materials	LDEF or STR	Active measurements to be taken but Orbiter may have poor environment.
Response Sheet No. 30	M. Duhl	AFML	Space Charging of Composite Materials	Free Flyer	Needs high altitude for natural charging or a very elaborate experiment with induced charging.
Response Sheet No. 30	M. Duhl	AFML	Long Term Radiation Effects on Composite Materials	VLDEF or Free Flyer	Needs very long time in orbit or high radiation environment orbiter.

EXPERIMENT ASSESSMENTS

ID No. (Response Sheet or Assessment #)	Principal Investigator	Agency	Title	STS Facility (Spacelab, LDEF, etc.)	Assessment Comments
DF680130	P. Peterson Honeywell Bloomington, MN	AFML	Gallium Phosphide Materials Development for Satellite Attitude Sensors	Spacelab Pallet	Skylab experiments on similar processes showed beneficial results.
DF680200	M. Arst Rockwell Internat'l Anaheim, CA	AFML	Silicon Material Development for LADIR (Low Cost Arrays for the Detection of Infrared) Extrinsic Silicon Detectors	Spacelab Module or Pallet	Crystal growth of high purity. May be able to use Spacelab materials processing facility.
DF682690	F. Schmid Crystal Systems, Inc. Salem, MA	AFML	HEM Growth of Sapphire Bonle for IR Dome	Spacelab	This experiment is too complex for early flight. However, excellent candidate for future missions.
DF680730	George Webb Eagle Picher Ind. Miami, OK	AFML	Purification, Controlled Dopeing and Crystal Growth of II-VI Semiconductors	Spacelab Pallet	Purification and crystal growth could be improved by low-G processing. Dopeing aided by improved crystals.

EXPERIMENT ASSESSMENTS

ID No. (Response Sheet or Assessment #)	Principal Investigator	Agency	Title	STS Facility (Spacelab, LDEF, etc.)	Assessment Comments
DF901130	Worth P. Allred Crystal Specialities Monrovia, CA	RADC	Growth of Gallium Arenide	Pallet Only (hazard)	Crystal improvements on similar materials has been shown. Equipment modification may be necessary.
DF900150	David Miller Arcon Corp. Wakefield, MA	RADC	Global Ionospheric Model	N/A	Computer modeling of ionosphere. Could use ionospheric data by STS flights to check model.
DF708940	Ralph Berggren Itek Corporation Lexington, MA	RADC	Halo Optical Technology	N/A	Current optical design studies. Developed instruments could be flown as STS payloads.
DF901150	Robert H. Eather Keo Consultants Newton, MA	RADC	Imaging All Sky Photometer	Orbiter	Instrument development for aircraft flights. Could be flown on STS Orbiter mission with a pointing system.
DF742520	B. Marcovici Hughes Aircraft Fullerton, CA	RADC	Space-Based Laser Surveillance Technique	N/A	Requirements assessment study. Future lidar systems could fly on STS.
DF901290	Edward J. Weber	RADC	Large Scale Optical Mapping of the Ionosphere	Orbiter	Current airplane experiments. These experiments could be carried out better on STS Orbiter flights.
DF739390	Edward J. Femouw Stanford Research Institute Menlo Park, CA	RADC	Continued Modeling of Transionospheric Radio Communications Channel	N/A	Computer programs for scintillations. No direct use of STS, but should use data from STS flights to compare.

EXPERIMENT ASSESSMENTS

ID No. (Response Sheet or Assessment #)	Principal Investigator	Agency	Title	STS Facility (Spacelab, LDEF, etc.)	Assessment Comments
DF707370	Steven Weisbrod Teledyne, Inc. San Diego, CA	RADC	Adaptive Techniques for Tropospheric ERR	N/A	Radar data analysis. STS could be used as a calibrated target to generate new data.
DF749530 DOD Only	M. Persky Block Engineering	RADC	Aircraft Radiometry		Possible STS Experiment.
DF730950 DOD Only	H. Strachman ESL Inc.	RADC	Bistatic Radar Analysis		Possible STS Experiment.
DF712080 DOD Only	B. Dersch General Electric	RADC	Tide Re-entry Study		Possible STS Experiment.
DF739700 DOD Only	J. Henry Hughes Aircraft	RADC	Millimeter Wave Technology Evaluation System		Possible STS Experiment.

EXPERIMENT ASSESSMENTS

ID No. (Response Sheet or Assessment #)	Principal Investigator	Agency	Title	STS Facility (Spacelab, LDEF, etc.)	Assessment Comments
DF244780	Jules Aarons	AFGL	Global Scintillation Studies	N/A	Ground receiver studies of spacecraft telemetry signals. Could use any STS flight telemetry signals.
DF248530	Uri Shamir	AFGL	Interaction of a Spacecraft with its Environmental Space Plasma	Free Flyer or Spacelab Pallet	Current spacecraft measurements. Instrumentation could be flown either on orbiter or free flyer. Free flyer would need IUS booster.
DF248900	Kenneth G. Yates	AFGL	Direct Observations of Solar Particles Affecting the Communications Environment	Free Flyer or Spacelab Pallet	Current SOLRAD experiments. Could be flown on sortie or free flyer STS flights. Solar viewing platforms needed.
DF247630	Robert Fioretti	AFGL	Satellite Data Processing and Analysis	Free Flyer	Accelerometer instrument should be flown on low altitude STS launched free flyers.
DF249440	Joseph P. McIsaac	AFGL	Satellite Density Measurements using Ionization Gauges and Laser Sounding	Orbiter or Free Flyer	Several density measuring instruments suitable for STS Orbiter or low altitude free flyers.
DF249450	Frank A. Marcos	AFGL	Satellite Accelerometer Density Measurements	Free Flyer	Accelerometer instrument suitable for low altitude STS-launched satellites.
DF279890	Bell Aerospace Company	AFGL	Satellite Density Accelerometer	Free Flyer	Accelerometer for low altitude satellites. Suitable for STS-launched free flyers.

EXPERIMENT ASSESSMENTS

ID No. (Response Sheet or Assessment #)	Principal Investigator	Agency	Title	STS Facility (Spacelab, LDEF, etc.)	Assessment Comments
Response Sheet No. 53	Melvin Buck	AFFDL	Aerodynamics Parameter Identification Autonomous Instrumentation Packages	Orbiter OFT Flights	Flight test of STS orbiter characteristics. Must be flown on orbiter.
Response Sheet No. 54	Melvin Buck	AFFDL	Shuttle Windward Heating Measurements by Airborne IR Imagery	Orbiter OFT Flights	Flight test of STS orbiter characteristics. Must be flown on orbiter.
Response Sheet No. 55	Melvin Buck	AFFDL	Heating Amplification Factors Due to Protuberances (Ascent)	Shuttle Launch Configuration OFT Flights	Flight test of STS launch phase. Must be flown on STS launch configuration.
Response Sheet No. 56	Melvin Buck	AFFDL	Boundary-Layer Definition and Transition Criteria	Orbiter OFT Flights	Flight test of STS orbiter characteristics. Must be flown on orbiter.
Response Sheet No. 57	Melvin Buck	AFFDL	Flight Measured Aerodynamic Coefficients	Orbiter OFT Flights	Instrument to measure STS orbiter flight characteristics Must be flown on orbiter.
Response Sheet No. 30	David A. Roselius	AFFDL/FBC	Outgassing of Composite Materials	LDEF	Requires VECC; may require EPDS.
Response Sheet No. 30	David A. Roselius	AFFDL/FBC	Space Charging of Composite Materials	LDEF	May require VECC and EPDS, special fixture, plus instrumentation. May require a synchronous orbit.
Response Sheet No. 30	David A. Roselius	AFFDL/FBC	Thermal Fatigue Behavior of Composite Materials	LDEF	Experiment tray position should be chosen for maximum ΔT during each orbit.
Response Sheet No. 30	David A. Roselius	AFFDL/FBC	Long Term Radiation Effects on Composite Materials	LDEF	Experiment tray position should be chosen for maximum exposure to charged particle radiation. Orbit will determine position.

EXPERIMENT ASSESSMENTS

ID No. (Response Sheet or Assessment #)	Principal Investigator	Agency	Title	STS Facility (Spacelab, LDEF, etc.)	Assessment Comments
DF139570 Gov't Only Prelim	Darlton K. Breaux Garrett Corp. Torrance, CA	AFFDL	Preliminary Design and Analysis of a Cryogenic Turbo Refrigerator for a High Altitude Large Optics System		Possible STS Experiment.
DF137430 Gov't Only Prelim	Charles Balas North American Phillips Briarcliff Manor, NJ	AFFDL	Flight Design of an Oil Lubricated Cryo-cooler		Possible STS Experiment.
DF137020 Gov't Only	Louis Gomberg RCA Corporation Princeton, NJ	AFFDL	*Advanced Composite Precision* Mounting Platform Structure		Possible STS Experiment.
Response Sheet No. 47	Melvin Buck (Responsible Gov't Individual)	AFFDL	Body-Flap Flow Separation Phenomena	Orbiter OFT Flights	Actual test of STS orbiter performance. Must fly on orbiter.
Response Sheet No. 48	Melvin Buck	AFFDL	RSI Gap Heating	Orbiter OFT Flights	Actual test of STS orbiter performance. Must fly on orbiter.
Response Sheet No. 49	Melvin Buck	AFFDL	Flight Investigation of Leeside Flow Separation and Vortex Phenomena	Orbiter OFT Flights	Actual test of STS flight characteristics. Must be flown on orbiter.
Response Sheet No. 50	Melvin Buck	AFFDL	Aerodynamic Performance Sensing	Oribter OFT Flights	Test of STS flight characteristics. Must be flown on orbiter.
Response Sheet No. 51	Melvin Buck	AFFDL	Bow Shock/Wing Interaction	Orbiter OFT Flights	Flight test of STS orbiter characteristics. Must be flown on orbiter.
Response Sheet No. 52	Melvin Buck	AFFDL	Aerodynamic Parameter Identification Coefficient Extraction	Orbiter OFT Flights	Flight test of STS orbiter characteristics. Must be flown on orbiter.

EXPERIMENT ASSESSMENTS

ID No. (Response Sheet or Assessment #)	Principal Investigator	Agency	Title	STS Facility (Spacelab, LDEF, etc.)	Assessment Comments
DF184300	Luke Brown General Electric	AFWL	Satellite Component Sensitivity Study		Possible STS Experiment.

EXPERIMENT ASSESSMENTS

ID No. (Response Sheet or Assessment #)	Principal Investigator	Agency	Title	STS Facility (Spacelab, LDEF, etc.)	Assessment Comments
DF306567 DOD Only	L. B. Anderson MIT Lincoln Labs	AF/ESD	Space Object Surveillance and Identification		Possible STS Experiment.

EXPERIMENT ASSESSMENTS

ID No. (Response Sheet or Assessment #)	Principal Investigator	Agency	Title	STS Facility (Spacelab, LDEF, etc.)	Assessment Comments
DF344140	William J. Guman Fairchild Industries Farmingdale, NY	AFRPL	Pulsed Plume studies	Spacelab Pallet	Present laboratory study of engine plume. Testing of plume easily done on STS Orbiter flight.
Response Sheet No. 23	Gerald C. Sayles	AFRPL	Satellite Secondary Propulsion	Spacelab Pallet	Thrusters would be studied by placing them on STS Orbiter mission.

EXPERIMENT ASSESSMENTS

ID No. (Response Sheet or Assessment #)	Principal Investigator	Agency	Title	STS Facility (Spacelab, LDEF, etc.)	Assessment Comments
Response Sheet No. 14	Dr. C. Y. Ang	Aerospace	Continuous Infiltration of Metal-Matrix Fiber Composites	Molecular Wake Shield	This facility will be operational in 1980-1983 time span.
Response Sheet No. 14	Dr. C. Y. Ang	Aerospace	Orbiter Environment Effects on IR Thin Films	LDEF	Requires VECC for contamination control; may require EPDS.

EXPERIMENT ASSESSMENTS

ID No. (Response Sheet or Assessment #)	Principal Investigator	Agency	Title	STS Facility (Spacelab, LDEF, etc.)	Assessment Comments
DN794805 - NA	Applied Physics Lab	Chief Naval Ops	Navy Space Systems		Possible STS Experiment.

EXPERIMENT ASSESSMENTS

ID No. (Response Sheet or Assessment #)	Principal Investigator	Agency	Title	STS Facility (Spacelab, LDEF, etc.)	Assessment Comments
DN675180	J.B. Reagan Lockheed Missile & Space Co. Palo Alto, CA	Office of Naval Research	Communication: Plasma Interaction Experiments at Synchronous Altitude	Free Flyer	Currently planned for SCATHA satellite. Follow-ons could be launched from STS. Would need IUS booster.
DN575039	E.G. Shelley Lockheed Missile & Space Co. Palo Alto, CA	Office of Naval Research	Communications: Magneto-Ionospheric Plasma	Low Altitude 90° Inclination. Free Flyer	Continuing program. Follow-ons could be flown on a special-built STS-launched subsatellite. Non recoverable.
DN223316	R.D. Sharp Lockheed Missile & Space Co. Palo Alto, CA	Office of Naval Research	Navy Environment: Spectrometric measurement of ion masses and velocities within the Earth's magnetosphere.	Free Flyer or Shuttle	Continuing program. Follow-ons could be easily flown on several different STS flights or on an MMS free flyer.
DN624361	J.A. Van Allen Univ. of Iowa Iowa City, IA	Office of Naval Research	Communications: Solar Radiations in Near Space and Their Effects on the Earth's magnetosphere and ionosphere.	90° inclination Free Flyer	Needs low EMI levels, does particle surveys. Easily accommodated on STS-launched free flyer.
DN775381	J.T. Ely Univ. of Washington Seattle, WA	Office of Naval Research	Communications: Determining the Earth's magnetospheric state from cosmic ray anisotopies.	Free Flyer	Appears suitable for an MMS mission. Cosmic ray detector would like long term up-looking exposure.
DN775325	B. Ottar Norwegian Inst. for Air Research Norway	Office of Naval Research	Navy Environment: Arctic Haze Layers	Any	Study program. Future arctic aerosol studies could be done by lasers from an STS Orbiter flight.
DN575349	W.P. Olson MDAC Huntington Beach, CA	Office of Naval Research	Communications: Quantitative Global Model of Ionospheric Electron Density	Any	Need data to incorporate into quantitative model. Can utilize data from other STS users.

EXPERIMENT ASSESSMENTS

ID No. (Response Sheet or Assessment #)	Principal Investigator	Agency	Title	STS Facility (Spacelab, LDEF, etc.)	Assessment Comments
DN223438	T. J. Pepin Univ. of Wyoming Laramie, WY	Office of Naval Research	Stratospheric Constituants (PAM)	Free Flyer	Instrument developed in this contract can easily be flown on any low altitude, STS launched free flyer.
DN675373	H.J. Schaefer Univ. of West Florida Pensacola, Fla.	Office of Naval Research	Navy Environment: Eval. of Physical Hazards to Man Subject to Space Atmosphere	Any STS flight	Looking at human radiation dosage. All STS crew should have photo dosimeters which would be analyzed by this experiment.
DN223246	A. E. Miller Univ. of Notre Dame Indiana	Office of Naval Research	Surveillance: Magnetostrictive Behavior of Rare Earth-Transition Metal Compounds	Spacelab Pallet	Present laboratory study of magnetic alloys. Possible use of Spacelab material processing for future experiments.
DN775346	S.E. DeForest U.C. San Diego La Jolla, CA	Office of Naval Research	Communications: Low Energy Particle Detector	Free Flyer	Current SCATHA experiment. Could be flown on an STS-launched free flyer. An IUS booster would be needed.
DN675252	S.E. DeForest U.C. San Diego La Jolla, CA	Office of Naval Research	Communications: Spacecraft Charging Experiment	Free Flyer	Current experiment for SCATHA. Could easily be adapted to an STS-launch free flyer. Would require IUS booster.
DN123357	R.S. White U.C. Riverside California	Office of Naval Research	Communications: Solar Neutron Detection	Pallet or Free Flyer	Currently a balloon experiment, but easily modified to fly either an STS or a solar pointing free flyer.
DN023137	R.C. McPherron UCLA Space Science Center	Office of Naval Research	Communications: Magnetospheric Substorms	Any	Primarily data analysis contract. Could use magnetometer data from any STS or free flyer mission.

EXPERIMENT ASSESSMENTS

ID No. (Response Sheet or Assessment #)	Principal Investigator	Agency	Title	STS Facility (Spacelab, LDEF, etc.)	Assessment Comments
DN675201	E.L. Mollo-Christensen MIT Cambridge, MA	Office of Naval Research	Navy Environment: Spacecraft Oceanographic Remote Sensing	Orbiter	Present contract for study of ocean data. Future data would be generated by STS Orbiter or free flyer flights.
DN623870	R.A. Helliwell Stanford Univ. Electronics Lab.	Office of Naval Research	Communications: VLF Magnetospheric Duct Propagation and Wave Induced Precipitation	Any	Theoretical data being compared to experimental measurements. Might be able to use data from STS-flown experiments.

EXPERIMENT ASSESSMENTS

ID No. (Response Sheet or Assessment #)	Principal Investigator	Agency	Title	STS Facility (Spacelab, LDEF, etc.)	Assessment Comments
DN120126 DoD Only	S. W. Ereiman Naval Research Lab	Naval Sea Systems Command	Optical Radiation Program: Ceramics and Coatings		Possible STS Experiment
DN120129 DoD Only	J. T. Schriempf Naval Research Lab	Naval Sea Systems Command	Optical Radiation Effects: Metals		Possible STS Experiment
DN485155 DoD Only	M. T. Madden Naval Surface Weapons Center Lab	Naval Sea Systems Command	High Energy Laser Project		Possible STS Experiment
DN530834 DoD Only	A. Adicoff Naval Weapons Center	Naval Sea Systems Command	Energy Conversion, Structure and Chemistry of New Polymers		Possible STS Experiment

EXPERIMENT ASSESSMENTS

ID No. (Response Sheet or Assessment #)	Principal Investigator	Agency	Title	STS Facility (Spacelab, LDEF, etc.)	Assessment Comments
DN560166	G. H. Sigel, Jr. NRL	Naval Electronics Laboratory	Radiation Hardening of Fiber Optics	N/A	Subjects current state of art fibers to radiation. No direct STS involvement. Might be able to use materials manufactured in space.
DN639154	E. W. Peterkin NRL	Naval Electronic Systems Command	SOLRAD (Solar Activity Satellite Monitoring and Forecasting)	Free Flier	Current SOLRAD satellite program. These satellites could be launched from STS by an IUS booster.
DN480096	B. S. Yaplee NRL	Defense Mapping Agency	Satellite Radar Altimetry	Free Flier	Data Analysis of altimeter outputs. Many opportunities for STS launched altimeter experiments exist. GPS can provide very accurate ground tracks.
DN235436	R. J. Anderle Naval Surface Weapons Center	Defense Mapping Agency	Analysis of Satellite Systems and Geodetic Applications	Pallet or Free Flier	Geodesy data analysis. STS provides a good altimeter platform and can launch geodetic satellites.
DN681244	K. J. Petri Naval Air Development Center	Naval Air Systems Command	LIDAR Atmospheric Measurement Program Index	Pallet	Current ground-based lidar studies. Lidar can be placed on STS to map low altitude atmosphere. They make use of large STS power capabilities.
DN531445	F. Finger National Oceanic and Atmospheric Admin.	Naval Air Systems Command	Stratosphere Analysis for Naval Weather Operations	N/A	Current data analysis contract could make use of data from STS launched satellites.
DN590014	F. Lemkey United Research Labs E. Hartford, Conn.	Naval Air Systems Command	Development of Directionally Solidified Eutectic Nickel and Alloys	Spacelab Pallet	Best ground based alloys should be directionally solidified in space for microstructure comparison and process upgrade.

EXPERIMENT ASSESSMENTS

ID No. (Response Sheet or Assessment #)	Principal Investigator	Agency	Title	STS Facility (Spacelab, LDEF, etc.)	Assessment Comments
DN833093	M. K. Thomas NADC Air Vehicle Tech. Dept. Warminster, PA	Naval Air Systems Command	Research on Refractory D.S. Eutectic Composites T Engines	Spacelab Pallet	Preparation of samples for comparison of earth and low-G processed eutectics should improve earth processing methods.

EXPERIMENT ASSESSMENTS

ID No. (Response Sheet or Assessment #)	Principal Investigator	Agency	Title	STS Facility (Spacelab, LDEF, etc.)	Assessment Comments
DN020110	M. M. Shapiro	Naval Research Lab	The Radiation Environment and its Effects on Military Personnel and Systems	LDEF and/or Orbiter Bay	Passive heavy ion detectors can readily be accommodated on LDEF. Study of charged particles with active detectors can utilize the standard test rack in the orbiter bay.
DN920046	G. E. Brueckner	Naval Research Lab	Target Surveillance: Command Control and Communications; Forecast, Prediction of Solar Flares	Free Flier	Current development of UV instrument planned for P78-1 spacecraft. STS could launch any future spacecraft of this type.
DN920016	R. W. Kreplin	Naval Research Lab	Solar Activity Interferences with Ionosphere Dependent Military Systems	Free Flier	Solar viewing X-Ray instruments will be flown on STP satellite P78-1. Such satellites can easily be launched from STS.
Response Sheet No. 3	Stephen Knowles	Naval Research Lab	Small Far Infrared Spectrometer	Pallet or Free Flier	Instrument could be mounted on either STS pallet or on STS-launched free flier.
Response Sheet No. 1	R. G. Craddace	Naval Research Lab	SPEAR - Small Payload Ejection and Recovery	Free Flier	Specifically designed free flier for STS launch and recovery.
DN736882 DOD Only	E. N. Carey Naval Research Lab	Naval Research Lab	Ocean Surveillance Detection and Targeting Systems (Sensor Techniques)		Possible STS Experiment.
DN120101 DOD Only	C. S. Weller Naval Research Lab	Naval Research Lab	Extreme Ultraviolet Environment		Possible STS Experiment.
DN320170 DOD Only	R. R. Meier Naval Research Lab	Naval Research Lab	Dynamic Responses of the Space Environment		Possible STS Experiment.

EXPERIMENT ASSESSMENTS

ID No. (Response Sheet or Assessment #)	Principal Investigator	Agency	Title	STS Facility (Spacelab, LDEF, etc.)	Assessment Comments
DN320027	R. J. Ginther	Naval Research Lab	Synthesis and Characterization of Optical Waveguide Materials	Spacelab	Current laboratory experiment in glass fibers. Does not look directly applicable to Spacelab processing, but should be considered.
DN620096	H. Lessoff	Naval Research Lab	Naval Material Support Technology: Research on the Growth and Preparation of Crystals for Navy Device Applications	Spacelab, maybe Pallet (complex handling)	Low gravity environment and containerless methods of crystalization may improve crystal properties.
DN480098	G. G. Fritz	Naval Research Lab	X-Ray Environmental Limits to Military Systems: Soft X-Ray Astronomy	Orbiter Bay	Can utilize standard test rack and possibly pointing mount in orbiter bay. Scientific objectives probably enhanced by greater than 7+ day missions.
DN620044	R. Tousey	Naval Research Lab	Target Surveillance and Command and Control Tech: XUV Spectroscopy Solar and Atmospheric Processes Affecting Comm. and Surv.	Pallet or Free Flier	Solar pointing platform would be needed. Ideal STS sortie or free flier candidate experiment.
DN920018	D. P. McNutt	Naval Research Lab	Far IR Environmental Limits to Military Systems	Pallet or Free Flier	Current IR instrumentation development. Developed instrument would need earth pointing platforms. Well suited to STS flights.
Response Sheet #63	K. Shivanandan	Naval Research Lab	Far Infrared Sky Survey Experiment	Pallet or STR	Needs pointing mount and cryogenic cooling of sensors.

EXPERIMENT ASSESSMENTS

ID No. (Response Sheet or Assessment #)	Principal Investigator	Agency	Title	STS Facility (Spacelab, LDEF, etc.)	Assessment Comments
DN682181 DOD Only Prelim.	P. Brunner Hughes Aircraft	Naval Weapons Center	Fleet Air Defense Study		Possible STS Experiment.

EXPERIMENT ASSESSMENTS

ID No. (Response Sheet or Assessment #)	Principal Investigator	Agency	Title	STS Facility (Spacelab, LDEF, etc.)	Assessment Comments
DN785081 DOD Only	W. T. Bozicij McDonnell Douglas	Naval Surface Weapons Center	Countermeasure Study - Integrated Barrier Materials		Possible STS Experiment.
DN789029 DOD Only	J. J. Gallagher Naval Underwater Systems	Naval Surface Weapons Center	Use of Satellite and In-Situ Data to Investigate Ocean Fronts of Tactical ASW Interest		Possible STS Experiment.

EXPERIMENT ASSESSMENTS

ID No. (Response Sheet or Assessment #)	Principal Investigator	Agency	Title	STS Facility (Spacelab, LDEF, etc.)	Assessment Comments
DN678247	L. D. Cole	Naval Ship R/D Center	Boundary Layer Pressure Fluctuations of Polymers Over Rough Surfaces		Possible STS Experiment.

EXPERIMENT ASSESSMENTS

ID No. (Response Sheet or Assessment #)	Principal Investigator	Agency	Title	STS Facility (Spacelab, LDEF, etc.)	Assessment Comments
DAOC2278 Gov't Only Prelim.	L. Williamson	ECOM Atmos. Science Lab	Meteorological Satellite Evaluation System (SATCAL)		Possible STS Experiment
DAOF1752 Gov't Only Prelim.	M. D. Kays	ECOM Atmos. Science Lab	Atmospheric Waves, Their Nature and Effects on Army Operations		Possible STS Experiment
DAOD3116 Gov't Only Prelim.	Dr. R. Belt Airtron Division Litton Systems Morris Plains, NJ	ECOM SC & TA Lab	Single Crystal Growth and Laser Rod Fabrication of Neodymium Doped Yttrium Orthovanadate		Possible STS Experiment
DAOF4692	J. L. Caslavsky	DARCOM Materials and Mechanics	Growth and Characterization of Laser Materials	Spacelab Module or Pallet	Current crystal growth study. Laser crystals might be grown more easily in Spacelab zero-G environment.
DAOD9305	Dr. R. H. Williams University of Ulster Coleraine, Ireland United Kingdom	DARCOM R&S Group	Growth and Application of Cadmium Telluride	Spacelab Pallet	Current crystal growth study. Crystals may be grown more easily in Spacelab zero-G environment.
DAOE9212	J. H. Perepezko University of Wisconsin Madison, WI	DARCOM Army Research Office	Solidification of Highly Undercooled Liquid Metals and Alloys	Spacelab Pallet	This project has one of the most important potantials for space processing to date. Numerous experiments have shown anticipated benefits.
DAOE9110	C. S. Bowyer UC Berkeley	DARCOM Army Research Office	Satellite Observations of Extreme Ultra-violet Radiation	Free Flyer or Pallet	Instrument built for satellite P78-1. Follow-on could be flown on STS or STS-launched free flyer.
DAOF1195	F. W. French W. J. Schafer Assoc. Wakefield, MA	MICOM DARPA Support Office	Study and Definition of Space Laser Test Program	Spacelab Pallet	Current study contract. Future space tests of laser systems can be flown on STS using high power capability.

