NASA Contractor Report 178153

CONCEPT DEFINITION FOR SPACE STATION
TECHNOLOGY DEVELOPMENT EXPERIMENTS

EXPERIMENT DEFINITION

(NASA-CR-178153) CONCEPT DEFINITION FOR SPACE STATION TECHNOLOGY DEVELOPMENT EXPERIMENTS. EXPERIMENT DEFINITION, TASK 2 Final Report (Research Triangle Inst.) 285 p CSCL 22B G3/15

N87-16860

Unclas 43981

RESEARCH TRIANGLE INSTITUTE
Center for Systems Engineering
P. O. Box 12194
Research Triangle Park, North Carolina

Contract NAS1-17639
April 1986

National Aeronautics and
Space Administration

Langley Research Center
Hampton, Virginia 23665

TABLE OF CONTENTS

LIST OF FIGURES

LIST OF FIGURES

(Continued)

LIST OF FIGURES

(Concluded)

LIST OF TABLES

LIST OF TABLES

(Continued)

1.0 INTRODUCTION

This report documents the second task of a study with the overall objective of providing a conceptual definition of the Technology Development Mission Experiments proposed by LaRC on Space Station. During this task, the information (goals, objectives, and experiment functional description) assembled on a previous task was translated into the actual experiment definition. Although still of a preliminary nature, aspects such as: environment, sensors, data acquisition, communications, handling, control, telemetry requirements, crew activities, etc., were addressed. Sketches, diagrams, block diagrams, and timeline analyses of crew activities are included where appropriate. An attempt at a uniform format is intended, however, this was relaxed in keeping with the objective and content of each experiment.

2.0 TDMX 2011 - SPACECRAFT MATERIALS AND COATINGS

CONTACT: Dr. D. R. Tenney/W. S. Slemp, MS 188/183
NASA LaRC (804) 865-2361/3420

2.1 GOALS AND OBJECTIVES

The goal of this experiment is to establish a capability for testing the response of advanced materials and coatings to long-term exposure in a space environment. This will be accomplished with the construction and implementation of a flexible, long life exposure facility for materials testing and evaluation. The results of subsequent experiments which utilize this facility will establish the conditions under which selected materials and coatings should be applied in future space missions. The exposure facility addressed in this experiment will be defined in such a manner as to accomodate the characterization of optical, thermal, and mechanical property changes for various exposure conditions. Emphasis will be placed upon obtaining data for the identification of damage mechanisms and formulation of degradation models to guide future materials development.

2.2 DESCRIPTION

As space systems become larger, more complex, and more costly, they will require much longer lifetimes in space to become economically feasible. This requires that structural materials be available that can perform for much longer lifetimes than those required for current spacecraft. It can be assumed that electrical or electronic systems may be replaced or repaired, but the structure should generally be maintenance free for the duration of these missions. Although ground laboratory testing programs are in progress, these programs are substantially impaired by lack of information on the effects of the total space environment on the properties of materials and coatings.

The data base that is to be generated as a result of this experimental capability will provide the information required to determine the applicability of candidate materials and coatings to various space missions. The proposed experiment will provide a unique opportunity to develop a long-term space environmental test facility for material studies. This capability will provide a durability data base on advanced thermal control coatings, adhesives, composites, and polymer films.

The test facility will consist of three panels mounted to the Space Station structure to be used for sample attachment. One potential layout of panels on the Space Station is shown in Figure 2-1. The method for constructing the test facility and the execution of experiments will utilize those experiences that have been gained from Long Duration Exposure Facility (LDEF) activities. Additional activities such as placement of the specimen tray and its recovery, experiment control, and data acquisition will incorporate the increased operational flexibility allowed by Space Station.

Specific experiments would be developed to evaluate the effects of each exposure parameter (e.g., atomic oxygen, solar radiation, temperature, solar flare), both individually and combined, on the properties of these materials. Each sample would be contained within passive trays such as those used on LDEF. However, two or three trays would be instrumented with equipment to monitor environmental exposure parameters, optical properties, or weight loss. Figure 2-2 shows a passive tray typical of that flown on LDEF containing material samples which would be monitored and returned to Earth for extensive testing. A diagram of a single panel showing the location of sample trays is shown in Figure 2-3.

Three panels attached to the Space Station structure would provide the space required for sample placement. These three panels would be located on: (1) the truss structure facing the velocity direction; (2) the truss structure facing the wake direction; and (3) the sun-facing solar array truss structure. The panels would be installed by the Remote Manipulator System (RMS) with extravehicular activity (EVA) required to complete power and electrical connections. The velocity facing panel would contain six