EXPERIMENT ASSESSMENTS

ID No. (Response Sheet or Assessment #)	Principal Investigator	Agency	Title	STS Facility (Spacelab, LDEF, etc.)	Assessment Comments
DH002792 Gov't Only	E. J. Fremouw Stanford Research Institute Menlo Park, CA	Defense Nuclear Agency	Gigahertz Measurements Operation Design		Possible STS Experiment.
[illegible]2553 Gov't Only	I. Kofsky Photometrics, Inc. Lexington, MA	Defense Nuclear Agency	Space Shuttle Experiments		Possible STS Experiment.

RESULTS AND CONCLUSIONS

RESULTS

Approximately 17,000 current research and technology Work Unit Summaries (DoD Form 1498) were screened for applicability to space experimentation. 994 of these were classified.

65 response sheets were received as a result of the presentations at DoD establishments.

From all of these, 175 were determined to have probable application to space flight experimentation using the STS. 132 of these were assessed by TRW to determine what payload carrier is appropriate for each experiment and the best method for its integration.

In the following tabulation, all of the investigations which were assessed are categorized by carrier vehicle and by type of investigation.

ASSESSMENT DATA

	ATTACHED	LDEF	FREE FLYER	MORE THAN ONE
SCIENCE & MEASUREMENTS	14 / 12	2 / 1	3 / 16	2 / 23
DEVICES	4 / 14	0 / 0	0 / 2	0 / 5
MATERIALS	3 / 21	2 / 6	0 / 0	2 / 0

LEGEND: MEDIUM LEVEL / LOW LEVEL

CARRIER:

- ATTACHED — USE OF SPACELAB, STR OR OTHERWISE MOUNTED ON OR IN THE ORBITER.
- LDEF — MAKING USE OF THE LDEF.
- FREE FLYER — REQUIRING A FREE FLYING SPACECRAFT.
- MORE THAN ONE — COULD BE PERFORMED ON MORE THAN ONE OF THE ABOVE VEHICLES.

TYPE:

- SCIENCE & MEASUREMENT — INVESTIGATIONS FOR MEASUREMENT OF BASIC PHYSICAL PHENOMENA.
- DEVICES — EXPERIMENTS FOR DEVELOPMENT AND SPACE QUALIFICATION OF SPECIFIC EQUIPMENT.
- MATERIALS — INVESTIGATIONS AIMED AT IMPROVEMENT OF MATERIALS.

These categorizations are summarized below:

STATISTICAL SUMMARY (ALL ASSESSMENTS)

BY CARRIER		
ATTACHED	68	
LDEF	11	
FREE FLYERS	21	
MORE THAN ONE	32	
BY TYPE		
SCIENCE AND MEASUREMENTS		73
DEVICE QUALIFICATION OR TEST		25
MATERIALS IMPROVEMENT		34
POSSIBLE MATERIALS PROCESSING IN SPACE (23)		
EFFECTS OF SPACE EXPOSURE ON MATERIALS (11)		

29 DoD organizations are represented in the 175 investigations listed in this report.

The investigations that were assessed at medium level were all found to be readily performable with the STS, making use of one or another payload carrying vehicle. These are summarized in the tabulation below. Of the three that must integrate directly with the Orbiter, one is small, passive, will want to fly frequently, has only a mechanical bonded interface; one has such large power and heat rejection requirements that it might not be able to work through a carrier; one has special deployment problems.

RESULTS FROM MEDIUM LEVEL ASSESSMENTS

- 3 EXPERIMENTS NEED FREE FLYERS
- 4 USE THE LDEF
- 21 REMAIN ATTACHED TO SHUTTLE
  - 16 NEED SERVICES, EITHER STR OR SPACELAB
  - 3 INTEGRATE DIRECTLY WITH ORBITER
  - 2 NEED A MULTIPURPOSE FURNACE FACILITY

Performance of these investigations, in many cases, depends on the development of specialized flight support equipment such as pointing platforms, extendable booms, and materials processing facilities. Six investigators need some type of pointing system to achieve more precise pointing or stability than the Orbiter can achieve. Three need small satellites to make ancilliary measurements in conjunction with the Orbiter mounted equipment. Five need masts or booms to deploy instruments or other equipment away from the Orbiter payload bay. In most cases, the investigations that need this flight support equipment are closely parallel to proposed NASA investigations. Much of this equipment could have joint usage once developed.

Within the work units that were assessed at low level were many investigations that appear to be quite similar, from an implementation standpoint, to one or another of those assessed at medium level. For these, there is strong confidence that they are readily performable with the STS and the payload carriers now under development. There are others that reflect data studies or phenomena modeling contracts. These are judged to need data from space flight and, should they be continued into the STS era, such data can be generated by the many flights now envisioned. Additionally, some work units are concerned with design studies for spacecraft equipment, or instruments. If these are continued, there could arise a need for flight test and qualification.

This review showed that a considerable amount of basic materials research is being performed within DoD. Many of the areas of research appear to be on subjects for which NASA's MPS program has demonstrated a strong possibility of improved materials, or better understanding of the basic processes, through space experimentation in a "zero"-gravity environment. Nevertheless, there was no evidence in the research work unit summaries that space experimentation is under active consideration. This impression was reinforced through discussion with several contacts within the DoD materials community and by the fact that no scientists known to be working for DoD were applicants on NASA's Announcement of Opportunity for materials processing investigations.

Examination of the work units suggests that DoD organizations who sponsor materials research, should investigate those areas in which "zero"-gravity experimentation could assist. Serious consideration should be given to the following:

a. The potentials of novel and unique materials breakthroughs, to be incorporated in the sensor and communications technologies of the next decade, suggest a careful and thorough exploitation should occur by the DoD sponsored R&D community.

b. DoD exploitation of low gravity processing environments is an extremely attractive opportunity based upon NASA providing the lead funding for baseline MPS capabilities.

c. It can be assured that the experiments selected for space research match DoD individual technical objectives only by DoD sponsorship of space experiments.

   These investigators could use the NASA capabilities as they evolve.

d. It may be desirable, however, to develop major MPS facilities unique to focused DoD needs as they become identified. To date, the general requirements appear to match the contemplated NASA program scope. Developing minor experiment unique flight apparatus may be necessary. This, along with sustained support of the on-going ground research projects, should be viewed as the minimum cost of participation.

e. Collaborative scientific teams should be formed to combine desirable capabilities and achieve critical effort sizes for sustained pursuit of research objectives.

CONCLUSIONS

Based on presently active research and development work units, the following conclusions can be drawn:

a. A considerable amount of traffic of DoD space flight experimentation can be projected for the STS flight era.

b. Most DoD experiments, not specifically requiring free flying spacecraft, will need to make use of one or another of the payload carriers that are being developed. Few experiments can, or should, interface directly with the Shuttle Orbiter.

c. Many experiments require flight support equipment of a specialized nature in addition to the payload carrier. However, much of such equipment is soon to be under development by NASA and could fulfill DoD requirements.

d. When an experiment has been approved for development, the investigator should be given assistance from a payload accommodation group to assist in achieving a low cost approach for its development and to improve overall mission efficiency.

e. Liaison between DoD and NASA materials science areas should be improved to assure consideration of DoD peculiar requirements for materials research and to promote potential collaboration in flight facility development.

APPENDIX A

DESCRIPTION OF THE SPACE TRANSPORTATION SYSTEM

INDEX OF APPENDIX A

DESCRIPTION OF THE SPACE TRANSPORTATION SYSTEM

Page

1.0 SHUTTLE SYSTEM

Shuttle system hardware and capabilities of importance to the user are summarized in this section. They include induced environments, and payload accommodations, such as attachments, the remote manipulator subsystem, electrical power availability, fluid and gas utilities, environmental control, communications links, data handling and displays, guidance and navigation systems, flight kits, and extra-vehicular activity provisions.

Users who expect their experiments to fly on a payload carrier (Spacelab, a propulsion stage, or a free-flying satellite) should refer to the section for that carrier. In those instances, the experiment will be integrated with the payload carrier and will not have a primary interface with the Space Shuttle Orbiter.

Detailed design and accommodations are shown in Space Shuttle System Payload Accommodations, JSC-07700, Volume XIV. Any payload modifications are the responsibility of the user.

All payloads using the Space Transportation System will be subject to a uniform set of basic safety and interface verification requirements. The verification system is designed so that the user will not need to duplicate or repeat verifications already made.

Payload Safety Requirements

The safety requirements, developed by the NASA Headquarters Office of Space Flight, will be tailored to the identified hazard potential of the payload. The Payload Safety Guidelines Handbook (JSC-11123) has been developed to assist the user in selecting design options to eliminate hazards.

The intent of the safety policy is to minimize active involvement of the STS, both at the design level and during actual flight, without compromising safety. The exact method of implementing payload safety will be negotiated between the payload supplier and the STS organization.

The STS safety policy requires that the basic payload design assure the limination or control of any hazard to the Orbiter, crew, or other payloads. The requirements are not intended to assure a high probability of mission success.

The payload supplier is responsible for assuring the safety of any hardware proposed for use in the STS. In turn, the STS organization will plan cargoes to minimize hazards created by interaction among payloads and between a payload and the STS.

Safe payload operation with a minimum dependence on the Orbiter and its crew is an STS goal. The payload supplier must identify all potentially hazardous operating sequences. Hazardous situations that require a rapid response should, if possible, be corrected by automatic systems that are part of the payload.

The STS provides a limited capability for display and subsequent command of payload parameters. Therefore, use of this capability should be limited to safety conditions that cannot or logically should not be handled by design or operational provisions. The status of safing systems and the indication of anomalous conditions occurring within a payload that do not meet these criteria should be handled in the same manner as general payload telemetry and command and control; i.e., by ground control or through the Orbiter payload station.

The same basic safety approach applied to attached payloads should also be used for those that are to be deployed and retrieved. Configuring payloads for safe retrieval should, if practical, be performed by ground control.

The payload design must preclude propagation of failures from the payload to the outside environment. In addition, safety-critical redundant subsystems must be arranged to minimize the possibility of failure of one affecting the other.

Previous manned space-flight standards for flammability, offgassing, and odor of materials have been reduced somewhat for payloads carried outside the Orbiter cabin (either in other pressurized areas or in the open cargo bay). The Orbiter cabin provides smoke detection, fire suppression, atmospheric scrubbing, and a trace gas analyzer, which mitigate the hazards from flammability and offgassing.

The major goal is to design the payload for minimum hazard by including damage control, containment, and isolation of potential hazards. Hazards that cannot be eliminated by design must be reduced as much as possible and made controllable through use of safety devices as part of the system, subsystem, or equipment.

Payload Interface Verification Requirements

Testing and interface verification of flight hardware is greatly simplified by the reuse of proven systems (Spacelab, Long Duration Exposure Facility, or Multimission Modular Spacecraft) or by the flight of identical expendable items (interim upper stage and spinning solid upper stage). The users will not need to repeat the verification process that the standard flight systems must undergo as part of their development. This will significantly reduce the time and cost of interface verification for the user.

The payload accommodation interfaces for the Space Shuttle system have been defined in the document, Space Shuttle System Payload Accommodations (JSC-07700, Volume XIV). Interface verification requirements are defined in Space Shuttle System Payload Interface Vericiation - General Approach and Requirements (JSC-07700-14-PIV-01). The latter document requires that new hardware projects have a verification program planned to assure that the necessary verification requirements of the respective interfaces are met before the payload is installed in the Orbiter.

Users of the standard payload carriers will assess their payload to determine if new or unique configurations require verification before flight. This assessment and necessary verification will be accomplished in conjunction with the STS operations organization.

Few or no additional verification requirements are anticipated for payloads that are reflown; however, some assessment of the payload should be made to assure that configuration changes to the payload or cargo do not create a new interface that would require preflight verification.

The term "payload" describes any item provided by the user having a direct physical or functional interface with the Space Shuttle system.

Payload verification plans shall be submitted to JSC for review and concurrence of the verification methods for safety-critical interfaces. When necessary, the verification methods for the safety-critical interfaces will be negotiated with the responsible payload organization to achieve an acceptable verification that will ensure a safe system. These safety-critical interface verification methods shall be subject to appropriate management control within the Space Transportation System.

A verification plan should contain the following information:

1. Scope.
2. Applicable documents.
3. Interface verification requirements and methods matrix, identifying specific direct (physical or functional) payload interfaces with the Orbiter and defining the verification method (test, demonstration, etc.) for each specific interface.
4. Safety-critical interface verification method synopsis.
5. Verification requirement waivers (these must be negotiated with JSC).
6. Verification requirement deferrals (i.e., deferral until flight, etc.). These deferrals will have to be negotiated in the same way as waivers.
7. Schedule for plan submittal and required approval date.
8. Schedule for payload interface verification testing program and specific dates for safety-critical interface verification tests.

Equipment suitable for interface verification testing is available at the launch site. The cargo integration test equipment (CITE) at KSC is capable of both payload-to-payload interface testing for mixed cargoes and cargo-to-Orbiter testing. The CITE simulates the Orbiter side of the interface in form, fit, and function.

At the completion of the interface verification process, but before the payload is installed in the Orbiter, a certificate of compliance confirming interface compatibility shall be prepared by the using payload organization and submitted to the Shuttle system organization. The certificate of compliance documentation shall include all interface verification requirement waivers, noncompliances, and deferrals; this documentation will become a permanent part of the payload data package.

1.1 Performance Capability

Operational flights will be launched from the NASA/KSC beginning in 1980. Orbital inclinations between 28.5° and 57° can be obtained with a maximum cargo weight of 65,000 pounds.

Operational flights from Vandenberg Air Force Base in California will begin in 1982. Higher orbital inclinations (56° to 104°) than from KSC can be obtained. Shuttle cargo weight capability decreases rapidly as the inclinations increase.

Two upper stage systems are currently planned. A solid propellant spin-stabilized stage called the spinning solid upper stage (SSUS) is designed for Atlas-Centaur and Delta classes of missions. The solid propellant three-axis-stabilized interim upper stage (IUS) is intended for boosting single or multiple spacecraft to higher orbits and escape trajectories.

The expendable upper stage is a reliable, simple, low-cost vehicle for spacecraft missions with altitudes, inclinations, and trajectories beyond the basic Shuttle capability. The upper stage systems consist of one or more solid-propellant propulsive stages, airborne support equipment, ground support equipment, software, and unique facilities.

Detailed definition of the Shuttle system's performance capability for circular and elliptic orbits of varying altitudes is included in:

(1) Space Shuttle Systems Payload Accommodations, JSC-07700, Volume XIV.

(2) Space Transportation System, User Handbook.

1.2 Induced Environments

Payload environments will vary for specific missions and will also depend on the spacecraft involved (type of free-flying system or Spacelab configuration, for example). Therefore, data in this section are general in nature. The figures represent recommended design qualification test levels.

Acoustic Vibration

The estimated random vibration and appropriate exposure durations for the cabin and midfuselage to payload interfaces caused by the fluctuating pressure loads are shown in Figure A.1.1. The levels shown are typical of liftoff, transonic flight, and performance at maximum aerodynamic pressure. The midfuselage/payload interface vibration environment is based on the response of unloaded interface structure and should be considered the upper limit. The vibration inputs at the interface will be reduced by addition of the payload and support structures between the interface and payload component.

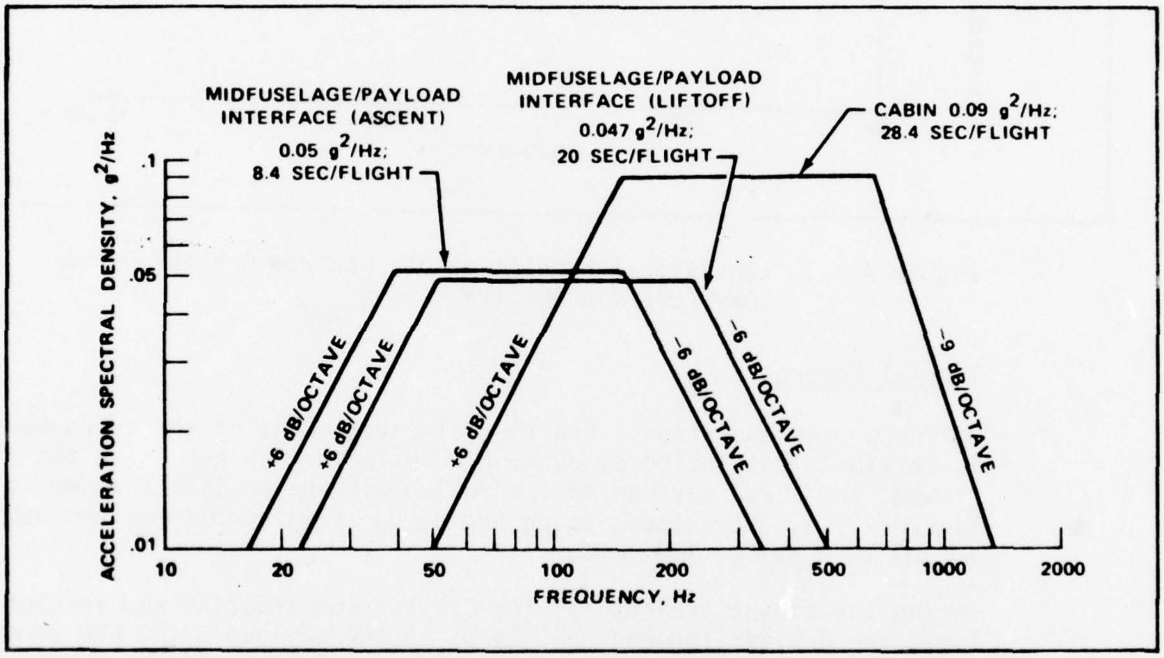


Figure A.1.1. Random Vibration at Midfuselage Main Longeron Payload Attachment Points Interface and in the Cabin. These Levels Are Typical of Liftoff, Transonic Flight, and Flight at Maximum Aerodynamic Pressure

Vibration resulting from acoustic spectra is generated in the cargo bay by the engine exhaust and by aerodynamic noise during atmospheric flight. These predicted maximums are illustrated in Figure A.1.2. The data presented are based on an empty cargo bay and may be modified by the addition of payloads, depending on their characteristics. Aerodynamic noise during entry is significantly less than on ascent.

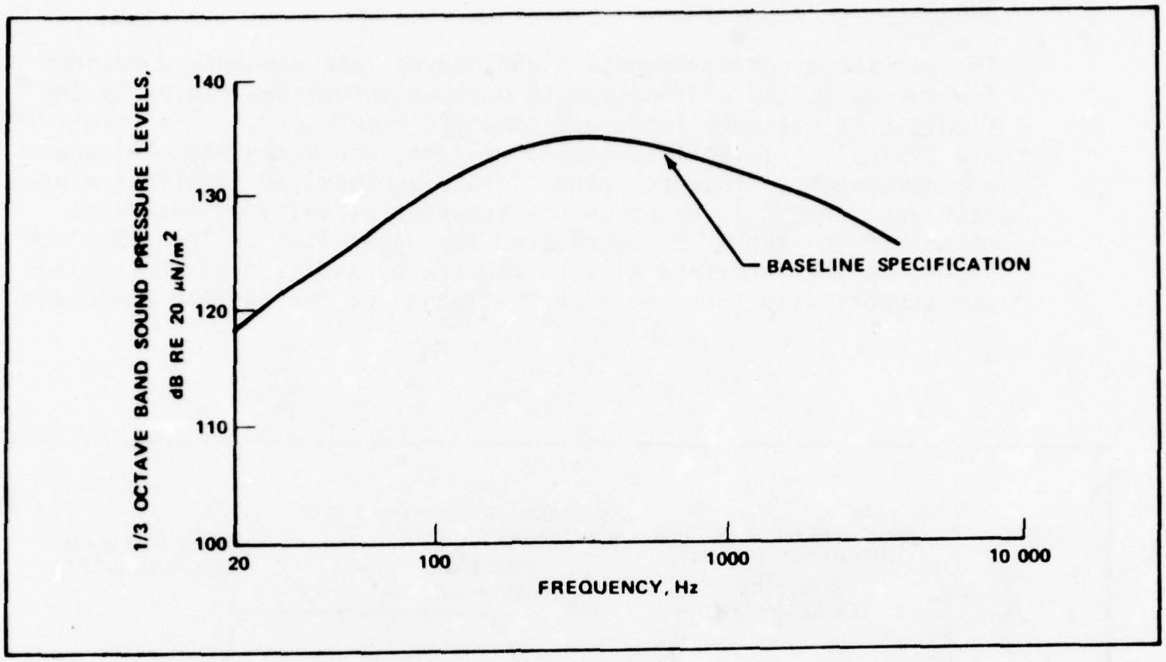


Figure A.1.2. Analytical Prediction of Maximum Orbiter Cargo Bay Acoustic Spectra

Thermal Control

During ground operations, the thermal environment of the cargo bay is carefully controlled by purging. While the Orbiter is on the ground, the cargo bay can be controlled within the limits shown in Figure A.1.3. Air-conditioning and purge requirements are defined by analysis for each launch.

During the ascent trajectory, the Orbiter construction and insulation limit the Orbiter induced heat loads on the payload. At 600 seconds after launch, the Orbiter is in the on-orbit phase and the cargo bay doors can be opened.

In space, with the cargo bay doors open, heating of payload components is based on the thermal, thermophysical, and geometric characteristics of each component. Additional factors influencing the incident thermal environment are launch date and hour, vehicle orientation, and orbital attitude. A detailed analysis of each payload may be necessary before thermal design and integration. For preliminary calculations, the optical properties of the cargo bay liner, Orbiter radiators, and insulated forward and aft bulkhead surfaces are as follows, where α is absorption and ϵ is emissivity.

Cargo bay liner	$\alpha/\epsilon \leq 0.4$
Radiator surface	$\alpha/\epsilon = 0.10/0.76$
Forward and aft bulkheads	$\alpha/\epsilon \leq 0.4$

Cargo temperatures for a typical flight, with emphasis on the entry phase, are shown in the Figure A.1.3.

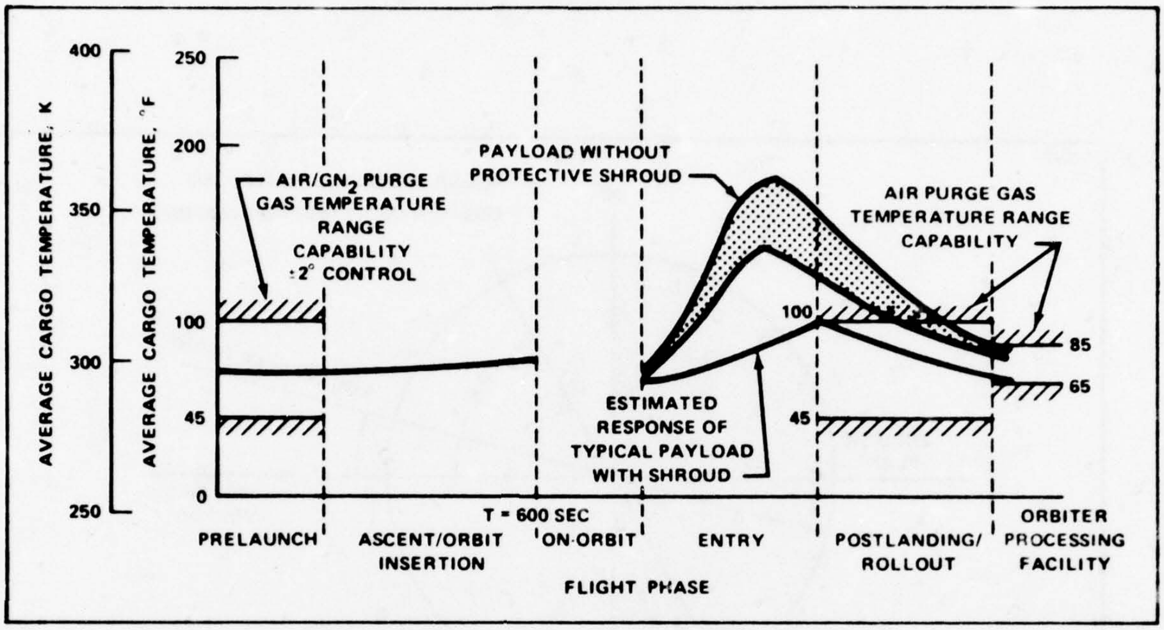
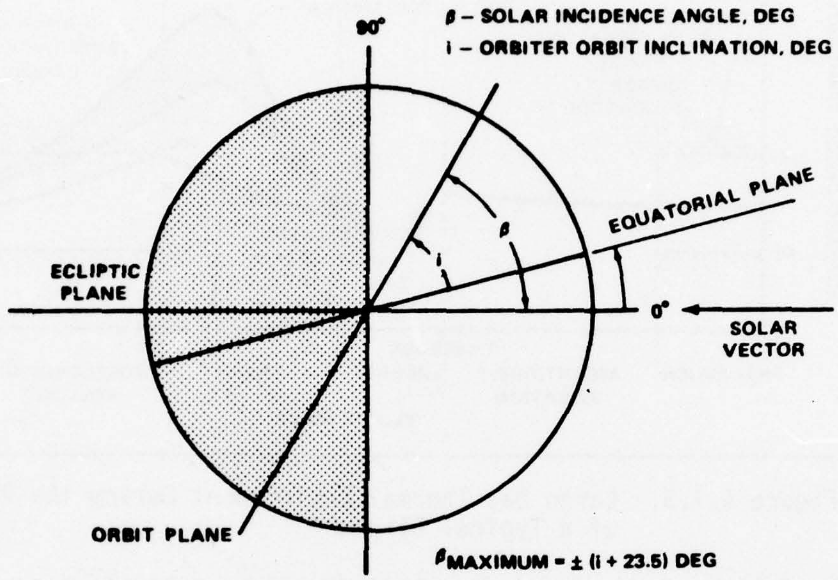


Figure A.1.3. Cargo Bay Thermal Environment During the Phases of a Typical Flight

The Orbiter is designed for attitude hold capabilities as shown in Figure A.1.4. During the 3-hour thermal conditioning periods, the vehicle rolls at approximately five resolutions per hour (barbecue mode) about the X-axis with the orientation of the X-axis perpendicular to the Earth-Sun line within $\pm 20^{\circ}$, or it can be oriented at preferred thermal attitudes. On-orbit thermal conditioning lasting as long as 12 hours (before the deorbit maneuver) is allocated for missions on which the thermal protection subsystem temperatures exceed the design limits associated with a single-orbit mission.