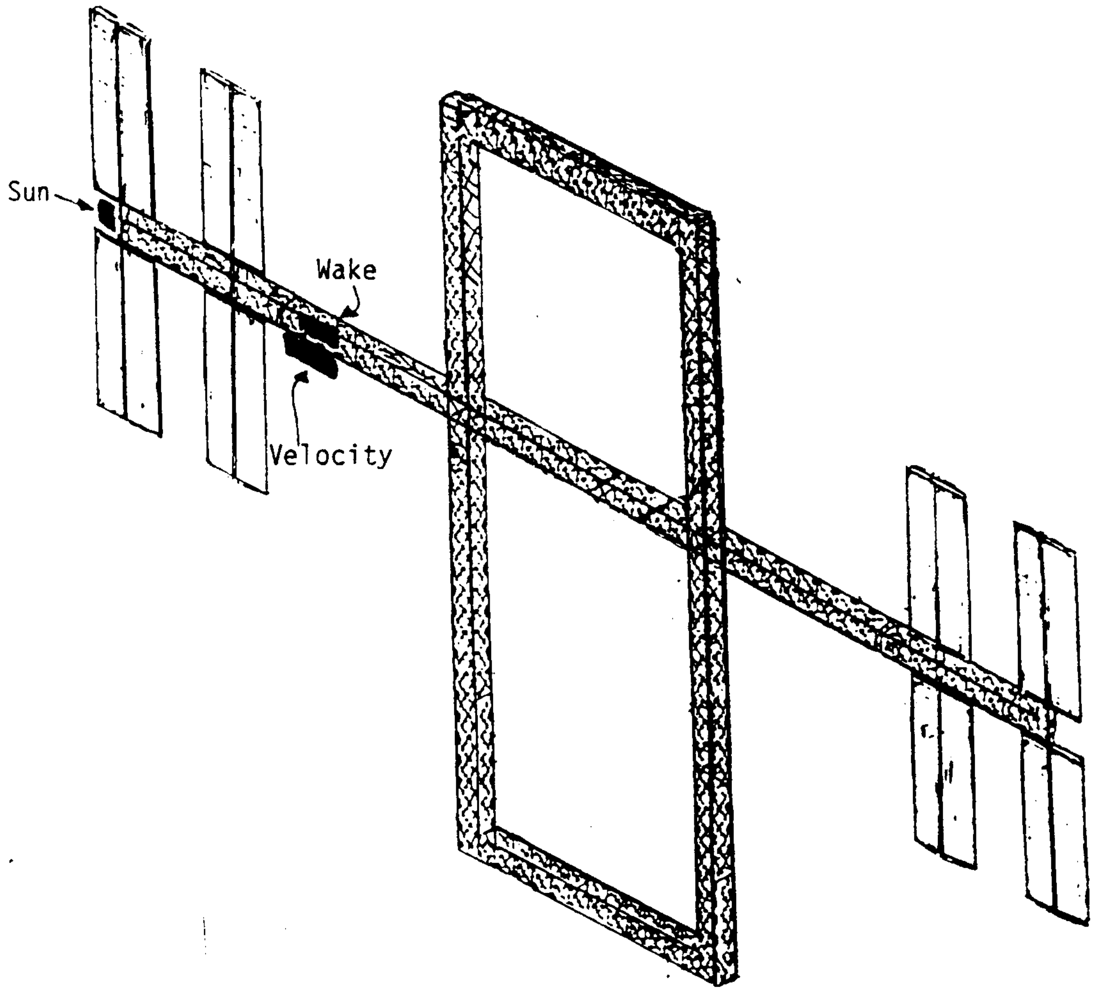
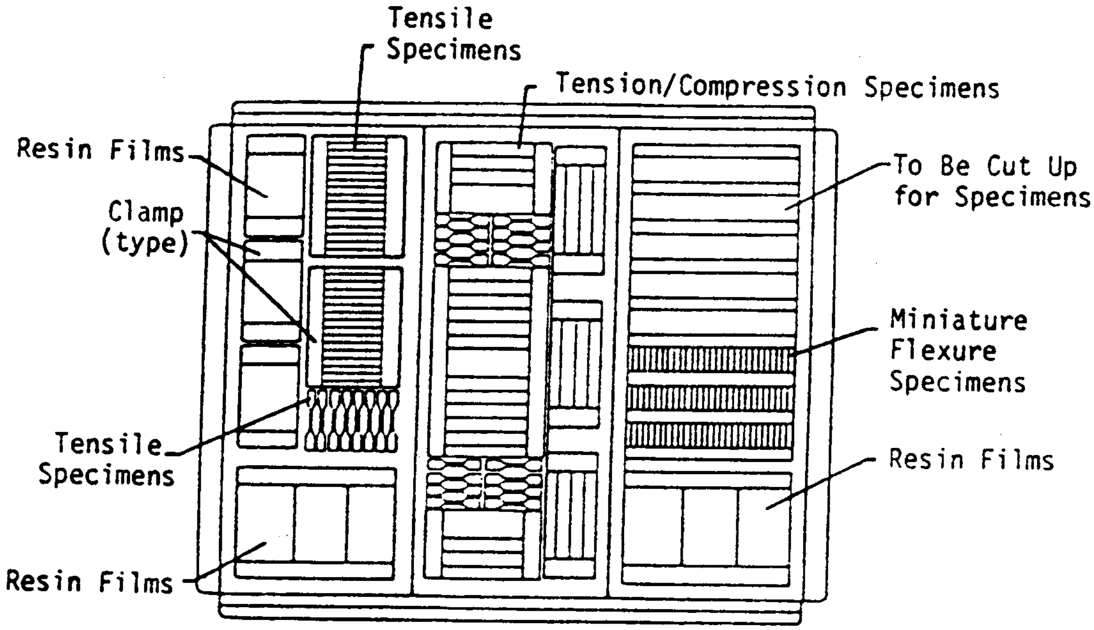


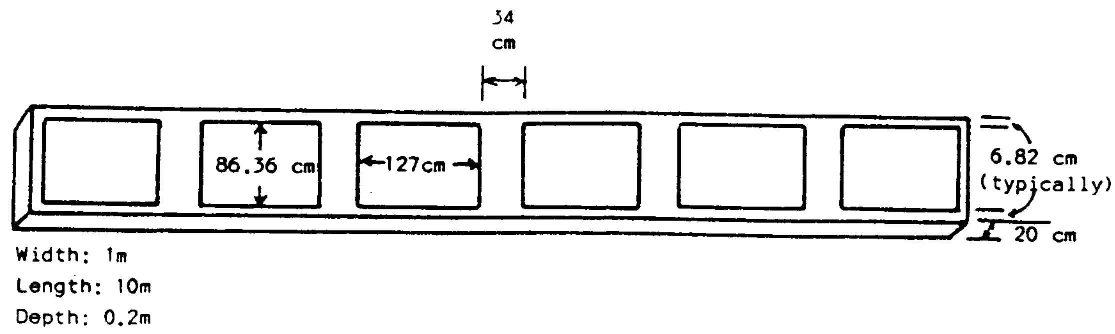
Figure 2-1. Location of Material Panels on Space Station.

ORIGINAL PAGE IS
OF POOR QUALITY

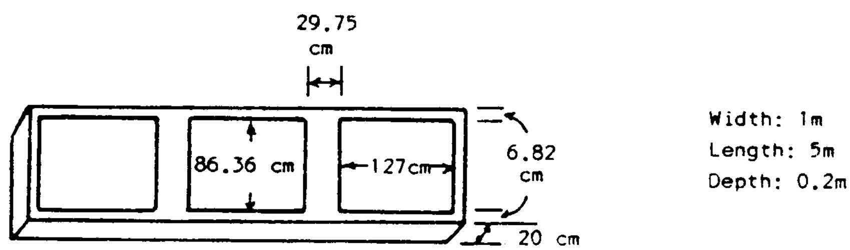


Dimensions of Trays: Width - 86.36 cm
Length - 127 cm
Depth - 15.24 cm

Figure 2-2. Tray Arrangement for Space Exposure of Composite Materials Experiment [1].



(a) Velocity-Facing



(b) Wake- and Sun-Facing

Figure 2-3. Panel Dimensions.

trays, the sun-facing panel three, and the wake panel three. Sample trays would be connected with standard latching power and data connections. Crew members will be responsible for sample placement on the panels. Data collection, power usage, and tray cover closing/opening will be controlled from the ground. Upon retrieval of the tray from the panel, the tray will be placed in a covered standardized transport container which will be secured to the station. Some containers will be designed to vent air slowly into the vacuum so as not to damage the samples while other containers would be designed to return the specimen trays under vacuum for opening inside a vacuum chamber on the ground.

Panels will undergo configuration changes at 90-day intervals in order to continually update the menu of experiments. Monitoring of solar flare activity, atomic oxygen, sample mass loss, incident solar ultraviolet (UV) radiation, temperature cycling, and optical property changes will provide a comprehensive data base against which variations in material properties can be evaluated.

Instrumentation required for data acquisition includes a mass spectrometer to measure atomic oxygen levels, a radiometer to measure incident solar (UV) radiation, a quartz crystal microbalance (QCM) to measure sample mass loss, and thermistors to measure temperature. In addition, instrumentation for real-time feedback of optical property changes could be included.

2.3 ENVIRONMENT

2.3.1 Orbit Parameters

The response of advanced materials and coatings to long-term exposure in a space environment does not depend on orbital considerations, therefore, no orbital parameters are specified. However, a record of orbital parameters during the entire exposure period of a panel would be required to calculate total atomic oxygen fluence and exposure time to the sun.

2.3.2 Platform

The exposure facility for material testing/evaluation in space, as currently planned, will be attached to the Space Station truss structure. However, the experiment could be extended to both the co-orbiting (COP) and polar-orbiting platforms (POP). Remote monitoring of these experiments on the platforms, either indirectly from ground control (via the Space Station) or directly from the Space Station is anticipated. The panel on each platform would contain its own power and telemetry system. Periodic scheduled visits to the platforms would enable sample rotation/retrieval and battery replacement/recharging if no solar panels are available.

2.3.3 Orientation

No special Space Station orientation needs to be specified for this experiment. Remote monitoring of the platforms (i.e., POP, COP), however, may require special considerations to be made with regard to orientation/pointing if the only data link for the platform is via the Space Station; i.e., no direct data link with ground control. The mass spectrometer that is incorporated in some of the active sample trays for atomic oxygen sensing, however, operates most effectively when viewing the velocity vector.