Payload Limit Load Factors

Payload structure and substructure must be designed with the appropriate margin of safety to function during all expected loading conditions, both in flight and during ground handling. The limit load factors at the payload center of gravity are shown in Table A.1.1. The recommended margin of safety to apply to these limit load factors is 1.4. Emergency landing loads shall be carried through the payload primary structure at its attachment fittings. Preliminary design criteria for emergency landing conditions (ultimate design accelerations) for linear g are, along the X-axis, +4.5 to −1.50; along the Y-axis, +1.5 to -1.5; and along the Z-axis, +4.5 to −2.0.



β range, deg	Orbiter orientation	Hold capability, hr	Preentry thermal conditioning requirements, hr
0 to 60	Any	>160	<12
60 to 90	Other than 3-axis inertial holds	Cycles of 6-hour holds followed by 3 hours of thermal conditioning for worst thermal attitudes	<7
	3-axis inertial holds	>160	<12

Figure A.1.4. Orbiter Attitude Hold Capabilities for Various Vehicle Orientations

The emergency landing design accelerations are considered ultimate; therefore, a 1.0 margin of safety should be applied.

Table A.1.1. Limit Load Factors[a]

Condition	Load factor		
	X-axis	Y-axis	Z-axis
Liftoff	-0.1	1.0	[b]1.5
	-2.9	-1.0	[b]-1.5
Booster staging	-2.7	.2	-0.3
	-3.3	-0.2	-0.3
Entry	1.06	1.25	2.5
	-0.02	-1.25	-1.0
Landing	1.0	0.5	[b]2.8
	-0.8	-0.5	[b]2.2

[a]For 65 000 lb (29 484 kg) up and 32 000 lb (14 515 kg) down.

[b]Angular accelerations of 10 rad/sec[2] applied from front cradle support to free end of spacecraft.

i.e.

$$N_Z = +2.75 + \frac{10\Delta X}{386},\ N_Z = -2.75 - \frac{10\Delta X}{386}$$

Landing Shock

Landing shock is another factor that must be considered in payload structure design. Rectangular pulses of the following peak accelerations will be experienced.

Acceleration, g Peak	Duration Msec	Applications per 100 Flights
0.23	170	22
.28	280	37
.35	330	32
.43	360	20
.56	350	9
.72	320	4
1.50	260	1
		125

Consideration should be given to analyzing the landing shock environment in lieu of testing, because the g levels are relatively low in comparison to the basic design shock.

Testing must be performed only on those items not covered in a static structural stress analysis.

Pressure and Venting

With the vents open, the cargo bay pressure closely follows the flight atmospheric pressures. The payload vent sequencing is as follows.

Prelaunch	Closed (vent No. 6 in purge position)
Liftoff (T = 0)	Closed
T + 10 seconds	All open
Orbit insertion	All open
On orbit	All open
Preentry preparation	All closed
Entry (high heat zone)	All closed
Atmospheric (75,000 ±5000 feet) (23 ± 1.5 kilometers)) to landing	All open
Postlanding purge	Closed (vent No. 6 in purge position)

During the orbital phase the cargo bay operates unpressurized. Pressures for other flight phases are shown in Figure A.1.5.

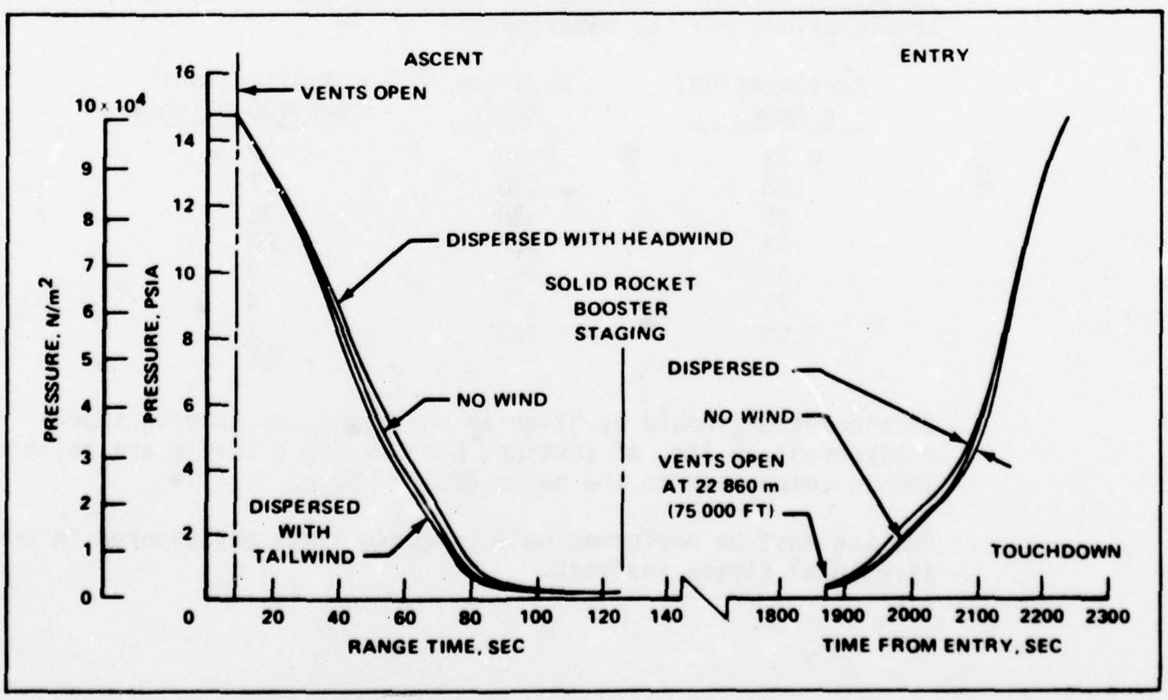


Figure A.1.5. Cargo Bay Internal Pressure

Contamination Control

Before liftoff and after landing, the cargo bay is purged and conditioned as specified in the description of thermal controls. At launch and during early ascent, the cargo bay vents are left closed to prevent exhaust products and debris from entering the bay. During final ascent and through orbit insertion, the cargo bay is depressurized and the payload is generally not subjected to contaminants.

On orbit there are three major sources of contamination: reaction control subsystem vernier firings, dumping of potable water, and release of particulates and outgassing. Predicted column density and return flux contributions are shown in Table A.1.2.

Table A.1.2. Predicted Column Density and Return Flux

Source	Number column density, molecules/cm^2			Return flux, molecules/cm^2/sec		
Outgassing (a) and (b)	$< 10^{12}$ after 10 hr			$< 10^{12}$		
				Values at 253 n. mi. (435 km)		
Vernier RCS	Aft -Z	Aft Y	Forward X/Z	Aft -Z	Aft Y	Forward Y/Z
(a)	4.4×10^{14}	2.0×10^{14}	3.9×10^{12}	7.6×10^{12}	3.4×10^{12}	6.6×10^{10}
(b)	1.8×10^{14}	8.1×10^{13}	2.7×10^{12}	3.2×10^{12}	1.4×10^{12}	4.6×10^{12}
Flash evaporator				378 n.mi. (700 km)	235 n. mi. (435 km)	108 n. mi. (200 km)
(a)		5.6×10^{12}		8.4×10^{8}	2.4×10^{12}	1.3×10^{12}
(b)		5.6×10^{12}		8.5×10^{8}	2.4×10^{10}	1.3×10^{12}
Leakage						
(a)		2.2×10^{13}		1.2×10^{10}	3.7×10^{11}	1.9×10^{13}
(b)		3.5×10^{13}		2.0×10^{10}	5.6×10^{11}	3.1×10^{13}

[a] Zero degree line-of-sight (in the $+Z_o$ direction) originating at X_o 1107.

[b] 50° off of +Z towards $-X_o$ (forward) originating at X_o 1107.

During deorbit and descent, the cargo bay vents are closed to minimize ingestion of contaminants created by the Orbiter systems. During the final phase of reentry, the vents must be opened to repressurize the Orbiter. To help prevent contamination during this phase, the vents are located where the possibility of ingestion is minimal.

Electromagnetic Compatibility

In general, close adherence to accepted electromagnetic compatibility design requirements will ensure compatibility of payloads with the Orbiter. The payload-generated -conducted, and -radiated emissions are limited to the levels specified in Space Shuttle Systems Payload Accommodations.

The magnetic fields (applied at a distance of 1 meter) generated shall not exceed 130 decibels above 1 picotesla (30 to 2 hertz), falling 40 decibels per decade to 50 kolohertz. The dc field shall not exceed 160 decibels above 1 picotesla.

For payload equipment in the cargo bay, broadband emissions shall be limited to 70 decibels above 1 μV/m/MHz in the frequency range of 1770 to 2330 megahertz; narrowband emissions shall be limited to 25 decibels above 1 μV/m from 1770 to 2300 megahertz, excluding the payload-intentional transmissions.

Electrostatic discharges are not permitted within the cargo bay unless they are contained and shielded by the payload.

Payload-generated power by single event switching or operations occurring less than once per second shall not generate transients 300 x 10^{-6} volt-seconds above or below normal line voltage when fed from a source impedance. Peaks shall be limited to $\pm$50 volts, and rise and fall times shall not be less than 1 microsecond.

1.2 Payload Accommodations

The Orbiter systems are designed to support a variety of payloads and payload functions. The payload and mission stations on the flight deck provide space for payload-provided command and control equipment for payload operations required by the user. Remote control techniques can be managed from the ground when desirable. When used, the Spacelab provides additional command and data management capability plus additional pressurized work area for the payload specialists. The following supporting subsystems are provided for payloads.

- Payload attachments
- Remote manipulator handling system
- Electrical power, fluids, and gas utilities
- Environmental control
- Communications, data handling, and displays
- Guidance and navigation
- Flight kits
- Extravehicular activity (EVA) capability when required.

Payload accommodations are described in detail in the document Space Shutle System Payload Accommodations (JSC 07700, Volume XIV).

All payloads have one or a combination of interfaces with the Orbiter vehicle. The vehicle is designed to provide adequate standard interfaces that can be used by or adapted to most potential payloads. Basic types of interfaces are summarized in Figure A.1.6. Additional support systems and flight kits are also available to accommodate payloads.

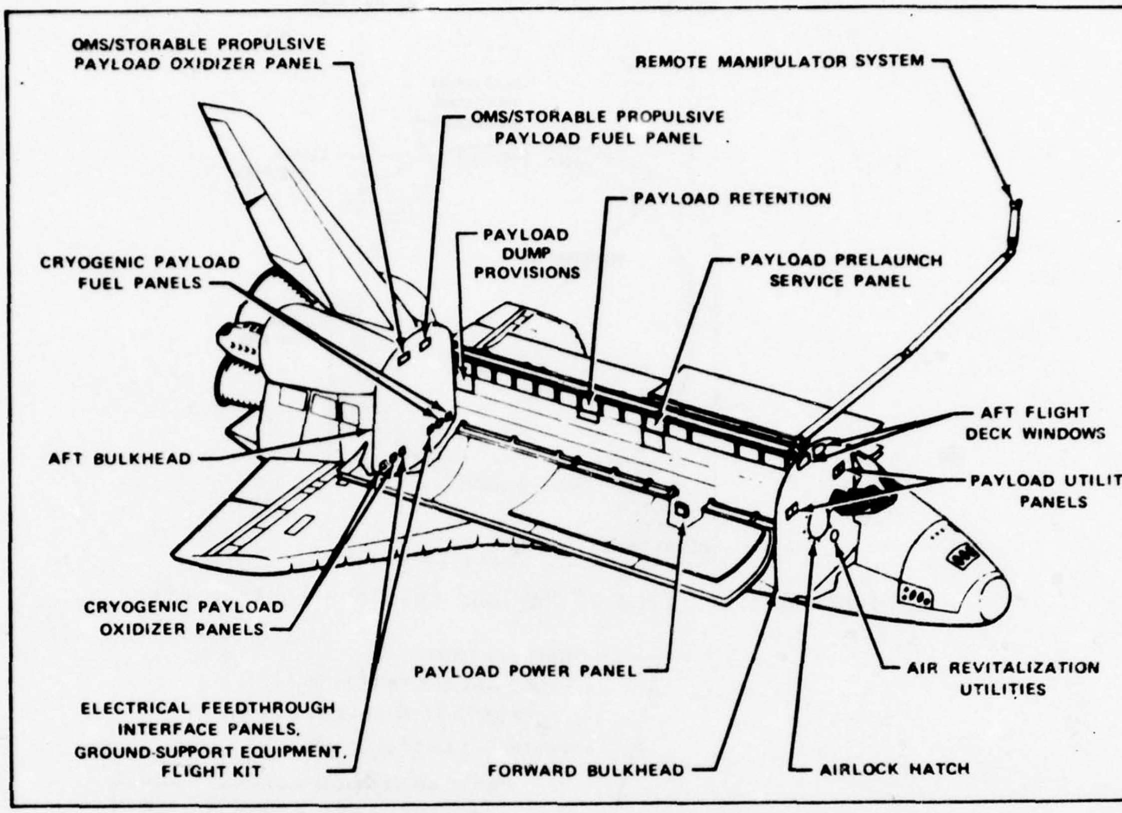


Figure A.1.6. Principal Orbiter Interfaces with Payloads

Envelope Available to Payload

Payload accommodations are provided in two general areas of the Orbiter: the cargo bay and the aft flight deck in the cabin.

The payload clearance envelope in the Orbiter cargo bay measures 15 by 60 feet (4572 by 18,288 millimeters). This volume is the maximum allowable payload dynamic envelope, including payload deflections. In addition, a nominal 3-inch (76-millimeter) clearance between the payload envelope and the Orbiter structure is provided to prevent Orbiter deflection interference between the Orbiter and the payload envelope. The payload envelope is shown in Figure A.1.7.

The payload space on the aft flight deck is intended primarily for control panels and storage. The available area is shown in Figure A.1.8.

Communications, Tracking, and Data Management

Voice, television, and data-handling capabilities of the Orbiter support onboard control of the payload or, when desirable, remote control from the ground. The Orbiter communications and tracking subsystem provides links between the Orbiter and the payload. It also transfers payload telemetry, uplink data commands, and voice

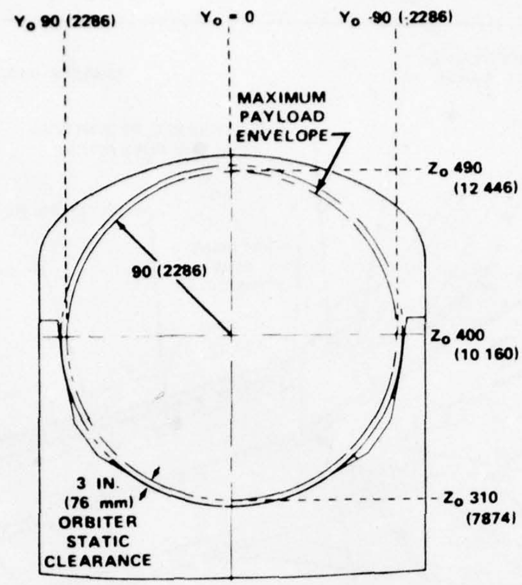


Figure A.1.7. View of Payload Envelope Looking Aft

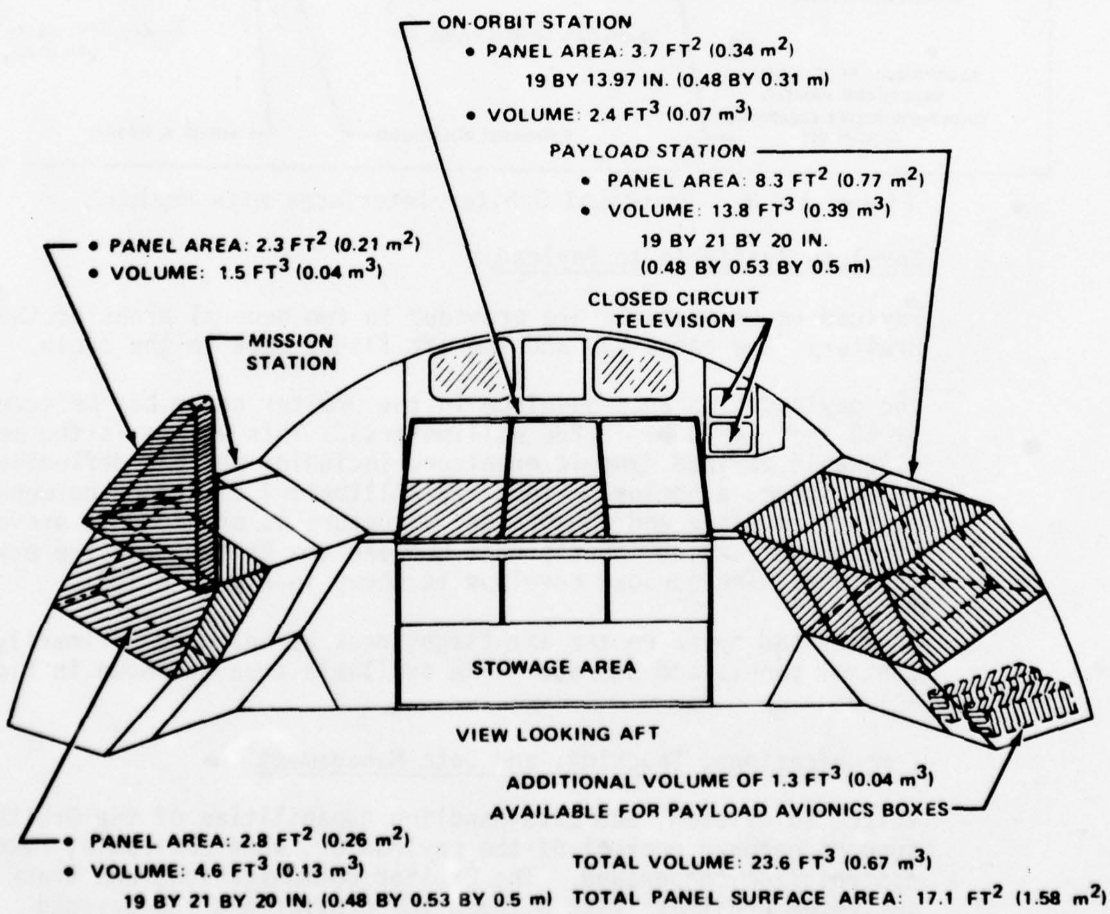


Figure A.1.8. Area Available for Payload Equipment or Controls in the Orbiter Aft Flight Deck.

signals to and from the space networks. The provisions in the Orbiter for communications, tracking, and data management are flexible enough to accommodate most payloads. Links through the Orbiter are outlined in Table A.1.3.

Table A.1.3. Orbiter Avionics Services to Payloads

Function	Direct or through Tracking and Data Relay Satellite		Hardline		Radiofrequency link	
	Payload to ground via Orbiter	Ground to Payload via Orbiter	Orbiter to attached payload	Attached payload to Orbiter	Orbiter to detached payload	Detached payload to Orbiter
Scientific data	X	X		X		
Engineering data	X	X		X		X
Voice	X	X	X	X	X	X
Television	X		X	X		
Command		X	X		X	
Guidance, navigation, and control		X	X	X	X	
Caution and warning	X			X		X
Master timing			X			
Rendezvous					X	X

The data processing and software subsystem of the Orbiter furnishes the onboard digital computation necessary to support payload management and handling. Functions in the computer are controlled by the mission specialist or a payload specialist through main memory loads from the tape memory. The stations in the Orbiter aft flight deck for payload management and handling are equipped with data displays, CRT's, and keyboards for onboard monitoring and control of payload operations.

Weight and Center of Gravity

The location of the cargo center of gravity is critical during aerodynamic flight. Weight and center-of-gravity calculations must take into account all items of supporting subsystems charged to the payload. Cargo center-of-gravity envelopes for each axis of the Orbiter are shown in JSC 07700 Volume XIV. During normal landings and abort operations the center of gravity must fall within these envelopes. Out-of-envelope conditions are permissible during launch and space flight. However, the conditions must be correctable before reentry or in the event of an abort on launch.

Deployment and Retrieval

The deployment and retrieval of payloads will be accomplished by use of the general purpose remote manipulator system. Deployment can also be accomplished with a tilt/spin table.

One manipulator arm is standard equipment on the Orbiter and can be mounted on either the left or the right longeron. A second arm can be installed and controlled separately for payloads that require handling with two manipulators. Manipulators cannot be operated simultaneously. However, the capability exists to hold or lock one arm while operating the other. Each arm is associated with remotely controlled television cameras and lights to provide side viewing and depth perception. Lights on booms and side bulkheads provide sufficient illumination levels for any task that must be performed in the cargo bay. Payload retrieval involves the combined operations of rendezvous, stationkeeping, and manipulator arm control.

Structural Interfaces

Numerous attachment points along the sides and bottom of the cargo bay provide structural interfaces in a multitude of combinations to accommodate payloads. Thirteen primary attachment points along the sides accept X- and Z-axis loads. Twelve positions along the keel take lateral loads. Vernier locations are provided on each bridge fitting.

The fittings are designed to be adjusted to specific payload weight, volume, and center-of-gravity distributions in the bay. The fittings to attach payloads to the bridge fittings are standardized to minimize payload changeout operations. To further minimize payload operations involving the Orbiter, standard payload handling interfaces have been provided.

Electrical Interfaces

Electrical power is provided to the payload from three fuel cells that use cryogenically stored hydrogen and oxygen reactants. The electrical power requirements of a payload during a flight will vary. During the 10-minute launch-to-orbit and the 30-minute deorbit-to-landing phases (when most of the experiment hardware is on standby or turned off), 1000 watts average to 1500 watts peak are available from the Orbiter. In orbit, as much as 7000 watts average to 12,000 watts peak can be provided to the payload.

For the usual 7-day flight, 50 kWh (180 megajoules) of electrical energy are available to payloads. If more energy is needed, flight kits can be added as required by the flight plan. Each kit contains enough consumables to provide 840 kWh (3024 megajoules). These are charged to the payload mass and volume.

Each of three fuel cell powerplants provides 2 kilowatts minimum and 7 kilowatts continuous, with a 12-kilowatt peak of 15 minutes duration every 3 hours.

Environmental Control

Cooling services are provided to payloads by the Orbiter. Prelaunch and postlanding thermal control is provided by ground support systems. In orbit, the primary Orbiter heat rejection is by use of radiators on the cargo bay doors. The payload heat exchanger is designed so either water or Freon 21 can be selected as a cooling fluid, according to the needs of the payload. The payload side of the heat exchanger has two coolant passages; either or both can be used. Coolant is provided to the payload at 40° to 45° F (278 to 280 K). Fluid circulation through the payload side of the heat exchanger must be supplied as part of the payload. A water flash evaporator is used to supplement the radiator cooling capacity. During ascent and descent, when the cargo bay doors are closed and the radiators are ineffective, cooling is provided by the water boilers. A summary of the payload heat rejection available is shown in Table A.1.4.

Table A.1.4. Payload Heat rejection Available

Flight phases	Capability, KW		Description
	Aft flight deck	Cargo bay	
Prelaunch, ascent, descent, postlanding (cargo bay doors closed)	.35 .42	1.52 NA	Average 2-min peak
On orbit without radiator kit (cargo bay doors open)	.75 1.00 .35	5.90 5.65 6.30	Average Peak for 15 min each 3 hr Minimum for aft flight deck, maximum for cargo bay
On orbit with radiator kit (cargo bay doors open)	.75 1.00 .35	8.10 7.85 8.50	Average Peak for 15 min each 3 hr Minimum for aft flight deck, maximum for cargo bay

Extravehicular Activity

Capability for extravehicular activity (EVA) is available on every Space Shuttle flight.

Payload EVA falls into three categories: planned before launch in order to complete a mission objective; unscheduled but decided upon during a flight in order to achieve payload operation success or advance overall mission accomplishments; or EVA involving contingency measures necessary to get any payload items out of the way of the cargo bay doors.

Equipment and consumables required for unscheduled and contingency EVA's are included on every Orbiter flight. Planned payload EVA is a user option.

Planned EVA can provide sensible, reliable, and cost-effective servicing operations for payloads. It gives the user the options of orbital equipment maintenance, repair, or replacement without the need to return the payload to Earth or, in the worst case, to abandon it in space. Therefore, the EVA capability can help maximize scientific return.

All EVA operations will be developed using the capabilities, requirements, definitions, and specifications set forth in Shuttle EVA Description and Design Criteria (JSC-10615).

Standard tools, tethers, restraints, and portable workstations for EVA are part of the Orbiter baseline support equipment inventory. The user is encouraged to make use of standard EVA support hardware whenever possible to minimize crew training, operational requirements, and cost. Any payload-unique tools or equipment must be furnished by the user.

Crewmembers using extravehicular mobility units (spacesuits and life support systems) can perform the following typical tasks.

- Inspection, photography, and possible manual override of vehicle and payload systems, mechanisms, and components
- Installation, removal, or transfer of film cassettes, material samples, protective covers, instrumentation, and launch or entry tiedowns
- Operation of equipment, including tools, cameras, and cleaning devices
- Cleaning of optical surfaces
- Connection, disconnection, and storage of fluid and electrical umbilicals
- Repair, replacement, calibration, and inspection of modular equipment and instrumentation of the spacecraft or payloads
- Deployment, retraction, and repositioning of antennas, booms, and solar panels
- Attachment and release of crew and equipment restraints
- Performance of experiments
- Cargo transfer.