The mass spectrometer ideally should be mounted in such a way that the sensor is pointed into the velocity vector during most data collection periods, so that it has an unobstructed 180 degree view of space, and so that the cover has room to open. For the purposes of this investigation, it is sufficient that the centerline of its aperture be within five degrees of the velocity vector.

2.3.4 Structural Considerations

The experiment requires that sample panels be mounted on the Station in the velocity, wake, and sun-facing directions. (See Figure 2-3 for panel dimensions.)

2.4 SENSORS

The spacecraft materials and coatings experiment requires at least five different types of sensors in the active trays aboard the Space Station. These include:

(1) A Quadrupole Mass Spectrometer,
(2) Thermistors,
(3) Quartz Crystal Microbalance,
(4) Radiometer, and
(5) Optical Reflectometer.

In addition to these five instruments, several optional instruments have been identified as possible candidates/substitutes; these will be addressed later.

2.4.1 Quadrupole Mass Spectrometer

The Air Force Geophysics Laboratory (AFGL) has constructed a Quadrupole Ion-Neutral Mass Spectrometer (QINMS) instrument that could be utilized in this set of experiments aboard the Space Station. QINMS, a compact, versatile quadrupole mass spectrometer, can detect either ion or neutral species; a schematic diagram of the instrument is presented in Figure 2-4. An electron impact ion source, located above the quadrupole assembly, would be activated when the mass spectrometer is in the neutral mode. The grids (located above and below the ion source) would be appropriately biased in each mode for attracting the species of interest, while repelling others. Depending on the mode, the grid current (sample $\lesssim$ 1000 Hz) can be related to either the total positive ion density or the total neutral pressure. The energy of incoming ambient ions would be measured by the energy analysis grid. Moreover, the motion of the Space Station could be utilized by the energy analyzer to separate the high energy ambient species from low energy contaminant species that would travel as a cloud with the Space Station. The preprogrammed quadrupole mass measurements are acquired at a rate of 100 Hz [2].