2.0 SPACELAB

Spacelab is a versatile, general-purpose orbiting laboratory for manned and automated activities in near-Earth orbit. The primary program objective is to provide the scientific community with easy, economical access to space. Involvement of ground-based scientific personnel in direct planning and flight support is an integral part of this program.

The Spacelab, built in Europe with European funds to joint U.S. and European requirements, is carried by the Space Shuttle and remains attached to the Orbiter during all phases of the mission. The overall physical characteristics of most importance to users of the Spacelab are summarized here. All accommodations are described in more detail in the Spacelab Payload Accommodations Handbook (European Space Agency SLP/2104).

Services to the Spacelab payload in excess of those included in the Orbiter and Spacelab baselines are available as user options as needed. Appropriate care must be taken to ensure that all items are included in payload flight planning.

The Spacelab consists of module and pallet sections used in various configurations to suit the needs of a particular mission. The pressurized module (Figure A.2.1) accessible from the Orbiter cabin through a transfer tunnel, provides a shirtsleeve working environment. The module consists of one or two cylindrical segments, each 13 feet 4 inches (4060 millimeters) in diameter and 8 feet 7 inches (2694 millimeters) long, and two end cones. The forward end cone is truncated at the diameter required to interface with the crew transfer tunnel. Spacelab subsystem equipment is located in the core segment, leaving about 60 percent of the volume available for experiments; all of the experiment segment is available for experiments.

Pallets accommodate experiment equipment for direct exposure to space. Each standard pallet segment is 9.8 feet (3 meters) long. Two or three can be connected to form a single pallet train, supported by one set of retention fittings. When no module is used, a cylindrical "igloo," mounted on the end of the forward pallet, provides a controlled, pressurized environment for Spacelab subsystems normally carried in the core segment. The pallet configuration is shown in Figure A.2.2.

In addition to the basic hardware inventory, the Spacelab Program provides a selection of mission-dependent equipment that can be flown according to the requirements of a particular mission.

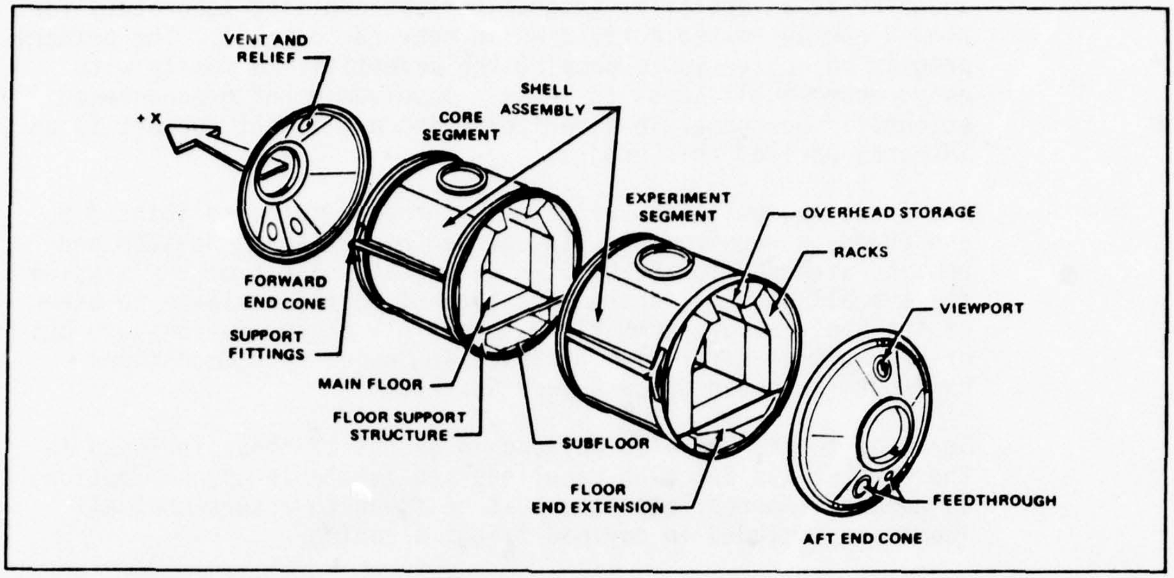


Figure A.2.1 Overall Configuration of the Module Showing Both the Core and Experiment Segments

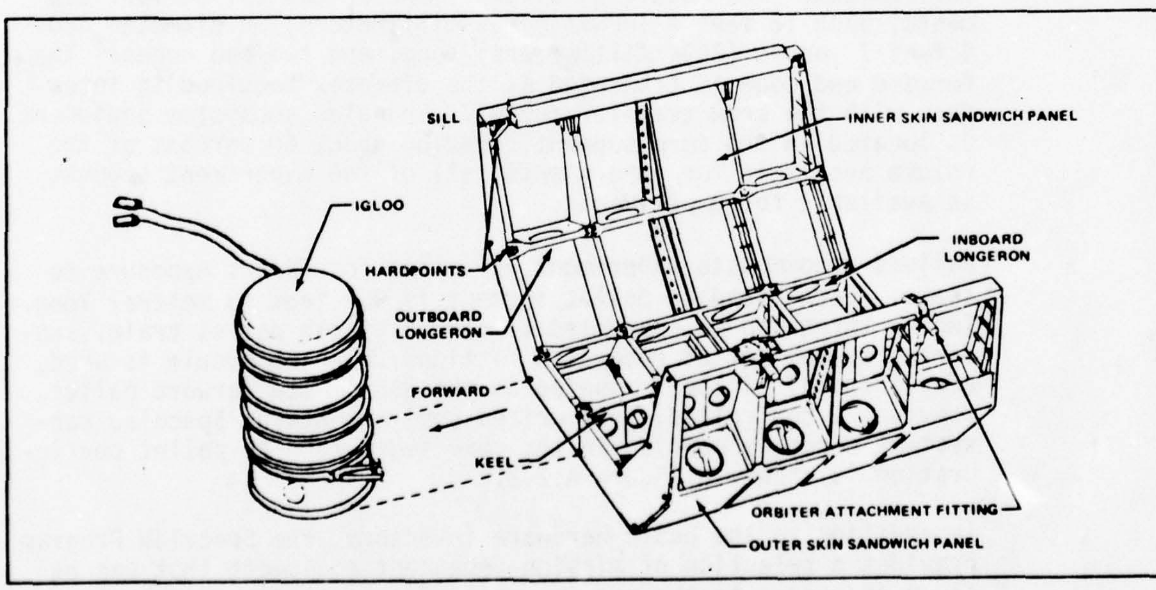


Figure A.2.2 Pallet Segment and Igloo

When the module is used, primary control of scientific equipment will be from the module itself. A Payload Operations Control Center (POCC) on the ground will function in a support and advisory capacity to onboard activity. In a pallet-only configuration, equipment is operated remotely from the Orbiter aft flight deck or from the POCC.

2.1 Basic Configurations

Eight basic flight configurations have been designed to meet most user needs. (The Spacelab hardware, however, allows other flight configurations by combining appropriate hardware elements.) The standard configurations are as follows:

(1) Long Module

(2) Long Module/One Pallet

(3) Long Module/Two Pallets

(4) Short Module/Two Pallets

(5) Short Module/Three Pallets

(6) Three Pallets/Independently Suspended

(7) Two Pallets Plus Two Pallets

(8) Five Pallets

Payload Mass and Center of Gravity

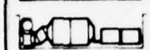
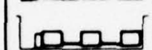
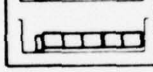
A wide range of payload mass capabilities exists. Maximums depend on configurations, mission-dependent Spacelab equipment, and other factors. The actual mass available to payloads for any given configuration of Spacelab and Orbiter hardware will be limited by the launch/landing mission capabilities of the Shuttle and the specific load-carrying capabilities of Spacelab. The information provided here will help users to determine the mass available for the total payload to meet specific mission needs.

Each of the many possible configurations will have a different total mass. The control masses listed in Table A.2.1 represent the maximum mission-dependent equipment that can be flown in each configuration.

The total mass available for both payloads and mission-dependent equipment is listed for each configuration. However, actual mass capability is further limited by structural limitations of various components. Additional localized constraints exist, such as mass-supporting capabilities of racks and hardpoints.

The Orbiter imposes center of gravity constraints on the Spacelab and its payloads. Refer to the Spacelab Payload Accommodations Handbook for detailed data for the standard configurations.

Table A.2.1. Mass Allocation to Spacelab and Payloads

Configuration	Total mission-independent Spacelab mass, lb (kg)	Mass of 100 percent mission-dependent equipment, lb (kg)	Mass available to payload and mission-dependent equipment, lb (kg)
	14 211 (6446)	2646 (1200)	14 066 (6380)
	16 182 (7340)	2976 (1350)	13 614 (6175)
	17 031 (7725)	3174 (1440)	12 765 (5790)
	14 941 (6777)	2006 (910)	14 414 (6538)
	15 790 (7162)	2116 (960)	13 717 (6222)
	8 369 (3796)	1164 (528)	20 613 (9350)
	9 129 (4141)	1190 (540)	19 564 (8874)
	10 357 (4698)	1301 (590)	18 263 (8284)

2.2 Module Segments

Modules for all flight configurations contain the same basic internal arrangement of subsystem equipment; the main difference is the volume available for experiment equipment installation. Although subsystem equipment is located in the core segment, about 60 percent of the volume is available for experiments.

The interior design (Figure A.2.3) provides flexibility to the user. The floor, designed to carry racks with installed equipment, is in segments. The floor itself consists of a load-carrying beam structure and is covered by panels on the main walking surface. Except for the center floor plates, the panels are hinged to allow underfloor access, both in orbit and on the ground.

Mission-dependent experiment racks are available for experiments, experiment switching panels, remote acquisition units, intercom stations, and similar equipment. The standard 19-inch (483-millimeter) racks can accommodate laboratory equipment. As many as two double and two single racks can be installed in the core segment, four double and two single in the experiment segment. If experiment racks are replaced by stand-alone experiment equipment, the same attachment points must be used.

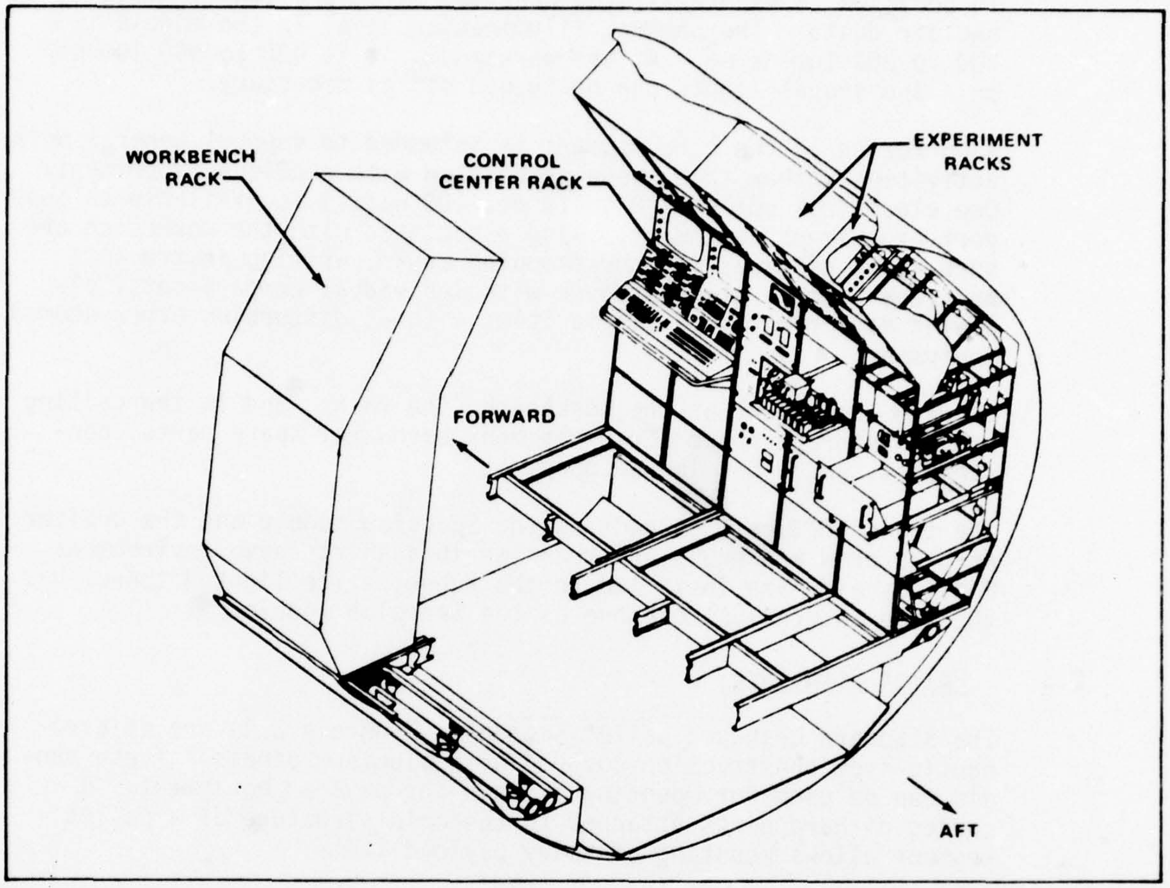


Figure A.2.3. Core Segment Cutaway View (Starboard).

The module interior is sized and shaped to allow optimum task performance by crewmembers in a weightless environment. The module can accommodate as many as three payload specialists working a 12-hour shift. For shift overlap, as many as four can be accommodated for an hour. The cabin air temperature is maintained between 64° and 81°F (291°and 300°K). The airflow is directionally controllable and is between 16.5 and 39.6 ft/min (0.084 and 0.201 m/sec).

Foot restraints, handholds, and mobility aids are provided throughout the Spacelab so that crewmembers can perform all tasks safely, efficiently, and in the most favorable body position. The basic foot restraint system is identical in Orbiter and Spacelab.

Fixed handrails and handholds are distributed throughout the habitable area, such as along the standard racks and along the overhead utilities/storage support structure.

In addition to handrails, the overhead structure contains lights and air ducts. The nominal illumination level in the module is 200 to 300 lumens/m^2. At the workbench, it is 400 to 600 lumens/m^2. Individual lights can be turned off as necessary.

A workbench in the core segment is intended to support general work activities rather than those associated with a unique experiment. One electrical outlet (28 volts dc, 100 watts) is available to support experiment equipment. Also associated with the workbench are such items as wipes for housekeeping tasks, writing instruments and paper, and a stowage pouch with individual compartments, allowing easy removal of single items without disturbing other stowed equipment.

Stowage containers at the workbench, the racks, and in the ceiling provide storage space for experiment hardware, spare parts, consumables, and other loose equipment.

The transfer tunnel connecting the Spacelab module and the Orbiter enables crew and equipment transfer in a shirtsleeve environment. Mobility aids are installed in the tunnel. The lighted tunnel has the same internal atmosphere as the Spacelab module.

2.3 Pallet Structure

The standard *U-shaped pallet segments (Figure A.2.4)* are of aeronautic-type construction covered with aluminum panels. These panels can be used for mounting lightweight payload equipment. A series of hardpoints attached to the main structure of a pallet segment allows mounting of heavy payload items.

The pallet provides basic services, such as:

- Subsystem and experiment electric power buses
- Experiment power distribution boses
- Subsystem and experiment data buses
- A subsystem RAU and as many as four RAU's for experiments
- Thermal insulation blankets
- Cold plates and thermal capacitors
- Plumbing

In a pallet-only configuration (one to five pallets with no module), *the Spacelab subsystem equipment that is ordinarily in the* module is installed in the igloo. The igloo, pressurized to 1 standard atmosphere (101 325 N/m^2), has a usable volume of 77.69 cubic feet (2.2 cubic meters). The internal temperature is compatible with commercial aviation and military equipment requirements. The equipment igloo weighs approximately 1410 pounds (640 kilograms).

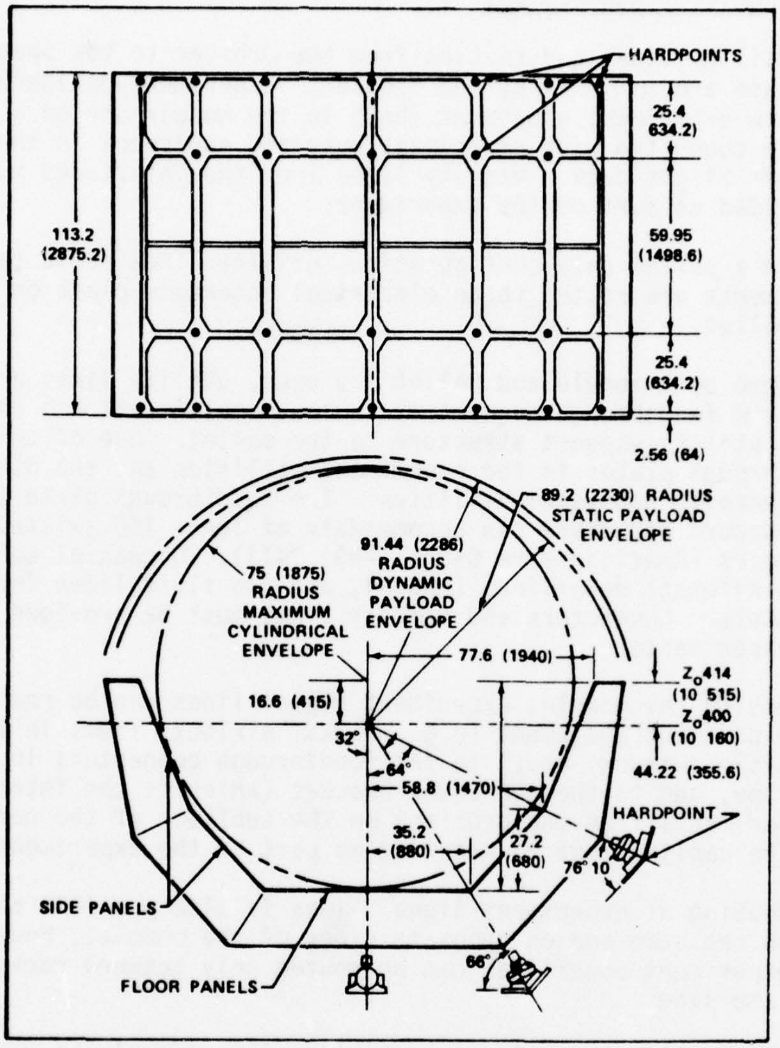


Figure A.2.4. Hardpoints and Envelope for Pallet Payloads

The following list is representative of items the igloo contains:

- Three computers
- Two input/output units
- A mass memory
- Two subsystem RAU's
- An emergency power box
- An experiment and a sub-system inverter (each 400 hertz) (each 400 hertz)
- A power control box
- A subsystem power distribution box
- A remote amplifier and advisory box
- A multiplexer
- A subsystem interconnecting station

On the ground, access to the igloo interior is through a removable bulkhead.

2.4 Utility Connections

Utility lines and routing from the Orbiter to the Spacelab interface are provided by the Orbiter. Experiment-dedicated lines allow experiment equipment (both in the module and on a pallet) to be connected with experiment-supplied equipment in the Orbiter aft flight deck. Utility lines from the interfaces must be provided as part of the experiment.

In a pallet-only configuration, utility lines dedicated to experiments are routed to an electrical interface plate on the first pallet.

When both module and pallet are used, utility lines are routed from feedthrough connectors in the module's aft end cone through a utility support structure to the pallet. One of the two feedthrough plates is for experiment utilities and the other for Spacelab subsystem utilities. The feedthrough plate and utility support structure can accommodate at least 100 twisted shielded pairs (American Wire Gauge (AWG) 241)), 20 coaxial cables, two 5-kilowatt powerlines (AWG 8), and two fluid lines for experiments. Connectors and utility lines must be provided with the experiments.

Inside the module, experiment signal lines can be routed between experiment equipment (e.g., racks, airlock, items in the center aisle, window, etc.) to the feedthrough connectors in the aft end cone, and to the connector bracket (which is the interface to experiments from the Orbiter) on the subfloor of the core segment. The cabling must be provided as part of the experiment hardware.

Routing of experiment signal lines is also possible between racks on the same and on opposite sides of the module. However, other experiment powerlines can be routed only between racks on the same side.

On a pallet, experiment utilities will normally be routed on top of the inner pallet panels.

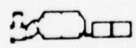
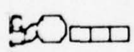
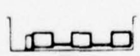
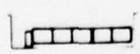
2.5 Payload Resources Summary

Tables A.2.2, A.2.3, and A.2.4 summarize the principal resources available to payloads using the Spacelab. All accommodations are described in more detail in the Spacelab Payload Accommodation Handbook.

Calculating power available to a payload is more complex than estimating mass or volume, because it depends on several other factors. Power for experiment use depends on the power consumption of the basic Spacelab subsystems and is also a function of the use of mission-dependent equipment. Therefore, no attempt has been made to provide a detailed power budget. To establish an accurate mission power budget, an extensive time-lining effort is required after basic experiment accommodation and functional requirements are fixed. In addition, the energy budget has to be considered. A maximum amount of power is available to the experiments if no discretionary subsystem or mission-dependent equipment is used, and a minimum amount of power is available if all the discretionary subsystem and mission-dependent equipment have been selected.

Table A.2.2

Electrical Power and Energy Resources for Payloads

Configuration	Parameter		
	Energy available to payload during flight, kWh (MJ)[a]	On-orbit power at electrical power distribution subsystem interface, kW[b]	
		Maximum continuous	Peak[c]
	123.2 (443.5)	2.5	6.9
	59.2 (213.1)	2.0	6.4
	55.2 (198.7)	2.0	6.4
	65.2 (234.7)	2.1	6.5
	62.2 (224)	2.1	6.5
	457.6 (1647.4)	4.6	9.4
	455.6 (1640.2)	4.6	9.4
	453.6 (1633)	4.5	9.3

[a]From basic 890 kWh (3204 MJ) Orbiter supply. Additional energy is available only by decreasing the mass capacity for experiments.

[b]With operation of mission-dependent nondiscretionary equipment. In addition, figures for all module configurations assume approximately 1 kW of power is available because some discretionary subsystem and mission-dependent equipment is not powered.

[c]15 minutes duration for 3-hour intervals.

Table A.2.3. Heat Rejection Capabilities and Module Atmosphere Aspects

Parameter	Configuration: Module	Configuration: Pallet
Atmosphere		(Igloo)
Nominal total pressure, bar (N/m^2)	1.013 ± 0.13 (101 300 ± 13 000)	1.096 (9600)[a]
Partial oxygen pressure nominal, bar (N/m^2)	.220 ± .017 (22 000 ± 17 000)	.035 (3500)[b]
Partial carbon dioxide pressure nominal, bar (N/m^2)	.0067 (670)	
Cabin air temperature, °F (K)	64 to 81 (291 to 300)	95 (308)[c]
Minimum humidity (dewpoint), °F (K)	43 (279)	
Maximum relative humidity, percent	70	
Maximum allowable internal wall temperature, °F (K)	113 (318)	
Air velocity in habitable area, ft/sec (m/sec)	0.33 to 0.66 (0.1 to 0.2)	
Total heat transport capability,[d] kW	8.5	8.5
Prelaunch/postlanding power,[d] kW		
GSE connected		
Orbiter powered down	Same as operational phase	Same as operational phase
Orbiter powered up	1.5	1.5
Ascent/descent	1.5	1.5
Peak heat rejection capability[d]		
For payload power peaks during operational phase, kW	12.4	12.4
Minimum interval between peaks, min	165	165

[a]Maximum gaseous nitrogen differential pressure.

[b]Minimum gaseous nitrogen differential pressure.

[c]Maximum internal temperature.

[d]Available to payload and Spacelab subsystems.

The command and data management subsystem is largely independent from the Orbiter. It provides data acquisition command, formatting, display, and recording. Communication with ground stations is through the Orbiter's communication system.

Table A.2.4. Command and Data-Handling Resources

Payload data acquisition	
Housekeeping and low rate scientific data (to computer via RAU's)	
Number of remote acquisition units (RAU's) of basic system	8
Maximum number of RAU's (extension capability)	22
Number of flexible inputs (analog or digital) per RAU	128
Analog: resolution of analog/digital conversion, bit	8
Discrete: number of inputs addressable as group	16
Number of serial pulse code modulation inputs per RAU	4
Clock rate, Mb/sec	1
Maximum number of words transferred per sample	32
Word lengths, bit	17
Maximum basic sampling rate, Hz	100
Data rate of transfer RAU/computer (including overhead), Mb/sec	1
Wideband scientific data	
Number of experiment channels of the high rate multiplexer (HRM)	16
Minimum data rate of HRM input channels, kb/sec	64
Maximum data rate of HRM input channels, Mb/sec	16
Number of closed circuit television video input channels	1
Number of 4.2-MHz analog channels	1
Data transmission to ground	
Nominal data rate for housekeeping and low rate scientific data from subsystem and experiment computer, kb/sec	64
Maximum data rate for wideband scientific data (via TDRSS), Mb/sec	50
Maximum data rate of high rate digital recorder (HRDR) bridging TDRSS noncoverage periods, Mb/sec	32
Storage capability of HRDR, bit	3.6×10^{10}
Payload command capability	
Telecommand rate from ground via Orbiter, kb/sec	2
Number of on/off command outputs per RAU	64
Number of serial pulse code modulation command channels per RAU	4
Clock rate, Mb/sec	1
Maximum number of words per command	32
Word length (including parity bit), bit	17
Payload data processing and displays	
Data processing:	
Word length, bit	16
Speed (Gibson mix), operations/sec	350 000
Floating point arithmatic, bit	32 (24+8)
Mass memory, Mbit	131
Display: alphanumerical display screen (tri-color diagonal, in. (cm)	12 (30.5)

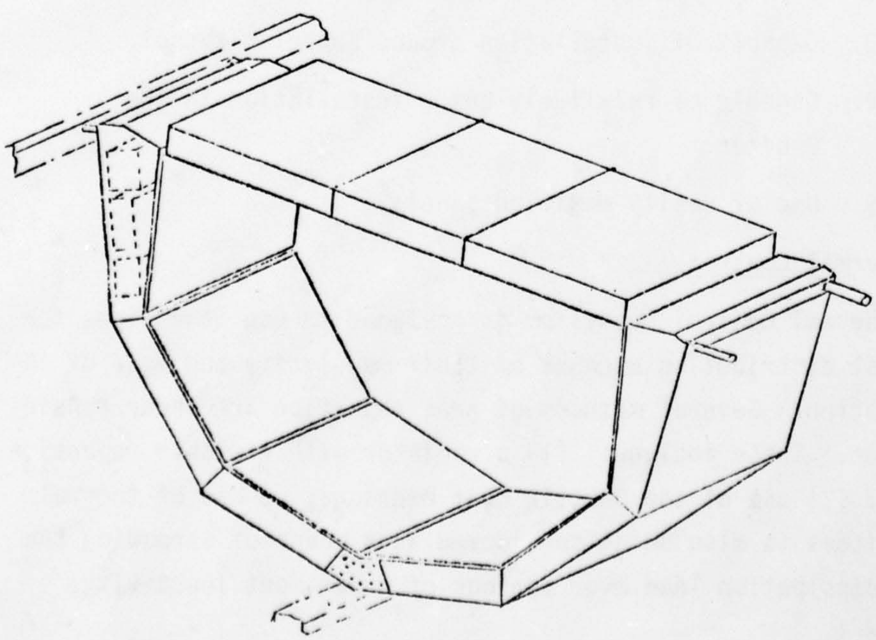
3.0 STANDARD TEST RACK

The Space Test Program is considering development of a general purpose experiment carrier to enhance program flexibility with the STS. This carrier, called the Standard Test Rack (STR), is presently in definition. Design goals are:

- Operate autonomously and limit the STS interface and installation activities to minimum. Provide for caution and warning, etc.
- Accommodate a wide variety of experiments by using modular subsystems that may be increased or decreased in size and performance capability to meet the needs of individual experiments in a cost effective and weight efficient manner.
- Permit integration installation of structure and subsystems into the Shuttle Bay and cabin (whenever necessary) quickly with minimum amount of time and effort.
- Develop a system with low life cycle costs.