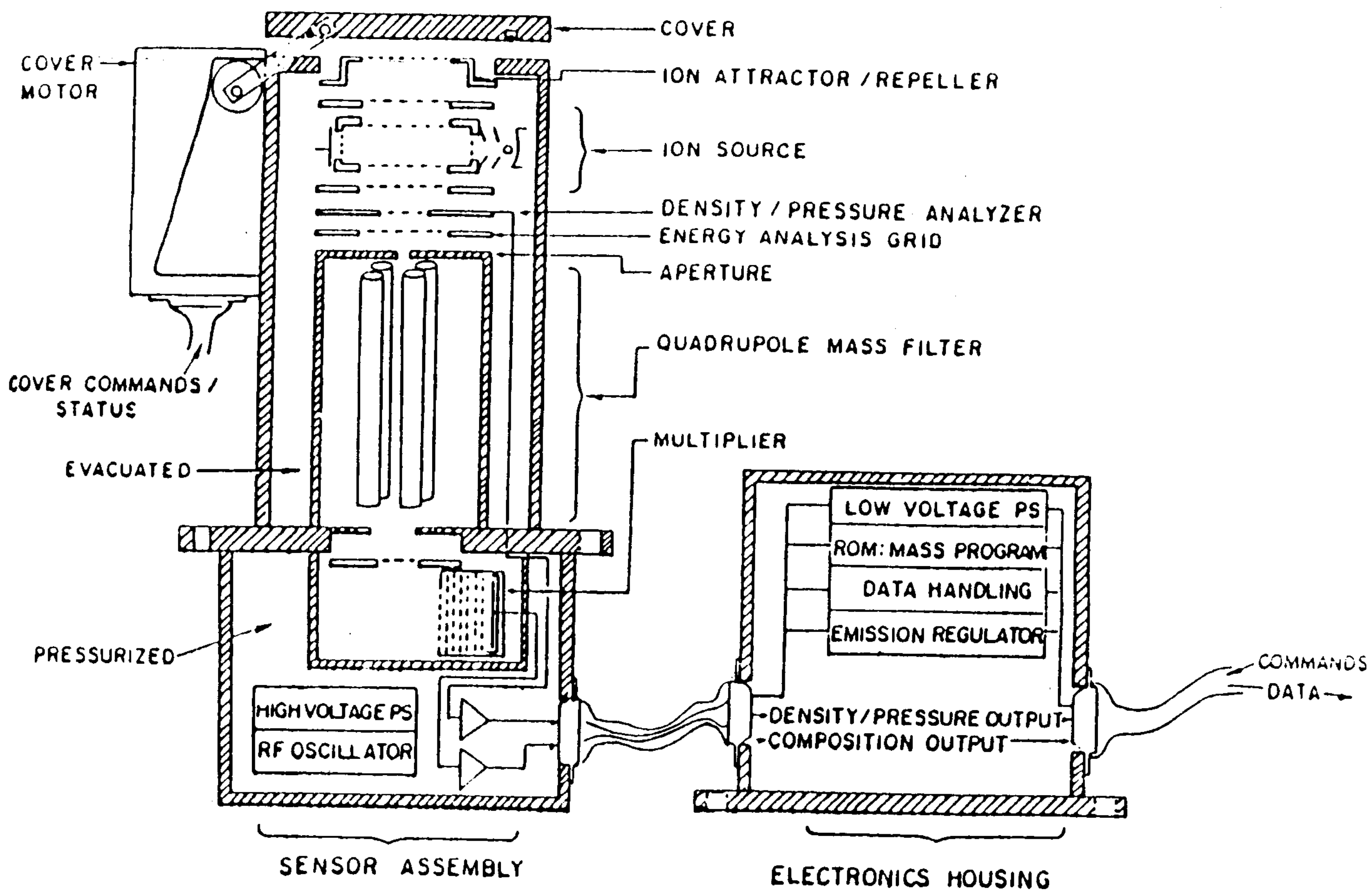


Figure 2-4. Quadrapole Ion-Neutral Mass Spectrometer [2].

The most accurate, complete data would be obtained when QINMS, located in an active tray, is aligned with the velocity vector of the Space Station. Ideally, as mentioned before, the position of the mass spectrometer on the Space Station and the attitude of the Space Station should keep QINMS in ram throughout all mass spectrometer data collection periods [2]. The neutral species to be monitored would include ambient species such as H, He, N, O, N_2, and O_2, and contaminants such as H_2, He, H_2O, CO, NO, and CO_2 [2].

A list of performance characteristics and parameters of QINMS is shown below [2]:

Mass Range: 0 to 150 amu

Sampling Rate: Mass spec output: 100 Hz (max)
Grid current: 1000 Hz (max)
The sampling rate could be matched to the data handling system capabilities.

Mass Program: 512 total steps (preprogrammed).

Grid Current Sensitivity: 10^2 to 10^7 ions/cm^3, ion mode
10^{-8} to 10^{-4}, neutral mode

Size: Electronics: 16.8 cm x 16.4 cm x 15.2 cm
Sensor: 18.7 cm x 14.6 cm x 39.1 cm

Weight: Electronics: 2.7 kg
Sensor: 10.5 kg

Input Voltage: 28 V $\pm$ 4 V unregulated

Average Power: 15 W ion mode
18 W neutral mode

Peak Power: 16 W ion mode
20 W neutral mode

Temperature: 0° to 50°C, operation
-25° to $+125^\circ$C, storage, transportation

Pressure: $< 5 \times 10^{-5}$ torr within QINMS during operation.

EMI Environment: QINMS has passed STS requirements both for production of and sensitivity to EMI.

Gaseous Environment: QINMS emits no gaseous or particulate contaminants other than through normal outgassing of exposed surfaces.

A preliminary assessment of the proposed instrumentation for this experiment concludes that the mass spectrometer is the most complex analytical tool of those identified and requires the greatest amount of data collection and telemetry. Therefore, an assumption is made that the data acquisition/handling and telemetry would be sized to accomodate this instrument with sequential data collection from the other instruments. The following sensor, data acquisition, data handling, and telemetry discussions reflect this assumption. Detailed discussion of these sensors is omitted with appropriate references to LDEF hardware and/or manufacturer's specifications for the instruments.

Mass spectrometers have been included on several groups of STS environment experiments. A quadrupole mass spectrometer was flown on STS-2,3,4 as a part of the Induced Environment Contamination Monitor (IECM) for detection of neutral gases [3]. A Bennet Tube Ion Mass Spectrometer was flown on STS-2,3,4 as part of the Plasma Diagnostics Package (PDP) [4,5,6]. The IECM and PDP instruments monitored both known ionospheric species and contaminants thought to be introduced by the STS [2].

The Ionospheric Disturbances and Modification Branch of the Air Force Geophysics Laboratory (AFGL) has developed a very versatile mass spectrometer that is under consideration for this experiment. QINMS was successfully flown on STS-4 [7]; it is also scheduled to be flown on at least two additional STS missions in the near future [2].

QINMS should have an extensive flight history by the late 1980's. QINMS is scheduled to be on at least one additional payload inside the STS cargo bay, and is also flying on the CRRES satellite for data collection during the chemical release part of the mission. Discrete sampling of different flight environments will provide valuable background data for the Space Station [2]. The primary advantages of the QINMS are: (1) detection of both neutral species and positive ions, and (2) differentiation between ambient ionospheric ions and neutrals and contaminant species introduced by Space Station/STS outgassing and operations [2].

2.4.2 Thermistors

Thermistors are the preferred transducers to sense temperature variations in the sample trays. Three varieties of Yellow Springs Instrument (YSI) thermistor models should cover the temperature ranges expected during this experiment. Model 44033 thermistors would be used for low temperature measurements; model 44032, for high temperature measurements, and model 44031, for wide-range temperature measurements (Figures 2-5 and 2-6). Their operational temperature range would be -55^{o} to $+70^{o}C$; their heat dissipation constant, 8 $mW/^{o}C$, would be transmitted to the high-level analog inputs of the data handling system (described later) [8].