3.1 Design Concept

The present design concept is illustrated in Figure A.3.1 below.



The Standard Test Rack features the following:

- 5000 lb. total load capacity
- Interchangeable equipment panels
- Subsystem components mounted inside structure
- Heat pipes for thermal distribution

3.2 Experiment Support Equipment

The following subsystems are proposed to provide experiment support. They are sized to accommodate the range of experiments for which STP has firm requirements defined. It should be noted that these subsystems are designed to make the STR nearly autonomous from the Shuttle Orbiter.

3.2.1 Structure

A box structure is designed for minimum weight to provide the following features:

1. Support masses of known experiments and the required subsystems for STS design loads.
2. Have modularity for a variety of installation options: full or half-rack; large or small shelf on a side; high or low bridge across full STR.
3. Capable of installation around Spacelab tunnel.
4. Capable of relatively quick installation in the Orbiter.
5. Use of easily modified panels.

3.2.2 Thermal Control

The thermal control subsystem is designed to use heat pipes for thermal distribution because of their modularity and ease of installation. Several methods of heat rejection are under consideration. These include: (1) a radiator with variable rejection ratio; (2) use of the Shuttle heat exchanger. Use of thermal capacitors is also being considered as a means of spreading the heat dissipation load over periods of experiment inactivity.

3.2.3 Electric Power

A battery power system will be used, with the following characteristics:

Power	600 W	average
	3 KW	peak
Energy	38 KWH	(7 day mission)
Voltage	28 $\pm$4 Volts DC	

3.2.4 Attitude Control

As needed, attitude knowledge will be provided by fixed star trackers, an inertial reference unit, and appropriate computational capability. The STR will be able to accommodate any of the proposed pointing systems (See Section A5)

3.2.5 Communications

It is proposed that the STR be capable of independent command reception and telemetry transmission, working with the Air Force Satellite Control Facility. In this mode, the STR would be equipped with an antenna, receive and transmit equipment, encode and decode equipment, and an appropriate tape recorder. The following capabilities are projected:

Command	1 KBPS
Telemetry	1.024 MBPS digital
	5-10 kHz analog streams
Downlink Power	20 Watts maximum

Capability will be provided for making use of the Orbiter communications system for those missions where operational problems preclude use of the autonomous system.

4.0 SMALL SELF-CONTAINED PAYLOADS

In late 1980, the Space Shuttle will have completed its half-dozen developmental flights and will be into routine operations. On each flight, there will be one or more "primary payloads" in the Shuttle's payload bay. However, such payloads will not always occupy the total space available, or add up to the maximum allowable weight.

The STS Operations Directorate at NASA Headquarters (Code MO), which has broad responsibility for system operations, is also responsible for establishing and conducting a program to fly small experiments that will take advantage of the extra space/weight opportunities as they arise. This constitutes an unusual opportunity for individuals, companies, and institutions that would like to conduct experiments in space at moderate cost. The name of the program is Small Self-Contained Payloads or "Getaway Special."

Small self-contained packages under 200 pounds and smaller than five cubic feet which require no Shuttle services (power, deployment, etc.), and are for R&D purposes, will be flown on a space-available basis during both phases of Shuttle operation. The price for this service will be negotiated based on size and weight, but will not exceed $10,000 in 1975 dollars. A minimum charge of $3,000 in 1975 dollars will be made. If Shuttle services are required, the price will be individually negotiated. Reimbursement to NASA will be made at the time the package is scheduled for flight.

Numerous safety precautions are being applied to the design and testing of all flight hardware, as well as to the operating procedures. An effort will be made to simplify and streamline these requirements as they apply to the Getaway Special experiments, but experimenters are cautioned that there will be a certain minimum set of requirements that NASA must invoke. It is anticipated that these requirements will be defined and issued by early 1978. They will include some review of the experiment design as well as the hardware "as built" and analysis and/or test to establish structural integrity of the package.

4.1 Power

It is recognized that most experimenters would like electrical power supplied "from outside" to provide thermal conditioning if for no other reason. Some types of experiments need considerably more power. On the other hand, the Orbiter fuel cells (batteries) will face heavy demands from the primary payloads and routine operations. To add fuel cells to the Orbiter is both expensive and time-consuming. NASA makes no assurances that any external power will be available on Orbiter missions, even at extra cost.

4.2 Crew Activity

The need for a few elementary commands that will be initiated by the crew is foreseen. The criteria and constraints for such signals must be simple, to avoid the need for training, computers, or uplink communications. Generally, there will be no opportunity for crew observation of your experiment, or for any form of in-flight servicing.

4.3 Pre-Launch Test of Servicing

NASA is considering the advisability of shake-testing experiments in lieu of certain safety-oriented analyses. In any event, you may need some testing and/or servicing of the experiment shortly before it is installed in the Orbiter bay. Examples are insertion of film or performance of an electrical continuity test. Such services are likely to be available at a very nominal extra charge. Access or servicing after the package is installed in the Orbiter will not be permitted.

4.4 Post-Flight Information

NASA will issue a certificate attesting to the fact that your experiment was orbited on a particular Space Shuttle flight and the associated date(s). In addition, we will furnish a brief summary of the timing and orbital parameters, so that you can determine the conditions prevailing when your experiment was active. More comprehensive after-the-fact computer data on the particular orbit may be available at some additional charge.

4.5 Containers

All experiment equipment must be contained within a package with the following characteristics:

(1) Mounting lugs or surfaces to attach to the supporting brackets NASA will furnish.

(2) A reasonably rugged enclosure to protect the contents from environmental damage, on the ground as well as in flight.

(3) Weight, excluding the enclosure, not to exceed 200 pounds.

(4) Volume, excluding the enclosure, not more than 5 cubic feet.

NASA is considering building a family of experiment containers and make them available at nominal cost. Probable characteristics are:

Size:	Choice of 1.5, 2.5, or 5.0 cubic feet
Shape:	Rectangular. A candidate set of internal dimensions for the large container is 15x 20x29 inches. The long dimension will run fore-and-aft when installed in the Orbiter, and the short dimension will be transverse.
Thermal Character-istics:	Coating to maintain the outside surface within the limits of 0° to 100°F through-out the anticipated flight. Insulation to make it easier for you to control internal temperature within whatever range you choose.
Venting:	A number of considerations favor a freely vented package. If there is sufficient demand for a sealed package, NASA will con-sider a standard sealed container of at least one size.
Opening or Window:	Consideration is being given to a lid in the top of the container (i.e., the 15x29 face) which can be opened and closed on command from the crew station. The hinge would be parallel to the longitudinal axis of the Orbiter. Alternatively, there may be a window as large as 9" diameter that will have good transparency throughout the visible spectrum and extend into the UV range.
Electrical Interface:	A receptacle in the 15x20 end of the con-tainer, to receive any signals and/or power that may be provided.
Weight:	Not over 30 pounds, including insulation, for the large package; or 18 pounds for the smallest one.
Internal Mounting:	You may mount your experiment to the shell of the container by bolting through the in-sulation, near the corners of the rectangular envelope. We will allow as much latitude as practicable; however, attachment to the cen-tral portion of flat panels (i.e., away from the corners) is discouraged because of the attendant questions concerning structural and dynamic integrity.
Reuse:	To control costs for all concerned, the con-tainers will be designed to be reusable for at least 5 to 10 experiments. Coordination with GSFC will be required before you drill holes in, or otherwise alter, the container assigned to you.

5.0 PAYLOAD SUPPORT SYSTEMS

Experiment support equipment which is adapable to many experiments is discussed in this section. This equipment is currently planned and can be used by experimentors. The following systems are included:

- Pointing and stabilization systems
- Spacelab airlock
- Spacelab optical window and viewport

5.1 On-Orbit Pointing and Stabilization Systems

The Orbiter is capable of attaining and maintaining specified inertial, celestial, or local (vertical) Earth reference attitudes. For payload pointing by use of the vernier thrusters, the Orbiter flight control system provides a stability (deadband) of ±0.1 deg/axis and a stability rate (maximum limit cycle rate) of ±0.01 deg/sec/axis. When using the primary thrusters, the Orbiter provides a stability of ±0.1 deg/axis and a stability rate of ±0.1 deg/sec/axis.

The Orbiter capability to point a vector defined in its inertial measurement unit (IMU) navigation base axes (using the Orbiter IMU for attitude information) is summarized in Table A.5.1. The duration of continuous pointing within a specified accuracy is primarily dependent upon the IMU platform drift.

Table A.5.1. Total (Half-Cone Angle) Pointing Accuracy Using Orbiter IMU

Reference	Half-cone angle pointing accuracy (3 sigma), deg[a]	Pointing accuracy degradation rate (3 sigma), deg/hr/axis	Duration between IMU alignments, hrs
Inertial and local vertical	·0 5	0 105	1 0
Augmented inertial	·0 44	0	NA
Earth surface fixed target	·0 5	0 105	0 5

[a]Mechanical and thermal tolerances may degrade pointing accuracy as much as 2°

With augmented pointing systems and procedures, however, the pointing duration may be restricted by operational constraints such as thermal or communication considerations. Typical Orbiter RCS maximum acceleration levels during maneuvering and limit cycle pointing control are also shown in Table A.5.2. These figures are for single-axis (one degree of freedom) maneuvers, based on an Orbiter with 32 000 pounds (14 515 kilograms) of cargo.

Table A.5.2 Typical Orbiter RCS Maximum Acceleration Levels

RCS system	Translational acceleration, ft/sec^2 (m/sec^2)					Rotational acceleration, deg/sec^2			
	Longitudinal		Lateral	Vertical					
	+X	−X	+Y	+Z	−Z	+ Roll	+ Pitch	− Pitch	+ Yaw
Primary thruster	0.6 (0.19)	0.5 (0.16)	0.7 (0.22)	1.3 (0.40)	1.1 (0.34)	1.2	1.4	1.5	0.8
Vernier thruster	0 0	0 0	0.007 (0.0021)	0 0	0.008 (0.0024)	0.04	0.03	0.02	0.02

Instrument pointing systems are available to provide precision pointing for payloads that require greater pointing accuracy and stability than is provided by the Orbiter. These systems can accommodate a wide range of instruments of different sizes and weights. Four systems are currently planned:.

1) Spacelab instrument pointing subsystem (IPS)
2) NASA/GSFC small instrument pointing system (SIPS)
3) NASA/LeRC annular suspension pointing system (ASPS)
4) USAF Payload Orientation and Instrument Tracker for Shuttle (POINTS)

Pointing and stabilization characteristics of these systems are shown in Table A.5.3.

Table A.5.3. Pointing System Requirements vs. Pointing System Capability

	IRBS REQUIREMENT	SIPS CAPABILITY	ASPS CAPABILITY	POINTS
Envelope	Sensor: .66 x 1.62M He Dewar: 1.17 dia x .25M	.9 x .9 x 3.0M	Sensor size relatively unconstrained. 1 Meter dia. interface ring	2 M Width
Weight	675 kg	600 kg/yoke	600 kg CM < 1.5 meters from the gimbal	680 KG or two x 340 KG
Pointing Angle Access	Hemispherical	Hemispherical	$\pm$100° elevation $\pm$60° cross-elevation	-60°, +90° elevation $\pm$180° azimuth
Scan Rates	~ 0.1 deg/sec	Up to 2 deg/sec	TBD	2 deg/sec.
Pointing Stability	Fine tracking to within 20 arc sec.	$\pm$0.3 arc sec (1 σ)	$\pm$0.01 arc sec (1 σ)	Can be accommodated, stability depends on pointing sensor used.

Spacelab Instrument Pointing Subsystem (IPS)

The IPS provides three-axis attitude control and stabilization for experiments. Overall control of the IPS during normal operations is exercised from the Spacelab control console using the keyboard and display of the command and data management subsystem. The flight operating software is capable of interfacing, through the Spacelab subsystem, with the Orbiter data-handling system. The IPS control system uses this Spacelab subsystem for all normal operations. Emergency retraction or jettison is exercised from a separate IPS control panel located on the Orbiter aft flight deck.

Attitude control of the payload is based on rate-integrating gyro error signals processed within the Spacelab computer to generate command signals. The rate-integrating gyro package is on the outer gimbal; therefore (aside from distortion or flexures occurring within the payload), it can maintain the payload as an inertially stabilized platform.

To correct for gyro drift and to provide an absolute attitude referance, a package of optical sensors is also included. In a stellar mission, this would consist of three star trackers; in a solar mission, one star tracker would be replaced by a solar sensor.

The IPS provides the following interfaces across the gimbal system for use of payloads.

- Wiring for four independent 800-watt peak (15 minutes maximum) power loads at 28 volts dc; one of the four sets of wiring is capable of carrying 115 volts, ac, 400 hertz.
- Wiring for three remote acquisition units.
- Ten coaxial cables, each adequate for transmission of the Orbiter high data rate of the Ku-band signal processor.

During ascent and descent, the payload is physically separated from the IPS to avoid imposing flight loads from the IPS to the payload. The payload is supported by the payload clamp assembly , which distributes the flight loads of the payload.into the pallet hardpoints. The payload clamp assembly is capable of mounting and distributing the load of a nominal 4410-pound (2000-kilogram) payload and the IPS into a single unmodified pallet without exceeding safe loading conditions.

Small Instrument Pointing System (SIPS)

The Small Instrument Pointing System provides precision control from a stellar inertial attitude reference system employing gyros and a strapdown sensor, all mounted on the SIPS canister. The majority of the software and electronic data processing functions are performed by programmable digital electronics (PDE). It has been demonstrated that the SIPS can be pointed with 95 percent confidence to within 18.8 arc-sec (2σ). The short-term (10 to 20 seconds) pointing stability error is less than 0.5 arc-sec (2σ).

A drawing of a pallet mounted SIPS is shown in Figure A.5.1. The SIPS has two instrument carrying canisters, each supported at its center by a yoke which can rotate independently of the other canister in an up-down direction (120 degrees freedom). Each canister in turn is connected to the yoke so as to provide a limited (± 10 degrees) left-right rotational degree of freedom. Both yokes are attached to a common ± 180 degree azimuth gimbal drive at the base. An optional roll gimbal about the instrument line of sight can be added internally to each canister. A strapdown star tracker and a gyro reference assembly, providing a precision inertial attitude reference for the instrument mounts to the right of the right canister. This attitude reference is transferred to the instruments in the other canister and to the pallet through the gimbal angle resolvers. Orbiter ephemeris data is required to go from the inertial attitude reference to earth referenced pointing.

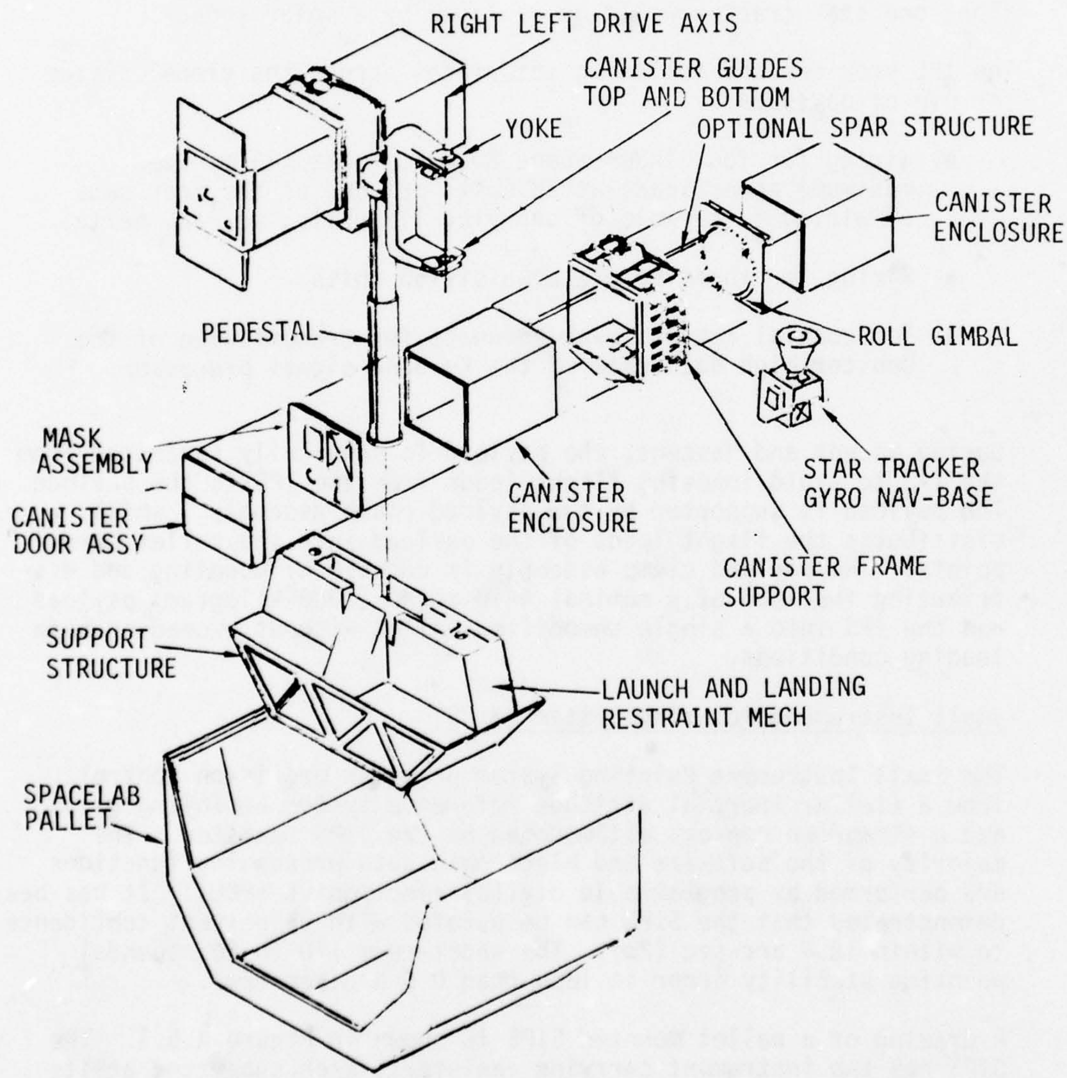


Figure A.5.1. Small Instrument Pointing System (SIPS)

The servo control laws and the gyro and star tracker data may be processed by programmable digitial electronics consisting of function dedicated microprocessors or the Spacelab/Orbiter computer. The gyro-sensed rates are integrated to provide a high bandwidth, stable attitude reference. The star tracker data is processed by a filtering algorithm that provides optimal attitude and gyro bias updates assuring long term attitude reference stability. In addition, the gyro data is processed to provide gimbal rate information for the SIPS gimbal servos.

Annular Suspension Pointing System (ASPS)

The NASA Annular Suspension Pointing System has a 600 Kg weight capability and is more compact that the IPS. A drawing of the ASPS is shown in Figure A.5.2. In this design, a three-axis magnetically suspended platform is mounted on top of a two-axis (elevation and cross-elevation) mechanical gimbal set. In operation, the suspended platform is only coupled optically to the base for data transfer, and the payload power is supplied by batteries.

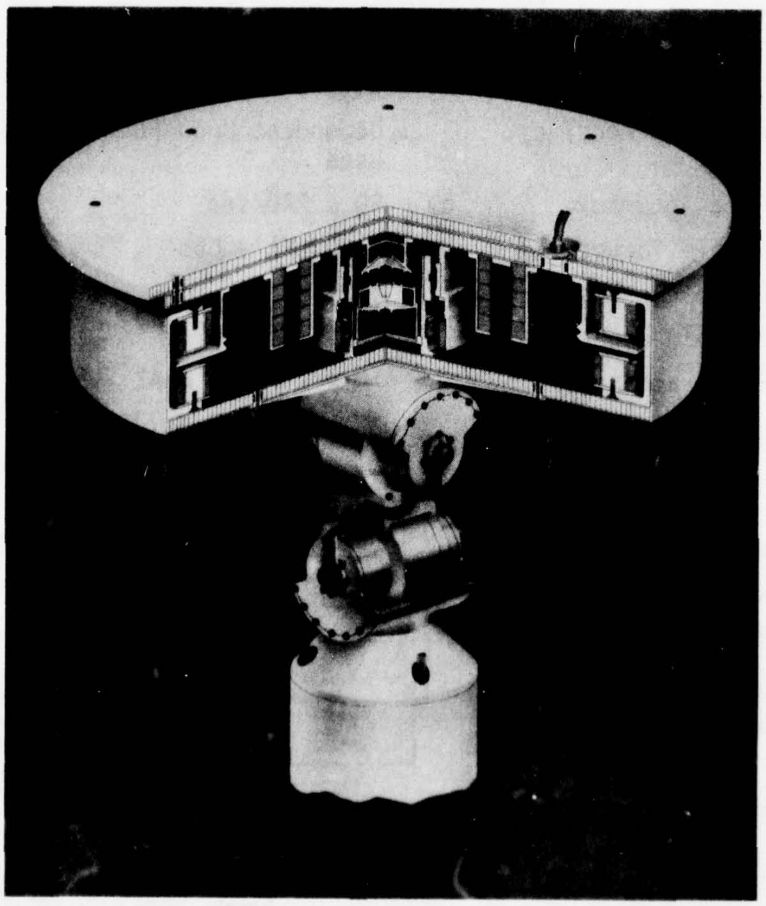


Figure A.5.2. Annular Suspension Pointing System

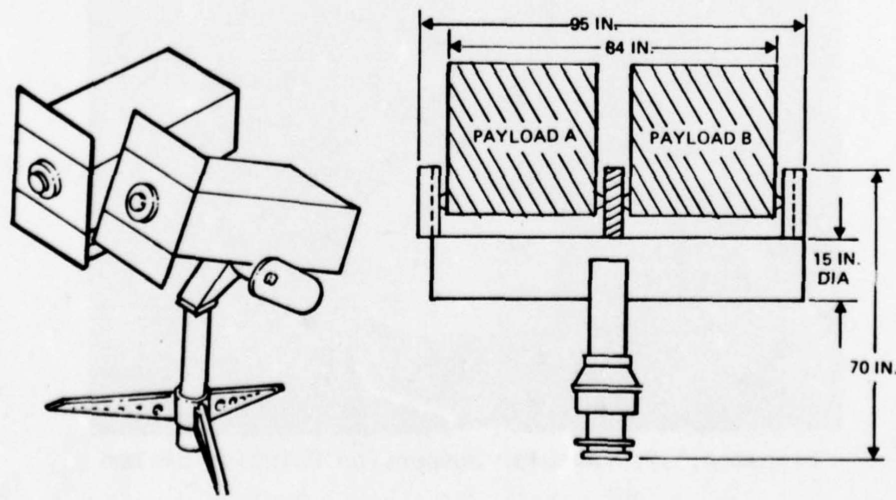
Payload Orientation and Instrument Tracker for Shuttle (POINTS)

The POINTS is a gimbal assembly that has been built and functionally tested for an Air Force program. It can independently point two 750 pound experiments or a single heavier experiment. It features: beryllium structure for high stiffness, low thermal deflection, and light weight; redundant torque motors; deployment to improve field of view; can be used with external sensors such as star tracker horizon scanners, etc., for closed loop control.

Capabilities of the system are shown below:

- Azimuth Gimbal Freedom: ± 180°
- Elevation Gimbal Freedom: -60° to +90°
- Torque Available: AZ = 44 Ft - Lbs: EL - 18 Ft - Lbs
- Nominal Slew Velocity: 2 Deg/Sec (1500 Lb Payload)
- Nominal Acceleration: 4-1/2 Deg/Sec^2 (1500 Lb Payload)
- Position Resolution: 20 Arc Sec (16-Bit Encoder)
- Position Accuracy: Dependent Upon Pointing Sensor Used
- Rate Accuracy: 80 μ RAD/sec
- Ripple Torque: 0.003 Ft - Lbs
- Gimbal Assembly Weight: 450 Lbs
- Payload Capability: 1500 lbs
- Power at Stall: 70 W El: 224 W AZ

A sketch of the POINTS is shown below.