2.4.3 Quartz Crystal Microbalance

A quartz crystal microbalance (QCM), used to measure sample weight loss, has been qualified for use on LDEF and is, therefore, flight ready. The critical element of the instrument is the crystal frequency which determines its sensitivity; higher frequencies correspond to increased sensitivity. Typically, a 1 MHz crystal has been flown in the past. However, for the materials experiment aboard the Space Station, a 15 MHz crystal is required for enhanced sensitivity. Optional instrumentation to replace the standard QCM would be a tapered element oscillating microbalance (TEOM) which would provide greater sensitivity. This instrument is not currently flight ready, but could be by 1992. More detailed information about the TEOM can be obtained from R & P Associates in Albany, New York. Additionaly, the use of a Vibron (operating in the 15-30 Hz range to look at changes in modulus) was considered as possible instrumentation to be considered in later Space Station material experiments.

Flight ready QCM's are manufactured by both Faraday Labs and Berkeley Controls. For more detailed information about this instrument, refer to LDEF experiment S1002, "Investigation of Critical Surface Degradation Effects on Coatings and Solar Cells Developed in Germany" developed by Ludwig Preuss [9].

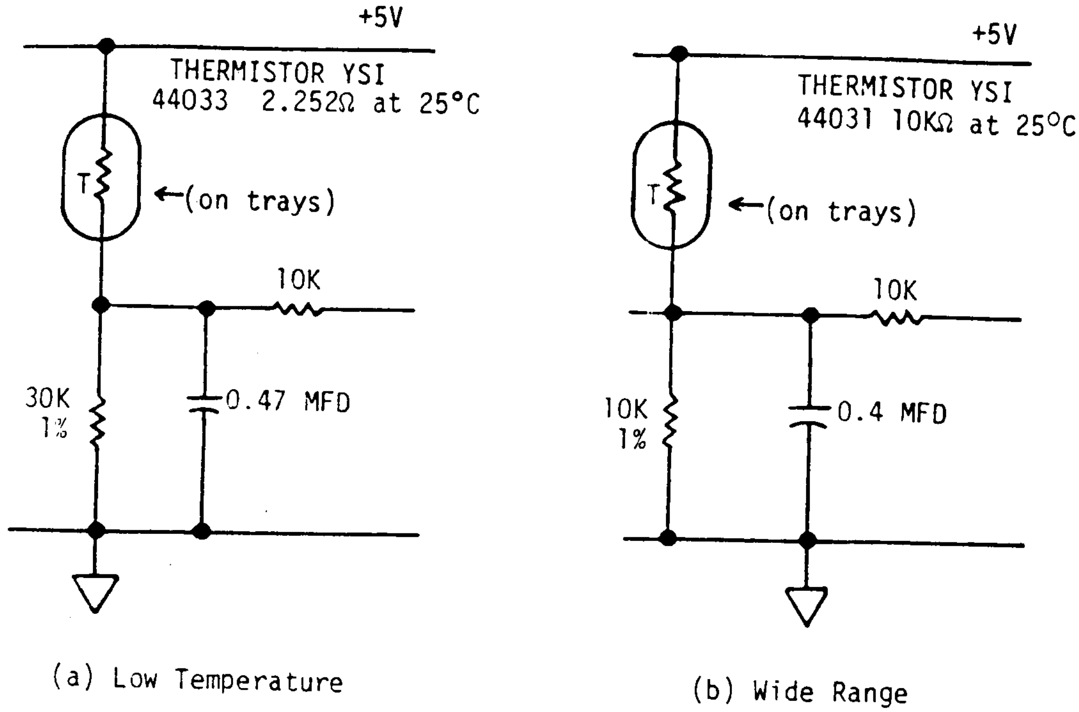


Figure 2-5. Low Temperature Thermistor Circuit and Wide Range Thermistor Circuit [8].

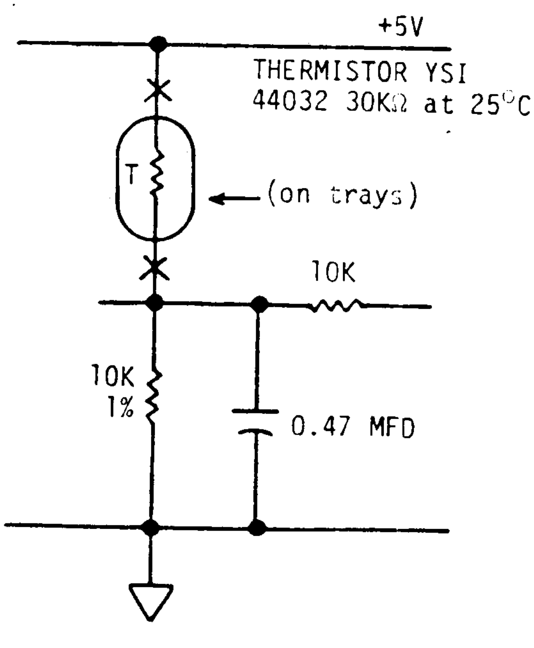


Figure 2-6. High Temperature Thermistor Circuit [8].