Payload Orientation and Instrument Tracker for Shuttle (POINTS)

5.2 Spacelab Airlock

The airlock enables experiments to be exposed to a space environment. Experiments are mounted on a sliding platform parallel to the airlock axis. This platform can be extended into space, where it is protected by a removable thermal shield. Experiments can be observed through an inner hatch window 5.9 inches (15 centimeters) in diameter that provides a 120° viewing angle. The platform can also be pulled back into the Spacelab module for experiment mounting and checking (both on orbit and on the ground). The inner hatch can be completely detached for payload installation and access. All controls are manual.

A control panel on the outside of the cylindrical shell provides for monitoring the airlock operations. Monitoring and display of airlock status are also provided by the command and data management system. Electrical and mechanical interlocks prevent dangerous operations sequences.

The platform, when extended into space, penetrates the Orbiter cargo bay envelope (as does the outer hatch). To preclude a critical situation if the retraction or hatch mechanism malfunctions, both the sliding platform and hatch are capable of being jettisoned.

The dynamic envelope available for experiments is illusted in Figure A.5.3. The top airlock is 3.28 feet (1 meter) in diameter and the same length. It is designed to carry an experiment or experiments with a total mass of 220 pounds (100 kilograms) during launch and descent.

Power connectors are provided at the platform for 28 volts dc primary, 200 watts; and 115/200 volts ac, 400 hertz/three-phase, 3 amperes.

Experiment data handling and control can be performed either by the command and data management subsystem (by use of an experiment remote acquisition unit (RAU) mounted on the airlock platform) or by hardwired lines through the airlock shell to payload equipment in the module.

A flexible cable harness connects the platform with feed through connectors in the airlock shell; thus, experiment equipment can be checked while the platform is pulled into the module.

Inside the airlock, illumination is 100 lm/m^2 (controlled from the airlock control panel).

Seven repressurization cycles per 7-day flight for the experiments can be accommodated by the basic environmental control subsystem nitrogen resources. Additional repressurizations will depend on usage of nitrogen by the environmental control and life support subsystem.

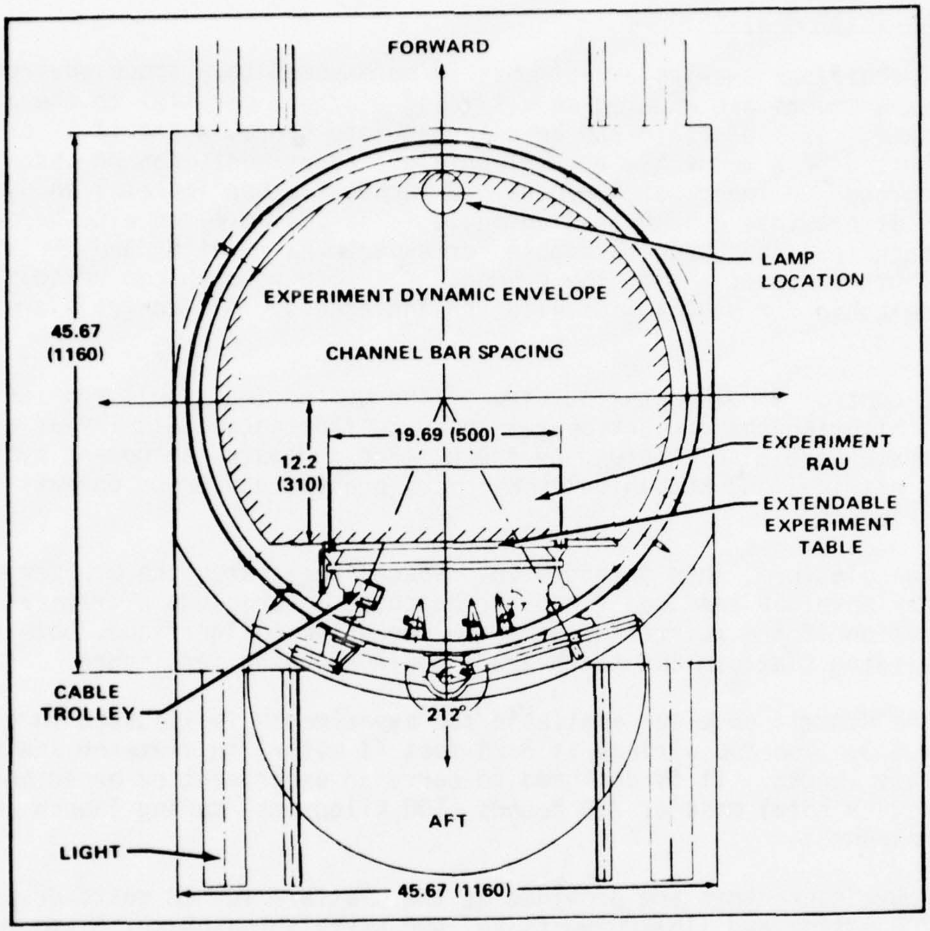


Figure A.5.3. Spacelab Airlock

The top airlock can be mounted into a single-segment module; however, planned or contingency ground operations (those requiring late access to the module) may prevent use of the airlock in the core segment.

5.3 Optical Window and Viewport

The optical window (adapted from the Skylab S190A window) consists of a single rectangular pane of BK-7 glass measuring 16.14 by 21.65 inches (41 by 55 centimeters) and having a thickness of 1.61 inches (4.1 centimeters). It is enclosed in a molded seal and supported by a flexible spring system in an aluminum frame. An automatic heating system controls window temperatures to minimize thermal gradients across the glass and to prevent condensation. This power use is charged to payload and mission-dependent equipment.

When the window is not in use, a manually operated cover protects the glass outside from radiation, meteoroid impact, contamination, etc. A removable glass safety shield inside protects the window from impacts and provides a redundant pressure seal.

The window transmission characteristics are shown in Figure A.5.4. Some other optical characteristics are as follows:

Parallelism	2 arc-seconds
Reflectance	2 percent on inside; 4 percent on outside
Seeds and bubbles	Total area 0.1 $mm^2/100\ cm^3$ of glass; maximum dimension of single imperfection, 0.76 millimeter
Surface quality	60 to 40 or better (as defined in MIL-13830)

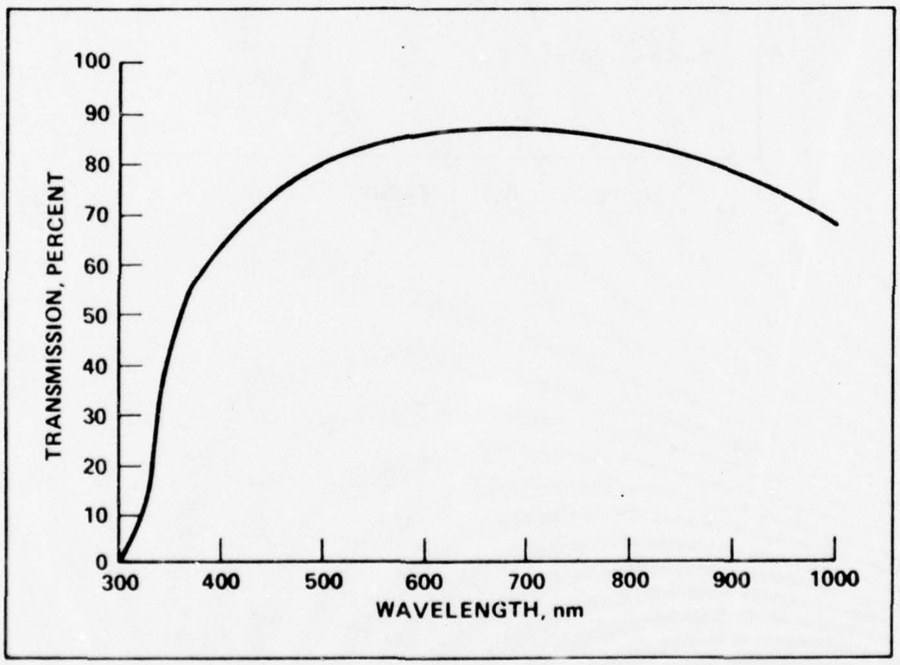


Figure A.5.4. Transmission Characteristics of S190A Window

Viewports can be installed in the aft end cone or as part of the window assembly. Each consists of two panes 11.81 inches (30 centimeters) in diameter, the outer one of quartz glass and the inner pane of safety glass. Design characteristics are shown in Figure A.5.5. Experiment-mounting capability can be provided on the interface flange.

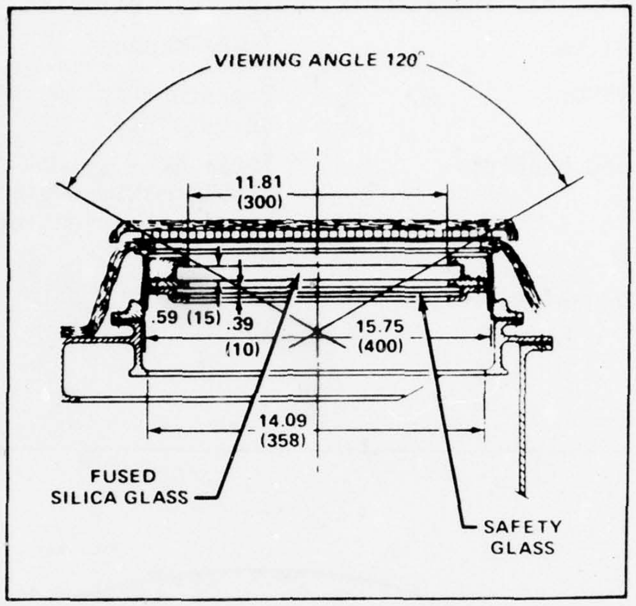


Figure A.5.5. Viewport Design

6.0 LONG DURATION EXPOSURE FACILITY (LDEF)

The LDEF is a reusable, unmanned, gravity gradient stabilized, free-flying structure on which many different experiments can be mounted. It provides an easy and economical means for conducting experiments in space.

The LDEF will be placed in an Earth orbit by the Shuttle where it will remain for an extended period of time. A subsequent Shuttle flight will retrieve the LDEF and return it to Earth so that experiments can be removed and returned to the experimenters. Experiments for LDEF can be either passive or active. For passive experiments, the data measurements will be made in the laboratory before and after exposure to the space conditions. For active experiments, the data gathering may require such active systems as power, data storage, etc. Such active systems must be provided by the experimenter and be an integral part of his experiment assembly. Each LDEF experimenter will be responsible for participating with the LDEF Project Office in establishing that the experiment is safe to fly and will not adversely affect the other experiments carried on the LDEF. The LDEF Project Office also has available certain common LDEF compatible equipments. Common items currently being developed include a vacuum exposure control canister and an electrical power and data system.

6.1 Mission Description

The first orbital exposures for LDEF are planned as part of the Shuttle Orbital Flight Test (OFT) series will begin about July 1979 and extend for a year. The LDEF will also be flown during operational Shuttle flights in accord with experiment needs. All flights will originate from the Kennedy Space Center (KSC). In the delivery flight, the LDEF will be placed in a circular orbit at an altitude of about 300 nautical miles (556 km) with an inclination to the equatorial plane between 28.5° and 57°.

In orbit, the Remote Manipulator System (RMS) of the Shuttle Orbiter will remove the LDEF from the payload bay. Before release, the longitudinal axis of the LDEF will be aligned with the local Earth vertical, other required orientations established and the angular velocities brought within specified limits. After release, gravity gradient stabilization will be used in combination with a viscous magnetic damper to null transients; within 8 days the steady state pointing will remain within 2° of local Earth vertical, and oscillations about the longitudinal axis will be kept within 5°. During the planned exposure of 6 to 9 months in orbit, the altitude will decay about 20 nautical miles (37 km); at that time a subsequent Shuttle flight will capture and restow the LDEF within the payload bay for the return to Earth. Upon landing, LDEF will be removed from the Shuttle payload bay and the experiments will be removed from LDEF and returned to the experimenters for analyses.

6.2 Orbital Environment

Table A.6.1 summarizes the principal "natural" contributors to the environment associated with orbital operation of the LDEF and represents the conditions available for the conduct of a space experiment. The orbital natural environment combines with the predicted thermal environment shown below to establish the conditions under which an experiment must operate. The overall environment will vary as a function of LDEF orbit and location on the LDEF. In addition, an experiment environment can be modified by such design parameters as shielding, pressure-sealed containers, special thermal coatings, etc.

Thermal Environment

PREDICTED LDEF TEMPERATURE RANGES (57° INCLINATION ORBIT)

LOCATION	°C MIN.	°C MAX.	TYPICAL ΔT/ORBIT	LOCATION	°C MIN.	°C MAX.	TYPICAL ΔT/ORBIT
LDEF INTERNAL AVG.	-30	35	-	TYPICAL EXPERIMENT			
EARTH END (ALUMINUM SURFACE)	-25	40	5°C	$\alpha = 0.3$, $\epsilon = 0.3$			
				INTERNAL SURFACES	-35	50	3°C
SPACE END	-25	45	25°C	EXTERNAL SURFACES	-40	75	-
TEMPERATURES, COULD VARY BY AS MUCH AS ±15°C DUE TO VARIATIONS IN DESIGN, COATINGS AND ACCURACY OF THE MATHMATICAL MODEL.				$\alpha = 0.25$, $\epsilon = 0.17$			
				INTERNAL SURFACES	-50	65	10°C
				THIN EXTERNAL SURFACES	-115	150	200°C
				$\alpha = 0.3$, $\epsilon = 0.8$			
				INTERNAL SURFACES	-47	30	6°C
				EXTERNAL SURFACES	-75	30	-
THESE DATA ARE FOR EXPERIMENTS USING TYPICAL SURFACE COATINGS; OTHER TEMPERATURES ARE OBTAINABLE BY USING DIFFERENT SURFACE PROPERTIES.							

The values shown in Table A.6.1 reflect the variations in the environment due to the possible range of orbits. Since the orbit cannot be defined at this time, the prospective experimenter will have to work initially with the values given.

After experiment selection, the LDEF Project Office will define the specific conditions available as a function of location on the LDEF and work closely with each experimenter so that the optimum location for his experiment can be chosen.

Modification of the environment by the design of the experiment package will be the responsibility of the experimenter. However, the LDEF Project Office will provide consultation on applicable techniques and design approaches.

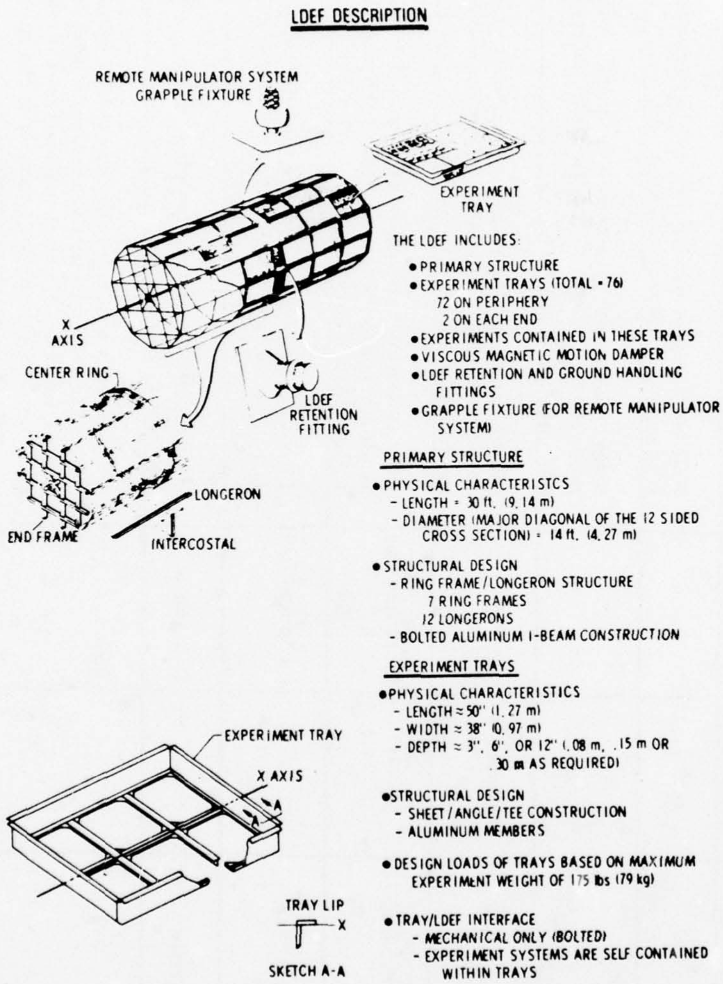
6.3 LDEF Description

The LDEF is a structural framework whose cross section is a 12-sided regular polygon. The LDEF description shown below gives concepts and principal dimensions. The primary framework consists of ring frames and longerons fabricated from aluminum extrusions. Trays containing experiments will be mounted into the bays formed by the ring frames and longerons. The LDEF can accommodate 72 trays on the periphery of the structure plus four additional trays, two on each end. The experiment trays,

TABLE A.6.1 ORBITAL ENVIRONMENT

ENVIRONMENT	MINIMUM	NOMINAL	MAXIMUM	COMMENT
Acceleration		10^{-6}g Continuous		See also Section 6.2 of Table 2
Ambient Pressure		3×10^{-8} torr		Will be modified by LDEF Interactions (Outgassing, Surface Chemistry)
Ambient Atmospheric Number Density		$10^8/cm^3$		Major component is atomic oxygen (0). Will be modified by LDEF interaction. (Outgassing and Surface Chemistry)
Ambient Atmospheric Ion Particle Density		$10^5/cm^3$		Major component is singly ionized atomic oxygen (0+) (Ref NASA Publication SP8049). Will be modified by LDEF interactions.
High Energy Particle Flux (Electrons)	$10^8/cm^2$ day		$3 \times 10^9/cm^2$day	Flux is for energies greater than 0.5 MEV. Orbit is electron radiation free 86% of the time (Ref NASA-GSFC X601-75-170).
High Energy Particle Flux (Protons)	$10^6/cm^2$day		$2 \times 10^7/cm^2$day	Flux is for energies greater than 5.0 MEV. Orbit is proton radiation free 83% of the time (Ref NASA-GSFC X601-75-170.)
Magnetic Field Intensity	0.12 gauss		0.6 gauss	Value varies as a function of LDEF position relative to Earth.
Solar Radiation	10%		25%	Percentages are referenced to the direct continuous solar radiation at normal incidence integrated over 1 year for a 28.5° orbit inclination (excludes albedo effects). The limits will vary with the mission duration and the launch time of the year.
Meteoroid Fluxes and Corresponding Masses	$10^{-14}/m^2$ sec for masses of 10^{-1}gram $10^{-5}/m^2$ sec for masses of 10^{-9}gram			Approximate intermediate values may be obtained by connecting the given end points on a log-log graph (Ref. NASA Publication SP-8013).

fabricated from aluminum sheets and extrusions, will be provided by the NASA through the LDEF Project Office.



Under "Experiment Trays" in above picture, the structural concepts are shown, the principal dimensions are listed, and the weight capacitities for the standard trays are defined. The availability of three standard tray experiment arrangements would be to use 1/6 of a tray, 1/3 of a tray, 2/3 of a tray or a full tray. Individual experiments will be bolted to the trays. Experiment sizes are not necessarily limited to the dimensions of trays; on a special case basis, heavier or larger experiments and other mounting locations or arrangements will be considered. In no case can an experiment protrude beyond the planes defining the 12-sided polygon of the LDEF.

6.4 Experiment Integration

The LDEF Project Office has the overall responsibility for the experiment integration. This includes integrating experiments into trays for partial tray experiments; providing for the correct placement of trays on the LDEF to obtain the desired exposure, field of view, etc.; and for assuring the mutual compatibility of all experiments.

The experimenter is invited to participate in launch-site operations involving his experiment and to verify readiness for flight. After the orbital exposure and return of the LDEF to the processing area at Kennedy Space Center, the experimenter will be provided an opportunity to view his experiment prior to removal from the LDEF. Experiments will be returned to the experimenter.

6.5 Flight Acceptance Tests

Environmental tests, formulated from Orbiter launch and return environments will be conducted in ground facilities to verify the flight worthiness of each experiment. Tests which might be part of flight acceptance include: shock, vibration, acoustics, steady-state accelerations, thermal-vacuum, and venting. The LDEF Project Office will be responsible for defining and conducting the flight acceptance tests. The experimenter will be responsible for defining and conducting the flight acceptance tests. The experimenter will be responsible for designing his experiment to pass the flight acceptance tests.

6.6 Experiment Assurance Considerations

In the design and fabrication of each experiment, appropriate considerations shall be given to assure that the experiment will reliably perform its function and do so without affecting other experiments on the LDEF.

6.7 LDEF Experiment Support Items

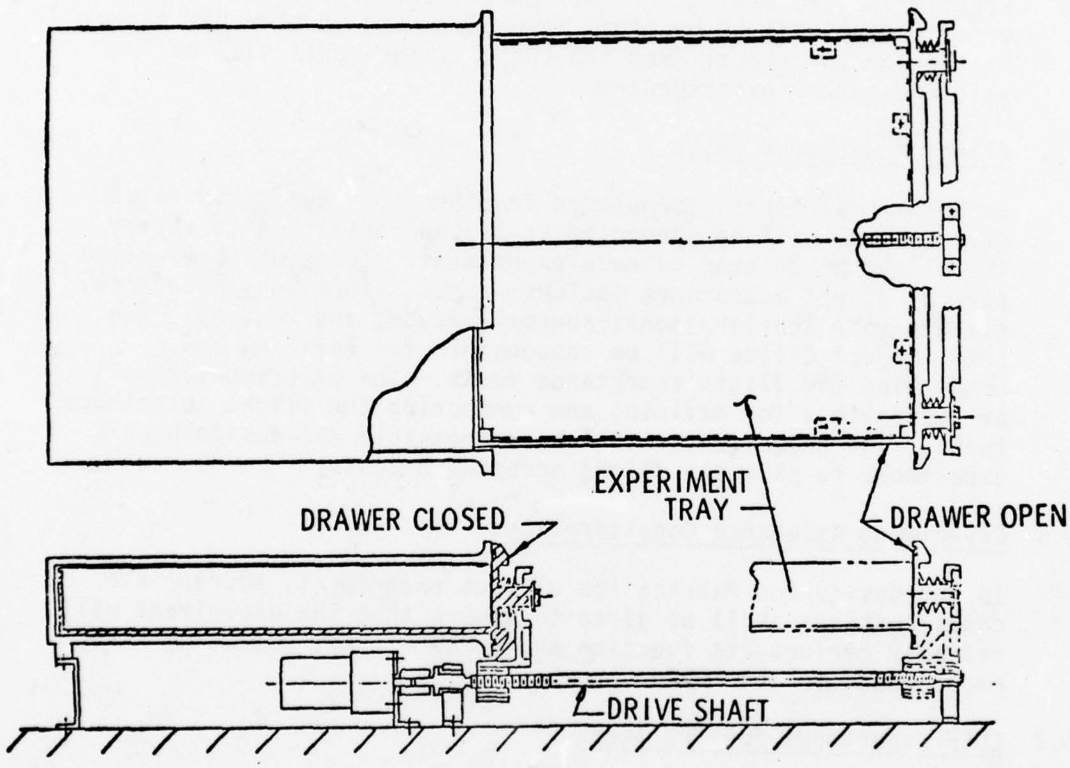
For experiments with an identified need, the LDEF Project Office can make available certain experiment support items. Two items of common equipment are currently being developed:

(1) Electrical Power and Data System (EPDS)

(2) Vacuum Exposure Control Canister (VECC)

The EPDS system is applicable to those experiments which require a number of measurements a few times each day during the course of an LDEF Flight. The system accommodates a range of both analog and digital formats together with providing tape recorder storage of the order 5.7 x 10^6 bits. A modularized lithium cell battery package powers the EPDS; battery modules can be sized to provide power for the experiment.

The VECC canister mounts within a tray and provides an area approximating 1/6 of a tray which moves out of a sealed container in a manner similar to the opening (and closing) of a drawer. The canister provides a means for maintaining a clean, low pressure environment during ground operations together with the opportunity to control the duration of an exposure to space conditions. In addition the design provides vacuum sealed penetrations for leads from electrical instrumentation.



Vacuum Exposure Control Canister
Preliminary Design

7.0 MULTIMISSION MODULAR SPACECRAFT

The Multimission Modular Spacecraft (MMS) can be used in low Earth and geosynchronous orbits for a wide range of remote-sensing missions. Although not classified as an STS element, it is a planned NASA payload carrier fully compatible with the launch environments and other requirements of the Space Shuttle as well as with a variety of expendable launch vehicles (including the Delta 2910 and 3910 series).

The reusable MMS offers several significant advantages over the conventional uniquely integrated spacecraft. Within its standard range of capabilities, it can be adapted to many varied payload requirements, eliminating the need for costly and time-consuming design, development, production, and procurement activity.

The Multimission Modular Spacecraft with its payload can either be brought back from space or reserviced on orbit by the Space Shuttle, as desired by the user. This represents a major cost-saving capability unavailable with uniquely integrated spacecraft. In instances where on-orbit repair or refurbishment is not desired, the MMS can be retrieved by the Space Shuttle, returned to Earth for refurbishment or upgrading, and relaunched.

The flight support system that carries the MMS in the Orbiter cargo bay also provides the on-orbit reservicing capability. In addition, this flight support system is versatile enough to be adapted to other types of spacecraft.

7.1 MMS Systems and Capabilities

The basic MMS consists of two major structural subassemblies plus three major subsystem modules, as shown in Figure A.7.1. The module support structure subassembly interfaces with the transition adapter subassembly, and is the central core structure of the MMS. It carries all structural loads imposed by, and all structural and functional interfaces with, the modules. In addition, when the MMS is launched on expendable vehicles, the module support structure carries all launch loads.

The transition adapter provides a standard payload interface to the MMS, provides the interface to the Space Shuttle Orbiter (through appropriate supporting hardware), and provides the capture point interface to the Shuttle remote manipulator system (RMS) for retrieval and on-orbit servicing or return to Earth.