2.4.4 Radiometer

Radiometers have already been flight qualified for use on LDEF for measurement of incident solar radiation. For example, a radiometer was utilized along with the optical reflectometer for the thermal control surfaces experiment developed at Marshall Space Flight Center (MSFC) noted below.

2.4.5 Optical Reflectometer

An optical reflectometer used to measure material optical properties has also been qualified for use on LDEF. The "Thermal Control Surfaces Experiment" (S0069), developed by Donald Wilkes and Harry King at MSFC (See Figure 2-7) could be utilized aboard the Space Station for sample optical measurements. Refer to LDEF experiment S0069 for more detailed information about this instrument [9].

2.5 DATA ACQUISITION

The in-flight data collection system for this experiment is comprised of various data sensors (thermistors, etc.), a signal conditioning subsystem, and a data handling and storage system. Three such data collection systems would be used, one for each of the panel locations on the Space Station.

Raw data is acquired from the various sensors on active trays through a Signal Conditioning Unit (SCU), similar in concept to the SCU flight qualified for LDEF experiment M0003, "Space Environment Effects on Spacecraft Materials" [9]. The SCU interfaces between the transducers and the data handling system recording networks. Sensor signals would be processed in the SCU prior to transmission to the data handling and storage system. On command from the data handling system, the SCU would first initiate and then disengage (after completion) power to the experiments for the data scans. A typical SCU would house the monitoring circuits for the

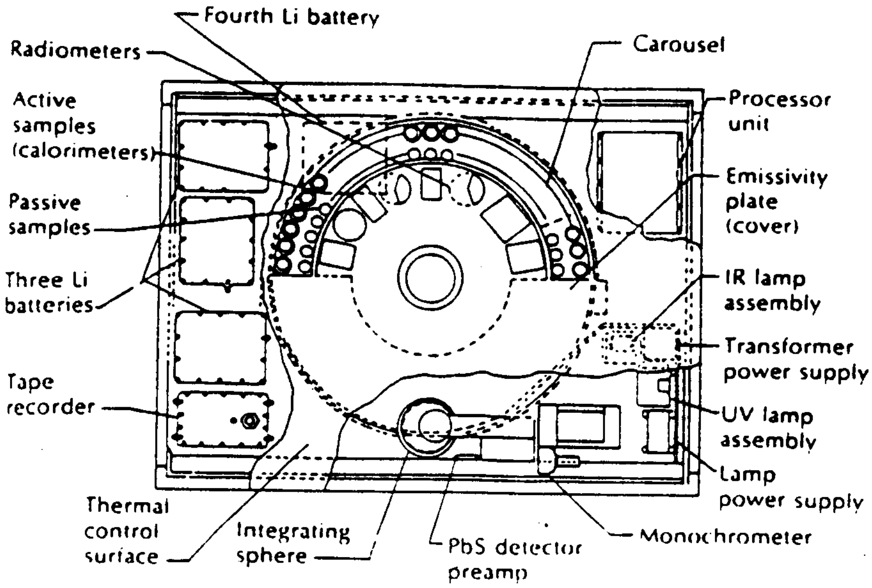


Figure 2-7. LDEF Thermal Control Surface Experiment (S0069) Which Included An Optical Property Measurement System [9].

thermistors, any electronics for the mass spectrometer or the quartz crystal microbalance, and the interface voltage translation circuits for the parallel digital data. A functional block diagram of the SCU is provided in Figure 2-8 [8].

As previously mentioned, of all the sensors in this Space Station experiment, the quadrupole ion-neutral mass spectrometer (QINMS) requires the greatest amount of data collection. QINMS also requires a minimum of eight commands in its present design configuration which would be supplied via the SCU by linking to the Space Station data handling system. These commands are as follows:

a. COVER POWER ON
b. COVER POWER OFF
c. COVER SELECT OPEN
d. COVER SELECT CLOSED
e. QINMS POWER ON
f. QINMS POWER OFF
g. ION MODE
h. NEUTRAL MODE

Various restrictions to the sequence of commands would apply; these would be detailed during subsequent experiment development. In addition to the eight command requirements, QINMS would also require a 0-5 V square wave telemetry clock to drive the mass program, which would be synchronized with the Space Station telemetry system, having a maximum frequency of 100 Hz.

Data from the other instruments would typically be in analog form, except for the QCM, which transmits data to the data handling system in parallel digital form. The other instrumentation, therefore, requires analog-to-digital (A/D) conversion.