The three major subsystem modules, each having a standard range of performance capabilities, provide communications and data handling, power, and attitude control services. Optional propulsion modules are available as required, and a variety of mission-specific subsystem elements can be added to tailor the capabilities of the MMS to the user's requirements. Examples would include a tape recorder in the command and data-handling module, or additional batteries in the power module. Additional features such as antenna systems

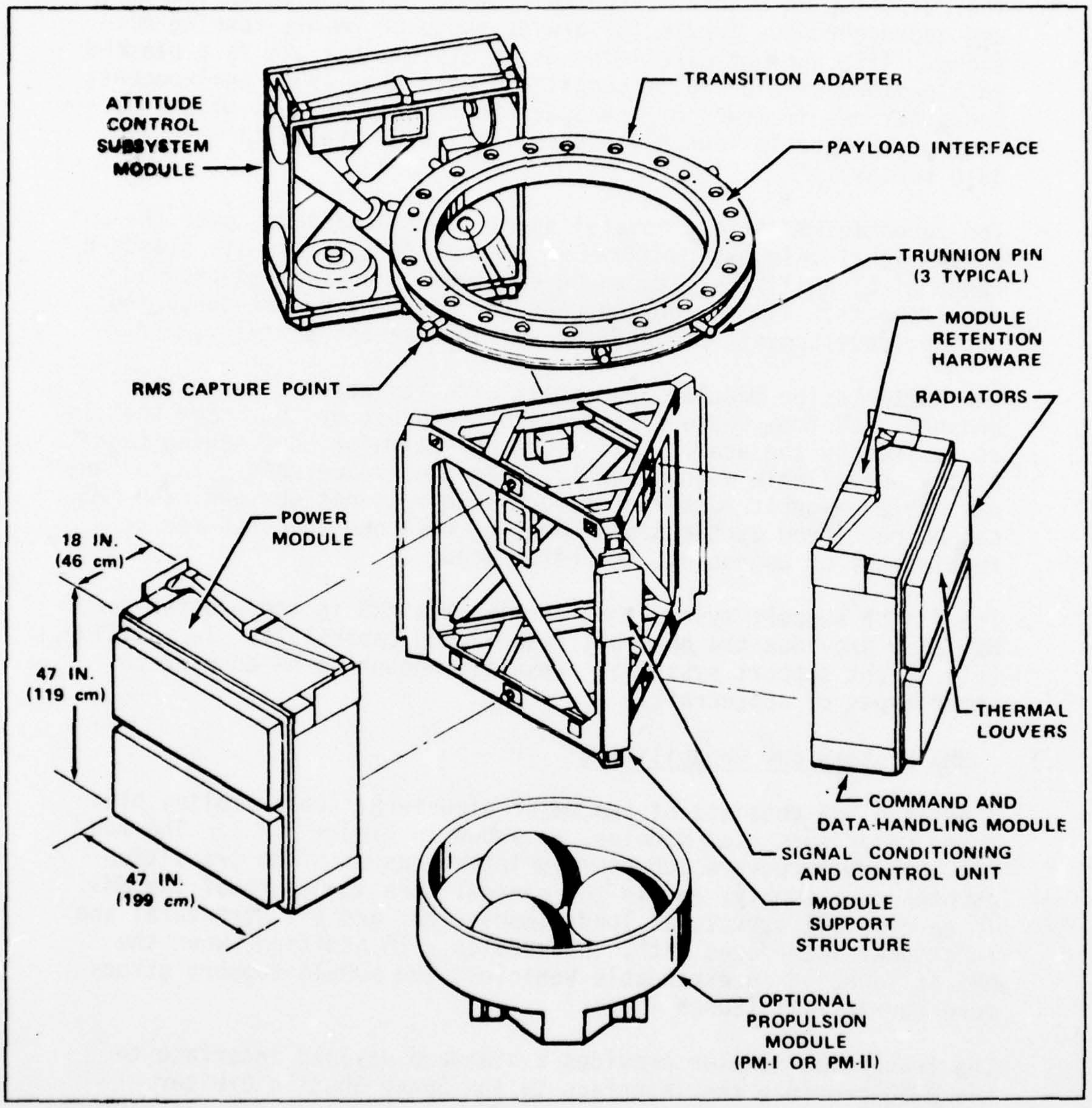


Figure A.7.1. Modular Mechanical Subsystem Components of the MMS and solar arrays, also considered mission-specific, must be supplied by the user. The general capabilities of the MMS are summarized in Table A.7.1.

Table A.7.1. General MMS Capabilities

Payload weight	
For Shuttle launches, in excess of 10 000 pounds (4536 kilograms) limited by payload configuration.	
Orbital capability	
Low-Earth, 270 to 864 nautical miles (500 to 1600 kilometers) (any inclination) and geosynchronous.	
Life expectancy/redundancy	
Minimum life expectancy of 2 years. The MMS is designed to have no single-point failure that would prevent resupply or retrieval by Shuttle	
Subsystem performance capabilities	
Communications and data handling subsystem	
Transponder	S-Band, STDN/TDRSS, transponder output power at antenna port 1.0, 2.5, 5.0 watts
	Prelaunch selectable
Command rates	2 kilobits/sec baseline, 125 to 1 kilobits/sec selectable
Telemetry rates	1, 2, 4, 8, 16, 32 or 64 kilobits/sec
Telemetry formats	2 selectable prior to launch, plus in-orbit programmable capability; all formats contain 692 data word maximum
Onboard computer	18 bits per word, 32 000 words of memory, baseline expandable to 64 000 words
	5 microsecond add time
Payload accommodation	
Maximum remote interface units (RIU) and RIU expanders for experiments	27 units plus 3 expanders per RIU
Command capability per RIU	Eight 16-bit serial magnitude, 62 discrete/relay drivers
Telemetry capability per RIU or RIU expander	64 inputs
	All usable for analog/discrete bilevel, 16 usable for serial digital, 8 bits each
Attitude control subsystem	
Type	3-axis, zero momentum
Attitude reference (without payload sensor)	Stellar (inertial)
Pointing accuracy (one sigma)	
without payload sensor	<0.01°
with payload sensor (ideal)	<0.00001° (direct analog signal processing)
Pointing stability (one sigma)	<0.0001° (signal processing via computer)
Average rate	<0.000001 deg/sec
Jitter	
Without payload sensor	<0.0006° (20 min)
With payload sensor (ideal)	<0.000001° (direct analog signal processing)
	<0.00001° (signal processing via computer)
Slew rate	Based on spacecraft inertia
Power subsystem	
Regulation of load bus	+ 28 ± 7 V dc
Bus noise and ripple	1.5 V P-P (1 to 20 MHz) maximum
Load bus source inpedance	<0.1 ohm (dc to 1 kHz)
	<0.15 ohm (1 kHz to 20 kHz)
	<0.30 ohm (20 kHz to 100 kHz)
Typical load switching transients	± 2 V (50 millisec)
Fault mode transients	Down to 0 V or up to 40 V for 500 millisec
Batteries	Two 20-ampere-hour batteries as baseline and up to three 50 ampere-hour batteries maximum
Power capabilities	200 watts, average; 3000 watts, peak (allowable for 20 min, once per orbit, day or night)
Module temperature range	0 to 40° C (273 to 313 K)
Propulsion subsystem	
Propellant	Hydrazine (MIL-P-26536C, Amendment 1)
Propellant load	
PM - I	167 lb (75.75 kg)
PM - II	1060 lb (480.8 kg)
Pressurant	Gaseous nitrogen
Thrusters	12 at 0.2 lbf (0.9 N); 4 at 5 lbf (22.24 N)
System operating mode	3 to 1 blowdown
Design operating pressure	400 psia (2758 kN/m^2)
Design burst pressure	160 psia (11 032 kN/m^2)
Thermal control	Active and passive
Operating temperature range	10 to 60° C (283 to 333 K)

7.2 Flight Support System

Both for transport to orbit and for servicing and retrieval, the MMS is supported in the Orbiter cargo bay, structurally and functionally, by the flight support system. This system, as shown in Figure A.7.2, consists of four major subsystems: the retention cradle, the payload positioning platform, the module exchange mechanism, and the module magazine.

Each of the four major elements can be operated independently or they can be used collectively as a unified system, depending on the specific mission requirements.

During Shuttle launch and landing, the MMS is carried in the retention cradle, which provides mechanical interfaces to the MMS transition adapter (through which launch and landing loads are transmitted). The retention cradle may be the only element necessary for a launch or retrieval mission (if RMS or spring-ejection deployment is used).

If the mission requires erection out of the cargo bay, to a predetermined position relative to the Orbiter, the payload positioning platform is added to the retention cradle. For deployment, the MMS is erected by the platform to a vertical position. It is grappled by the remote manipulator system, released from the positioning platform, deployed by the RMS and released. For retrieval, these operations are reversed. After the Shuttle establishes rendezvous and stationkeeping with a free-flying MMS, the RMS grapples the spacecraft and berths it onto the erected positioning platform. If the MMS is to be returned to Earth, it is lowered into the retention cradle.

In the case of a servicing mission, the payload positioning platform, module exchange mechanism, and module magazine are required. Replacement modules are carried into space in the module magazine. After the MMS is captured and berthed, the module exchange mechanism replaces the module in the MMS. After the servicing operation is completed and systems are checked out, the MMS is again deployed by the RMS.

The baseline envelope requirement for the retention cradle is the support of two MMS in an over-and-under orientation. Other spacecraft configurations or a complement of mixed spacecraft can be accommodated by use of interface hardware that satisfies the unique spacecraft requirements on one side and adapts to standardized support system fittings on the other side.

The basic retention-cradle core structure is designed to accommodate a single large spacecraft such as the MMS with a large payload; a small spacecraft piggyback with a satellite such as the Intelsat-5; two MMS; an MMS with two smaller spacecraft; or four smaller spacecraft.

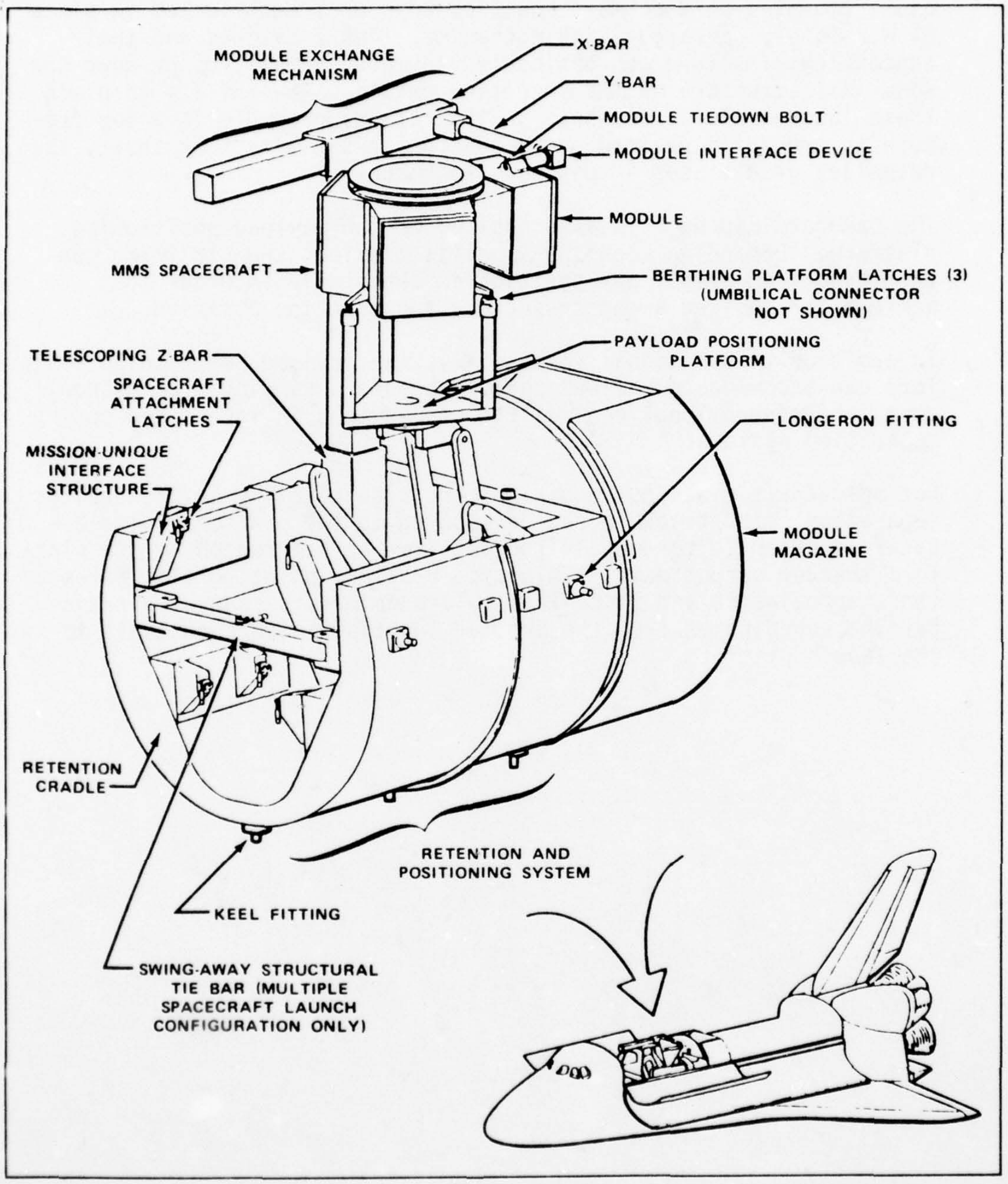


Figure A.7.2. Structural Assemblies of the Basic Flight Support System, Mounted in the Orbiter Cargo Bay

The spacecraft is structurally attached with a set of three trunnions (mounted on the MMS transition adapter), each locked in place by a remotely operated latch mechanism. These latches and their supporting structure can be located forward and aft, up or down the side wall structure of the retention cradle. The ability to place these latches almost anywhere in the retention cradle is a key feature in providing payload accommodation versatility. In short, the retention cradle uses a pegboard approach.

The pegboard approach is also applied to the payload positioning platform. Depending upon the specific mission, the platforms can be hinged from almost any position in the cradle in order to achieve the desired swing trajectory and erection position.

In addition to positional versatility, the payload positioning platform can accommodate various payload adapters ranging from standoff posts and conventional conical structures to spin tables and spring separation systems.

For spacecraft missions requiring precision pointing before Orbiter separation, star trackers can be mounted to the platform. The Orbiter is oriented to the desired position as referenced by the platform tracker output data. This type platform pointing eliminates the thermoelastic and mechanical tolerance errors caused by transferring coordinates from the Orbiter inertial measurement unit to the launch platform.

8.0 STS USER CHARGES

The STP plans, as in the past, to provide flights for DoD experiments without charge to the investigators. However, certain aspects of the STS are sufficiently different from previous experience, that experiment design can significantly affect these charges. For this reason, some parts of the NASA charging policy are included here to help investigators understand these factors and thus make more efficient use of the services provided.

8.1 Basic Charges

The overall objective of the NASA pricing policy is to encourage full use of the Space Transportation System. A key part of this policy is guaranteeing a fixed price during the early years of STS operations: NASA offers this fixed price in contract-year dollars from now through fixcal year 1983 (ending 30 September 1983). After that date, prices will be adjusted annually.

Additionally, the policy results in a price that permits economical transition from existing expendable launch vehicles to the STS. Finally, the STS pricing policy will reimburse NASA the cost to operate the STS, and it has provisions to ensure price stability over the life of the STS Program.

The price for exclusive use of an entire Orbiter (excluding Spacelab, interim upper stage, etc.) depends on the class of users. The price ranges are shown in the accompanying table, in which costs are expressed in constant fiscal year 1975 dollars.

User	Cost $ millions
Private: domestic, foreign	19.0 to 20.9
Foreign government	19.0 to 20.9
Government: U.S. civil, participating foreign	16.1 to 18.0
Department of Defense	12.2
Exceptional program	9 to 12

AD-A064 766 TRW DEFENSE AND SPACE SYSTEMS GROUP REDONDO BEACH CALIF F/G 22/2

STS UTILIZATION STUDY EXPERIMENT ASSESSMENTS.(U)

DEC 77 F04701-77-C-0112

UNCLASSIFIED TRW-31933-6001-RU-00 SAMSO-TR-77-188 NL

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The basic charges are for a dedicated, single-user Shuttle flight. With NASA approval, a user of a dedicated flight may apportion and sublet STS services to other users, provided those users satisfy STS requirements. The price of integrating additional payloads will be negotiated.

For a payload that will not require an entire flight capability and that can share the cargo bay with others, the cost to the user will be a fraction of the dedicated-flight price, calculated as follows.

1. The payload weight is divided by the Shuttle payload weight capability at the desired inclination to find the weight load factor. The figures shown as examples are for a 160-nautical-mile (296-kilometer) orbit.

Inclination, deg	Weight capability lb (kg)
28.5	65 000 (29 484)
56	57 000 (25 855)
90	37 000 (16 783)
104	30 000 (13 608)

2. The payload length is divided by the length of the cargo bay, 60 feet (18.29 meters), to find the length load factor.
3. The load factor (length or weight, whichever is greater) is divided by 0.75 to determine the cost factor.
4. The calculated cost factor is multiplied by the price of a dedicated Shuttle flight (for the user's class) to determine the price for that payload.

In Table A.8.1 are listed those services that are considered by NASA as standard Shuttle services included in the basic charges and those for which additional charges are made to all users except NASA and DoD. They are included here as an indication of the basic thinking that has been used to develop the charging policy.

TABLE A.8.1

STANDARD SHUTTLE SERVICES	OPTIONAL SHUTTLE SERVICES
• Two standard mission destinations (1) 160 NM Altitude: 28.5° Inclination. (2) 160 NM Altitude; 56.0° Inclination. • One day mission operations • Orbiter flight planning services • Transmission of payload data to compatible receiving stations • A three man flight crew • On-orbit payload handling • Deployment of a free flyer • NASA support of payload design reviews • Prelaunch payload installation, verification and Orbiter compatibility testing • NASA payload safety review	• Revisit and retrieval • Use of Spacelab or other special equipment • Use of Mission Kits to extend basic Orbiter capability • Use of Upper Stages • EVA services • Unique payload/orbiter integration and test • Payload mission planning services, other than for launch, deployment and entry phases • Additional time on-orbit • Payload data processing • Launch from Western Test Range Two standard mission destinations are available from the Western Test range site: (1) 160 NM Altitude; 90.0° inclination. (2) 160 NM Altitude; 104.0° inclination.

8.2 STS Charges to DoD

In a NASA/DoD agreement of 7 March 1977, a price of $12.2M (in FY 1975 dollars) per Shuttle flight, was established. This price will remain fixed for the first six years of Shuttle operation.

It takes into consideration "the programmatic, operational, and technical services uncertainties in providing STS launch services..." As such, it is expected to cover all DoD flight activities, regardless of complexity, for the six years.

STP expects to make use of dedicated DoD flights as much as possible. However, on occasion, it will be necessary to share flights with NASA and other users. For such shared flights, specific negotiations will be entered into. These negotiations will reflect the services required and will be influenced by the factors in Table A.8.1.

8.3 Investigator Considerations

Although DoD investigators are not directly charged for launch services, there are certain things that they can do to increase the efficiency of DoD flight experimentation. Following are some factors that should be considered early in the development phase of an experiment.

8.3.1 Form Factor

Consideration of the nominal Shuttle launch load of 65,000 lbs., together with the payload bay length of 60 ft. can show that form factor in experiment equipment, and its support hardware are important factors. For the normal density of electro-mechanical equipment, the length of payload bay that is used is much more important to cargo efficiency than is the mass of individual items. Efficient payload designs make maximum use of the payload bay width and minimum use of its length.

8.3.2 STS Interfaces

Direct interfacing with the STS, either physically or operationally, can cause a burden on investigators. This is partly because the STS requires explicit proof of interface compatibility. It also comes about because Orbiter services to users are fixed and to some extent, lack flexibility. For complex experiments, it is advisable to consider use of standard interfacing systems such as Spacelab or the Standard Test Rack. These payload carriers, in addition to affecting the interface with Shuttle, can provide services that otherwise might have to be built into the instruments.

8.3.3 Standard Equipment

As the many Shuttle payloads approach maturity, more and more equipment is being developed for multipurpose application. The pointing and stabilization systems described in Section 5 of this appendix, are examples. Support equipment is being developed for the LDEF and the systems of the MMS also can have many users. Experimenters are well advised to explore the use of these, and others that are evolving, as an aid to reduction of direct development costs. In addition, these standard equipment items will become well accepted in respect to this interfacing with the STS, thus reducing integration costs and time.

APPENDIX B

THE ROLE OF THE SPACE TEST PROGRAM

THE ROLE OF THE SPACE TEST PROGRAM

The Space Test Program (STP) provides a means for Department of Defense personnel and their university and industry associates to place RDT&E payloads into space. The authority and charter for STP is found in the tri-service manuals AFM80-2, AR70-43, and OPNAV76P-2. The U.S. Air Force is designated the executive agent for the implementation of the program.

Although each mission has its special agreements, in general, STP manages the integration of the selected experiments and procures appropriate hardware and software to support their on-orbit activities. Additionally, STP will arrange and/or fund for the launch vehicle, launch operations, upper stages, spacecraft, integration and data collection as necessary to support the mission. The costs for unusual requirements, payload withdrawals, and specialized experiment hardware may be borne by the experimenter or sponsor. These decisions must necessarily be handled on a case by case basis. The sponsorship and funding for the experiment must come from an agency outside STP.

The general procedure and sequence of events can best be described by referring to the time line, Figure B-1. Due to the usual need for experimental data to support certain DoD requirements or objectives, it is most important that a dialogue be initiated with the plans division of STP. In some cases, the STP experiment file can be opened with a phone call (213) 643-1121 or AV 833-1121. But, it is preferred that a letter or a draft of a DD Form 1721 be submitted to STP directly so that the dialogue can begin. This informal submission can be helpful in several ways:

(1) STP can provide guidance or suggestions in the completion of the 1721 form.

(2) STP can obtain advance information on future experiments and requirements.

(3) STP can track the 1721 on its approval cycle and can sometimes help reduce delays.

STP requests that the forms submitted to the plans office be classified no higher than SECRET. Higher classification can be handled by special arrangement.

SPACE TEST PROGRAM TIME LINE

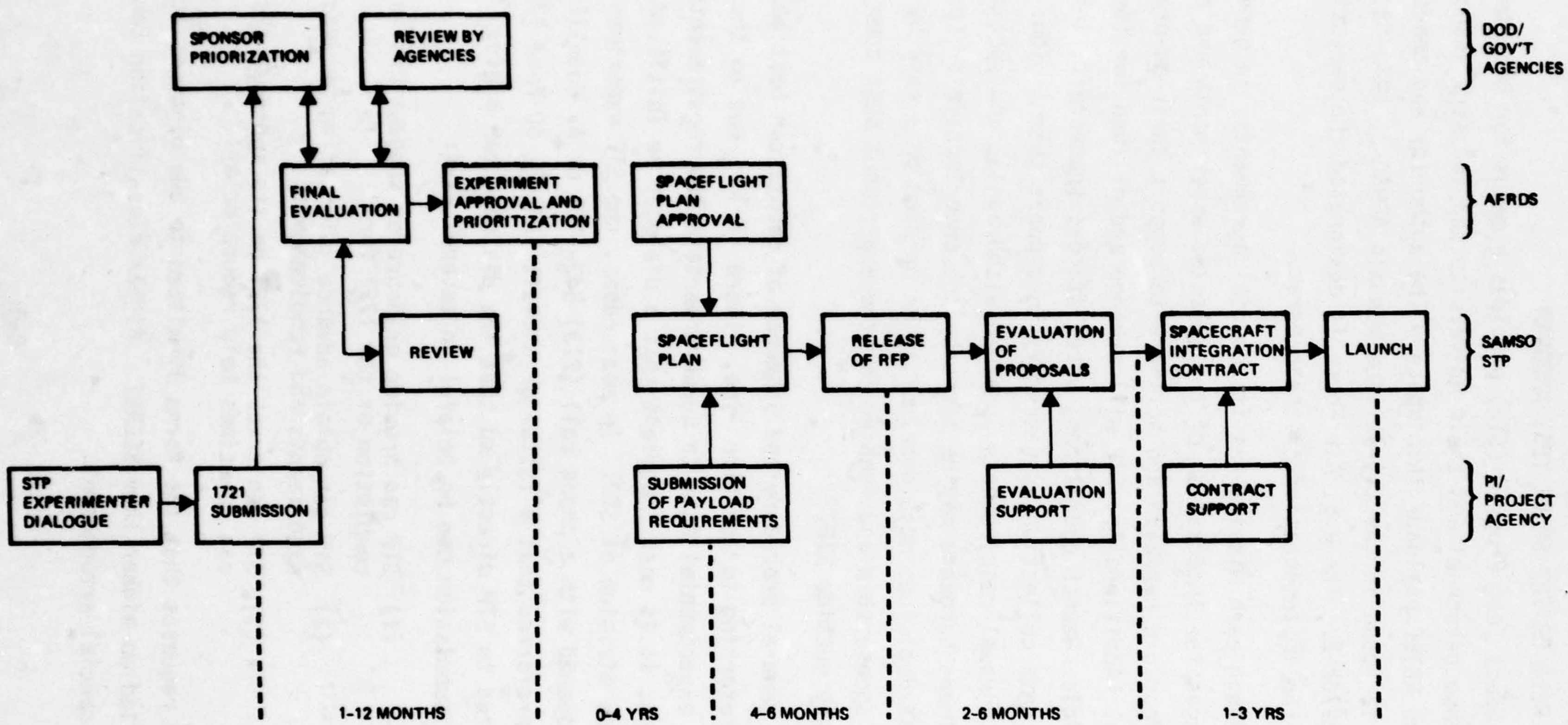

Figure B-1

In order to minimize the time between the initial 1721 submission and the other steps shown in the Program Time-Line, additional efforts are required by the Sponsoring Agency. Discussions with and briefings to DDR&E, and AFRDS regarding the relevance or time-critical nature of the experiment may be desirable. In general, the factors which enhance the visibility of the experiment can aid greatly in circumventing delays in the approval cycle. The experiments are prioritized by AFRDS according to their flight urgency, mission relevance and program importance. Those which obtain the highest ratings are designated priority I and consequently will be planned for flight as soon as possible. The priority II and III experiments are usually launched in conjunction with one or more priority I experiments. Those that are picked must be compatible with the mission drivers. However, as more frequent and diverse flight opportunities are offered by the Shuttle, it is envisioned that priority II and III experiments will be flown more often.

Your Request for Spaceflight (DD Form 1721) is submitted through military channels for eventual approval and prioritization by the DoD experiments review board. STP is only peripherally involved in the approval process, i.e., an experiment compatibility analysis is usually given to advise the board on technical suitability for spaceflight, previous efforts, etc.

Once your experiment is selected for spaceflight, STP personnel will require more detailed information. This process is initiated through a "payload requirements questionnaire," completed by the experimenter. This document is the basis for the development of the appropriate annexes in the request for proposal (RFP) for the integrating contract. Of course, the dialogue continues.

Experimenters are also asked to assist STP in the proposal evaluation phase and continue in this role for the duration of the program.

Following are key personnel at STP who can be contacted regarding experiment submission, operating procedures, flight opportunities, etc.

Col. C. Zimmerman/YCT
Headquarters SAMSO
Worldway Postal Center
Los Angeles, CA 90009
(213) 643-0840
AV (213) 833-0520

Maj. C. S. Jund/YCT
Headquarters SAMSO
Worldway Postal Center
Los Angeles, CA 90009
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Mail Station 125/1273
Los Angeles, CA 90009
(213) 648-7136

APPENDIX C

LIST OF ACRONYMS

ACRONYMS

AFAPL	Air Force Applied Physics Laboratory
AFAL	Air Force Avionics Laboratory
AFESD	Air Force Electronics Systems Division
AFFDL	Air Force Flight Dynamics Laboratory
AFGL	Air Force Geophysical Laboratory
AFML	Air Force Materials Laboratory
AFOSR	Air Force Office of Scientific Research
AFRPL	Air Force Rocket Propulsion Laboratory
AFSCF	Air Force Satellite Control Facility
AFWL	Air Force Weapons Laboratory
AMPS	Atmosphere, Magnetosphere, and Plasmas in Space (NASA Program)
APP	Astrophysics Payloads
APS	Auxiliary Power Subsystem
APU	Auxiliary Power Unit
ASE	Airborne Support Equipment
ASPS	Annular Suspension Pointing System
C&W	Caution and Warning
CDMS	Communications and Data Management System (Spacelab)
CITE	Cargo Integration Test Equipment
CNO	Chief of Naval Operations
CTS	Communications Technology Satellite
DARCOM	Army Development and Research Command
DMA	Defense Mapping Agency
DNA	Defense Nuclear Agency
DOD	Department of Defense
DSN	Deep Space Network
ECOM	Army Electronics Command
ECLSS	Environmental Control and Life Support System (Orbiter)
ECS	Environmental Control System
EMI	Electro-Magnetic Interference
EPDS	Electrical Power and Data System (LDEF)
ESA	European Space Agency
EVA	Extravehicular Activity
GSE	Ground-Support Equipment

GSFC	Goddard Space Flight Center
HRDR	High Rate Digital Recorder
HRM	High Rate Multiplexer
IMU	Inertial Measurement Unit
IPS	Instrument Pointing Subsystem
IUS	Interim Upper Stage
JPL	Jet Propulsion Laboratory
JSC	Lyndon B. Johnson Space Center
KSC	John F. Kennedy Space Center
LaRC	Langley Research Center
LDEF	Long Duration Exposure Facility
MCC	Mission Control Center (at JSC)
MICOM	Army Missile Command
MMS	Multimission Modular Spacecraft
MPS	Materials Processing in Space (NASA Program)
MSFC	Marshall Space Flight Center
NA	Not Applicable
NASA	National Aeronautics and Space Administration
NASCOM	NASA Communications Network
NASC	Naval Air Systems Command
NEL	Naval Electronics Laboratory
NESC	Naval Electronics Systems Command
NRL	Naval Research Laboratory
NSRDC	Naval Ship Research and Development Center
NSSC	Naval Sea Systems Command
NSWC	Naval Surface Weapons Center
OFT	Orbiter Flight Test
OMS	Orbital Maneuvering Subsystem (Orbiter)
ONR	Office of Naval Research
OPF	Orbiter Processing Facility (at KSC)
POCC	Payload Operations Control Center
POP	Perpendicular to Orbit Plane
RADC	Rome Air Development Center
RAU	Remote Acquisition Unit (Spacelab)
RCS	Reaction Control Subsystem (Orbiter)

RIU	Remote Interface Unit (Orbiter)
RMS	Remote Manipulator System
SAMSO	Space and Missile Systems Organization
SIPS	Small Instrument Pointing System
SSUS	Spinning Solid Upper Stage
SSUS-A	Spinning Solid Upper Stage for Atlas-Centaur Class Spacecraft
SSUS-D	Spinning Solid Upper Stage for Delta Class Spacecraft
STDN	Space Tracking and Data Network
STP	Space Test Program (DOD)
STR	Standard Test Rack
STS	Space Transportation System
TBD	To Be Determined
TDRSS	Tracking and Data Relay Satellite System
VAB	Vehicle Assembly Building (at KSC)
VAFB	Vandenberg Air Force Base
VECC	Vacuum Exposure Control Canister (LDEF)

APPENDIX D

DESCRIPTION OF NASA MATERIALS PROCESSING IN SPACE PROGRAM

MATERIALS PROCESSING IN SPACE PROGRAM APPROACH

- Early demonstrations of unique effects achievable in space
  - Use of all feasible flight opportunities
  - Diversified experiment program
- Concentration on areas related to high value applications
- Broad participation from potential user community
  - Experiment program defined by user proposals
  - Multiple experiments in general purpose apparatus
- Economical approaches to space experimentation
  - Minimum combined capital and operating
  - Low unit costs achieved through high productivity
- Encouragement of early privately funded activity
  - Cost levels appropriate for industrial R&D
  - Proprietary rights in experiment data

MATERIALS PROCESSING IN SPACE PROGRAM GOALS

- THE NASA PROGRAM GOALS BROADLY STATED REFLECT AN APPLICATIONS EMPHASIS BUILDING FROM BASIC INVESTIGATIONS IN SPACE.

1. DEVELOP UNDERSTANDING OF FUNDAMENTAL PROCESSES AND PROPERTIES OF MATERIALS THROUGH SCIENTIFIC UTILIZATION OF SPACE.
2. DEMONSTRATE THE VALUE OF SPACE FOR MATERIALS WORK BY DEVELOPING PRODUCTS AND/OR MATERIALS.
3. INITIATE USER-SPONSORED UTILIZATION OF SPACE FOR RESEARCH IN MATERIALS SCIENCE AND TECHNOLOGY.
4. REALIZE PRIVATELY-FUNDED ORBITAL MANUFACTURING OPERATIONS.

The uniqueness that going into space provides to the astronomer and to the meteorologist is well recognized; space offers a dramatic platform from which the universe can be observed better than on Earth, or the Earth can be observed as an integrated entity. But very few people truly understand a similar dramatic advantage to the materials scientist. By removing gravity one gains an ability to observe basic material forces at work.

The presence of a low gravity environment is the most unique feature for space processing of materials. The goal of the technical community in utilizing this low-g environment is to develop techniques that will enable future process development, of an economical nature, to proceed. The benefits to materials research and the effects on process equipment requirements, resulting from the low-g environment, occur in two ways.

First, the condition of low-g will mean that only very small forces will be required to maintain position control of the material sample. Quasi-static displacement or manipulations may similarily be accomplished by imposing very small forces. Electromagnetics, electrostatic or acoustic fields of relatively low intensity will provide restraining forces of sufficient magnitude to levitate a melt without the imposition of large power inputs to the sample. The capability of such types of levitated melting can have a prominent effect on many of the research process areas. Containers will be unnecessary and impurities from containment vessels for highly reactive materials of high melting points will be eliminated. This will impact such process areas as crystal growth and glass preparation. In the case of laser glass and glass for laser system optics, for example, crucible contamination is probably the most serious limitation to the state of the art. In the case of crystal growth, a reduction in the amount of impurities in crystals grown for semiconductor applications could have far-reaching effects. Elimination of the containment vessel will preclude crucible surface irregularities which are always present and which can produce undesired nucleation sites. Containerless melting will also increase the flexibility of heating arrangements, and this could result in beneficial temperature distributions not possible on Earth.

The second major advantage of low-g processing is the elimination of buoyancy effects which result in convection currents caused by thermally induced density variations in a liquid. Differences in densities among several materials in a mixture, which result in segregation effects on Earth, may also be eliminated. Reduction in convection currents will allow for much better control of heat and mass transport in liquids and gases. Improved processes and products may result from such control.

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WHY SPACE FOR MATERIALS RESEARCH

WHILE SPACE PROVIDES UNLIMITED VACUUM PUMPING AND UNFILTERED SOLAR RADIATION, THE LONG DURATION NEAR WEIGHTLESSNESS OF FLIGHT IS THE PRIME FEATURE OFFERING DISTINCT PROCESSING DIFFERENCE FROM GROUND LABS.

WHY SPACE?	EXAMPLES OF THE USES OF THESE ADVANTAGES
SPACE OFFERS FREEDOM FROM:	
• GRAVITY INDUCED CONVECTION	• REDUCTION OF UNWANTED CONVECTION WHICH CAUSES INHOMOGENEITIES
• CONTAINER STRESS AND CONTAMINATION	• FABRICATION OF HIGH TEMPERATURE COMPONENTS WITHOUT CONTAINER-INDUCED CONTAMINATION
• PLASTIC DEFORMATION AT HIGH TEMPERATURES	• FABRICATION OF MORE PERFECT CRYSTAL METALS WITHOUT CLASSIC DEFORMATION-INDUCED DEFECTS
• BUOYANCY SEPARATION OF SUSPENSIONS	• PURIFICATION OF KEY BIOLOGICAL AND RELATED MATERIALS

In preparing for a broad program of low-g materials processing experiments, five basic types of processes are envisioned as described below.

- Crystal Growth. There are three broad categories of crystal growth that are considered most conducive to in-space processing: growth from a melt, growth in solution and growth from a vapor phase. The experimental procedures involved will be strongly dependent upon the problems of positioning, stirring and shaping the melts and solutions under weightless conditions.

- Purification/Separation. This area will benefit from in-space processing because of greatly reduced buoyancy and convective effects. The production of super-pure materials becomes possible when one can use high temperatures, ultrahigh vacuum and containerless samples — especially in multipass, molten-zone refining of ultrapure elements. Also included in this process area are low temperature separations of biological materials such as living cells, serums, vaccines and other macromolecular materials of potential medical or pharmaceutical utility.

- Mixing. This process area includes procedures where homegenization of materials is a problem on Earth because of density differences that cause segregation problems upon solidification. This is apparent in two specific areas: immiscible materials and composite materials. On Earth, inhomogeneities are caused by variations in density, compatibility and surface tension between the separate components.

- Solidifications. There are three areas of investigation included in this category: controlled or directionally solidified eutectic structures; preparation of glasses; supercooling and homogeneous nucleation.

- Processes in Fluids. This area consists of two types of processes as they occur under weightless conditions: chemical processes which are concerned with reactions and reaction rates, and physical and thermodynamic phenomena (not changes of state or compositions). The condition of very low gravity will permit evaluations that have never before been possible in these fields.

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MPS EXPERIMENT CLASSES

FOUR BROAD CLASSES OF MATERIALS INVESTIGATIONS HAVE EVOLVED SO FAR. THE ELECTRONIC AND OPTICAL MATERIALS CLASSES REPRESENT ARENAS OF GREATEST APPLICABILITY.

CLASS

BIOLOGICAL PREPARATIONS	SOLIDIFICATION OF METALS, ALLOYS COMPOSITES, AND ELECTRONIC MATERIALS	CERAMICS AND GLASSES	PHYSICAL OR CHEMICAL PROPERTIES OF MATERIALS
• SEPARATION OF PROTEINS	• CRYSTAL GROWTH	• CERAMIC COMPOSITIONS	• BUBBLE NUCLEATION
• SEPARATION OF CELLS	• UNIQUE ALLOYS	• CERAMIC MATERIAL PURIFICATIONS	• DEPRESSION OF PARTICLES
• DIALYSIS	• COMPOSITES	• OPTICS	• SURFACE TENSION
• GROWTH	• HOMOGENEOUS SOLIDS	• WAVE GUIDES	• MASS TRANSPORT
•	• THIN FILMS	• SEMICONDUCTORS	• CHEMICAL KENETICS
	•	• SPECIAL GLASSES	• POLYMER RESEARCH
		•	• HETEROGENEOUS CATALYSTS

INITIAL STS/MPS PLANS

- Shuttle payload capabilities will be built-up starting with FY 78 and supplemented by every other FY new start requests. Facilities will be added or expanded and reflight supported with the later new starts.

- The MPS FY 78 new start will develop 4 or 5 facilities: bio-processing, multi-purpose fluid phenomenon, multi-furnace, float zone crystal growth and containerless processing, and support their flight for the first of many contemplated times.

- NASA operating with a constrained budget has selected 20-30 investigations for initial development out of over 140 submitted earlier this year. More will be added each year.

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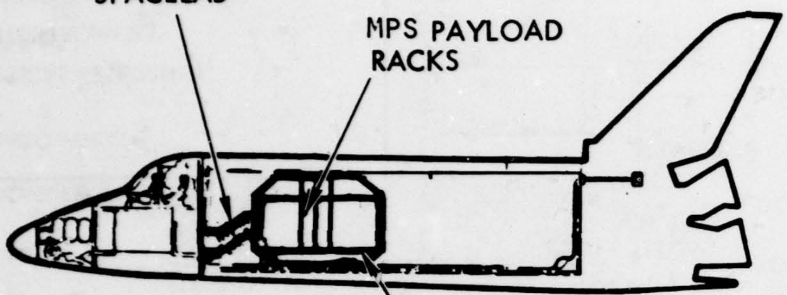
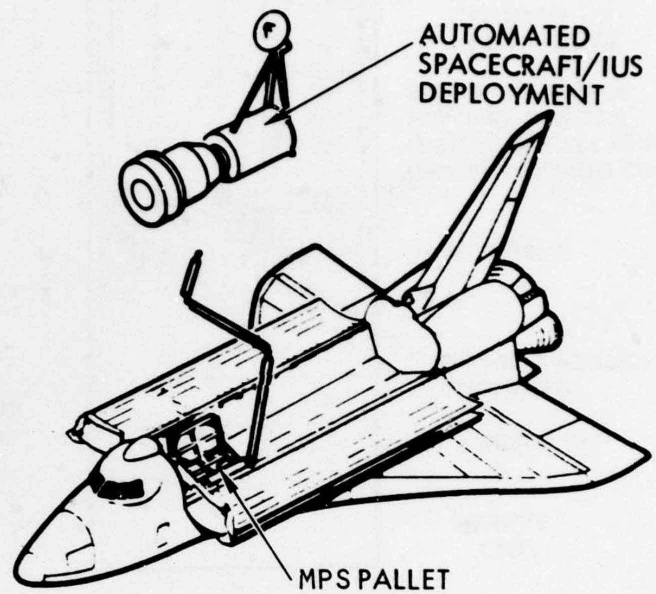
MATERIALS PROCESSING REFERENCE FLIGHTS

RACK MOUNTED FACILITIES



- SPACELAB FLIGHT 3, SHUTTLE FLIGHT 11
- APPROXIMATE LAUNCH DATE: 31 JAN 1981
- MPS FLIGHT MODE: RACKS IN SPACELAB PRESSURIZED MODULE
- MPS EXPERIMENT FACILITIES: BIOLOGICAL, MULTI PURPOSE FLUIDS
- CO PASSENGERS: ACPL DROP DYNAMICS
- ALT/INCL: 370 KM/28.5 DEG

PALLET MOUNTED FACILITIES



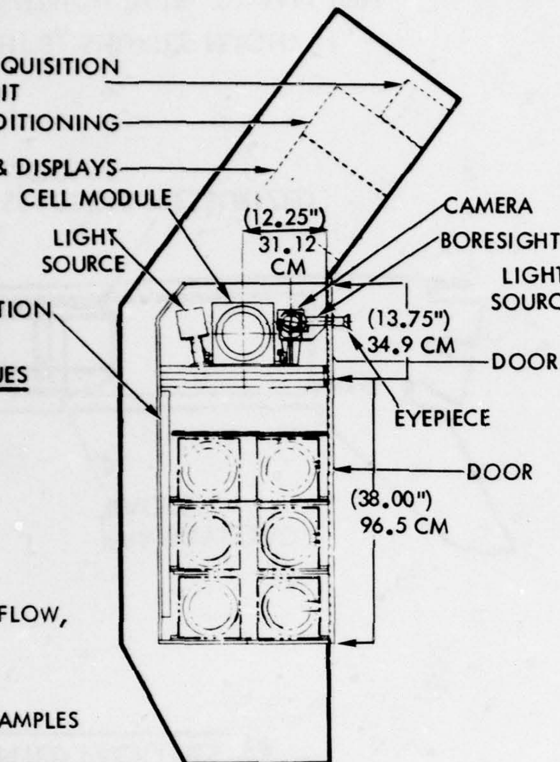
- SHUTTLE FLIGHT 16
- APPROXIMATE LAUNCH DATE: JUNE 1981
- FLIGHT MODE: UNPRESSURIZED PALLET STRUCTURE IN SHUTTLE BAY
- MPS EXPERIMENT FACILITIES: MULTI FURNACE, FLOAT ZONE REFINE/CRYSTAL GROWTH, CONTAINERLESS PROCESSING
- CO PASSENGERS: NONE, AFTER SATELLITE IS DEPLOYED (FOREIGN COMSAT)

ALT/INCL: TBD

MULTIPURPOSE FLUID PHENOMENA FACILITY

DESCRIPTION

- A FACILITY TO MEASURE AND OBSERVE FLUID PHENOMENA, CRYSTAL GROWTH, BUBBLE DYNAMICS, AND PARTICLES IN MICROGRAVITY

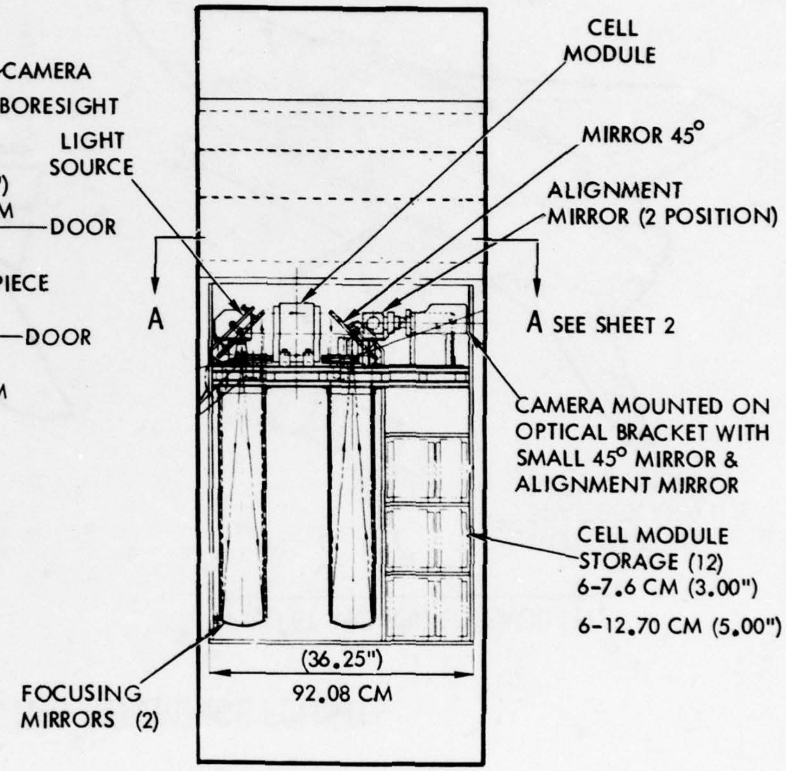


MEASUREMENT TECHNIQUES

1) SHADOWGRAPH
2) OPTICAL ABSORPTION
3) INTERFEROMETRY
4) SCHLIEREN
5) DIRECT MEASUREMENT OF FLOW, TEMPERATURE

INVESTIGATION EXAMPLES

- CRYSTAL GROWTH RATE, BUBBLE DYNAMICS
- BUBBLE MIGRATION
- CONCENTRATION PROFILES IN EMULSIONS
- TEMPERATURE GRADIENT PROFILES
- PROFILES OF TEMPERATURE, OR FLUID DENSITY
- HEAT AND MASS TRANSPORT EXPERIMENTS



VIEW WITH DOORS OPEN

MULTI-PURPOSE FURNACE INSERT

Description

- This facility contains apparatus suitable for material processing up to 2200 C

Samples Accommodated Include:

- Encapsulated samples
- Sting-mounted samples

Research Categories Supported Include:

- Heterogeneous nucleation in glasses
- Complex glass formation
- Directional solidification of ceramic compositions
- Crystal growth by chemical vapor transport
- Bridgman crystal growth
- Flux crystal growth
- Liquid phase sintering
- Controlled solidification
- Molten zones in microgravity
- Directional solidification

Processing Atmospheres:

- Air, oxygen, nitrogen, argon, oxygen-free argon - 0.1 n/m^2 to 4×10^5 n/m^2 (hydrogen later?)

Vacuum Purging for Process Atmosphere Filling:

- 0.1 n/m^2 venting to space
- 1.3×10^{-4} n/m^2 with turbopump

Vacuum Processing:

- Same as purge

Range of Specimen Sizes:

- Up to 6 cm diameter x 25 cm long

Electrical Power:

- 1 kw/3 kw

Processing Temperatures:

- Isothermal 700° to 2200°C (.1°C/cm over 6 cm) gradients (max 300°C/cm over 6 cm)

Operating Time/Specimen:

- Up to 5 days (7200 minutes)

Accelerations:

- Depends upon shuttle movement

Number of Specimens/Mission:

- 4 to 160

FLOAT ZONE REFINER/CRYSTAL GROWTH INSERT

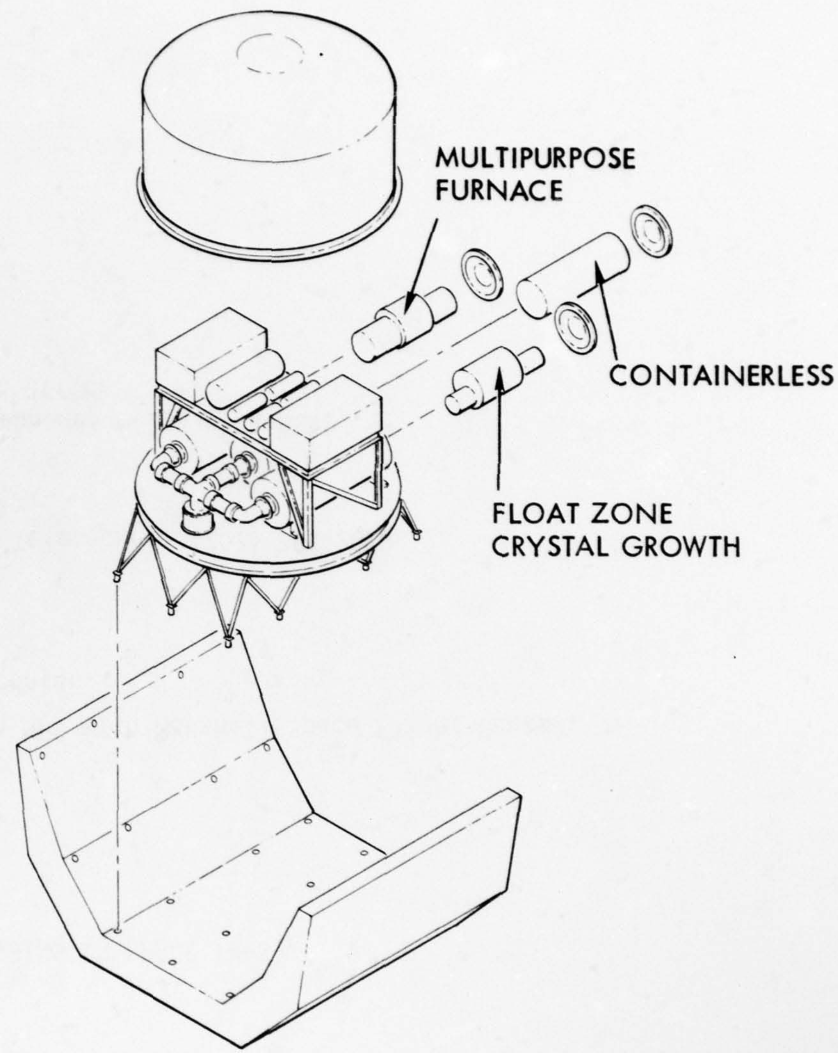
Processing atmospheres: Air, oxygen, nitrogen, argon, helium (hydrogen later)

Processing atmosphere pressures:		$0.1\ N/M^2$ to $4 \times 10^5\ N/M^2$	
Vacuum purging and processing:		$0.1\ N/M^2$ venting to space $1 \times 10^{-4}\ N/M^2$ with turbopump	
Range of specimen sizes:		1 to 2 cm diameter 10 to 35 cm long	Float zone refiner
		1 to 2 cm diameter 1 to 10 cm long	Crystal growth using seed
Electrical power:		1 to 2 kw nominal	
Processing temperatures:		100°C to 1700°C	
Processing time/specimen:	Crystal growth:	Sweep 1 MM/hr x 25 MM = 300 minutes Sweep 5 MM/hr x 100 MM = 1200 minutes	
	Zone refining:	Sweep 5 MM/hr x 250 MM = 3000 minutes Sweep 1 MM/hr x 120 MM = 5 day mission	
Sweep rates for heater:		1 MM/hr 5 MM/hr 10 MM/second (fast return sweep)	
Specimen rotation rates:		1 RPM to 60 RPM in steps - TBD Seed and rod turn in same or opposite direction	

MODULAR PROCESSING FACILITY (PALLET MODE)

DEFINITION:

PROVIDES BASIC ENVIRONMENTAL ENCLOSURES AND SUPPORTING STRUCTURAL, MEASUREMENT, GAS AND CONTROL SUBSYSTEMS. REQUIRES ADDITION OF MODULE INSERTS: FURNACE, CRYSTAL GROWTH OR CONTAINERLESS TO COMPLETE EACH RESPECTIVE CAPABILITY. (MAX WT. ~ 6000 LBS/OPERATES ON 4-5 KW MAX.)



MATERIALS PROCESSING IN SPACE (MPS)

- NASA Office of Applications is directing MPS with Marshall Space Flight Center, Huntsville, Alabama acting as the lead center.

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