2.6 DATA HANDLING

The data handling and storage system for this experiment must have the capacity to process and record information from many separate digital and analog data channels. The system will be user programmable to provide the

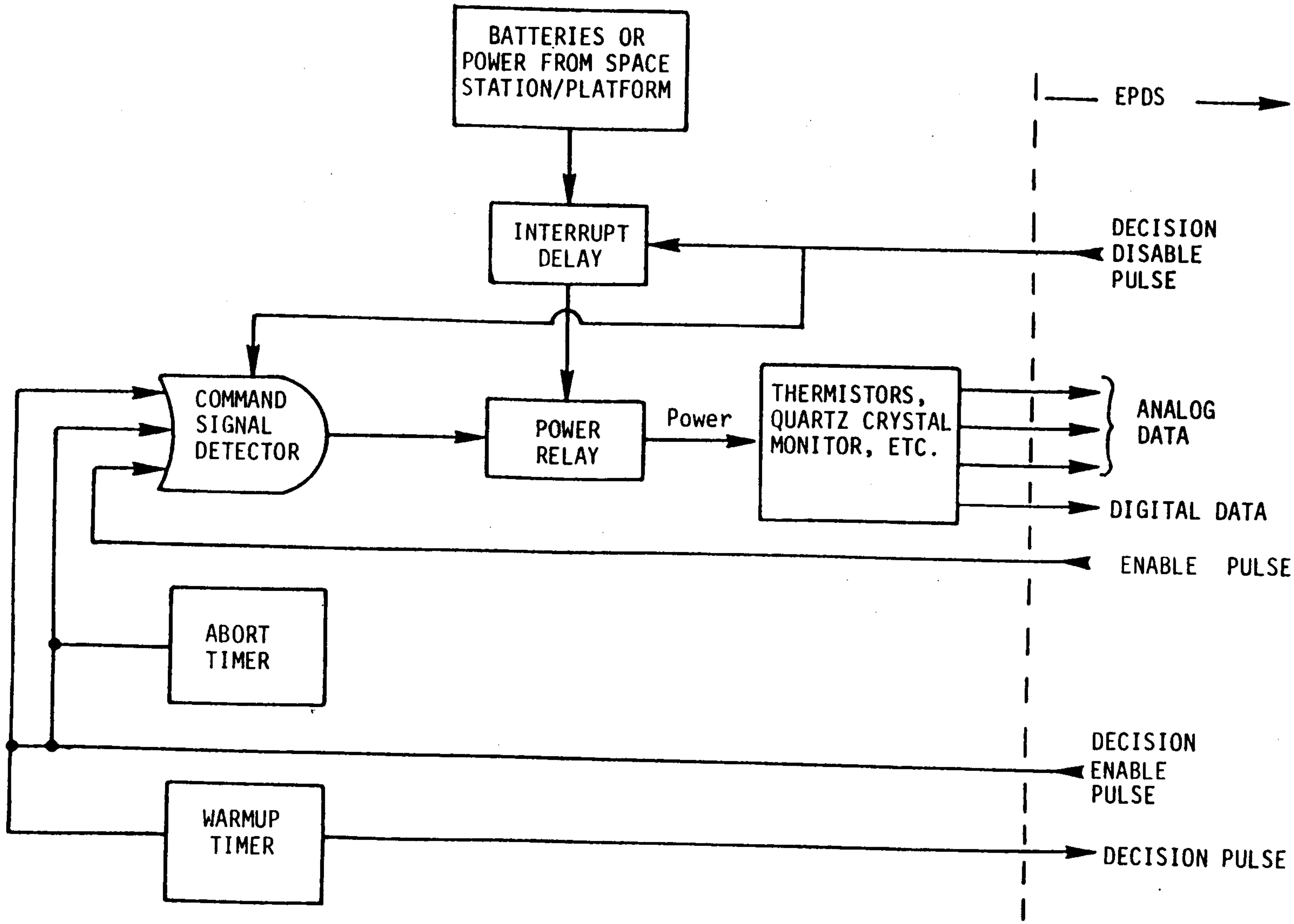


Figure 2-8. SCU Functional Block Diagram [8].

desired timing for the data collection and the binary word length used in the A/D conversion of the analog data. Use of a data handling and storage system similar to that qualified for LDEF would adequately meet inflight processing and recording requirements. LDEF utilized an Experiment Power and Data System (EPDS) for data collection and handling for which a functional block diagram is shown in Figure 2-9.

Adapting the experience gained from LDEF for handling in-flight data, both a Synch and time code would be generated by the data system immediately prior to data collection/recording. The Synch code identifies the onset of a data scan, the specific experiment, and the location of the data system [8].

Data will be collected using 5 consecutive data scans (to ensure data accuracy) which will be stored in a buffer memory of the data system. Data in the buffer memory would then be transferred to a single-track tape recorder (e.g., Lockheed Mark V Type 4200) [10] after the five data scans for eventual dumping to ground each orbital period [8].

The relatively self-sufficient nature of the mass spectrometer dictates that the only communications between flight crew or ground personnel and the sensor are: (1) transmission of the eight previously described commands, (2) reception of data from its health monitors, and (3) scientific data reception. By the time the Space Station is operational, the proposed mass spectrometer (QINMS) will have flown on missions where an STS mission specialist has demonstrated control of the sensor from the flight deck and where ground-to-air telemetry has been established, thus demonstrating flight performance.

2.7 TELEMETRY

Minimum telemetry requirements, based on the mass spectrometer (QINMS), are eight 0-5 V analog telemetry monitors with eight bit resolution. These provide the health status information and the scientific data for the instrument. Additionally, two discrete telemetry monitors are required for the QINMS cover position. One set of analog monitors would

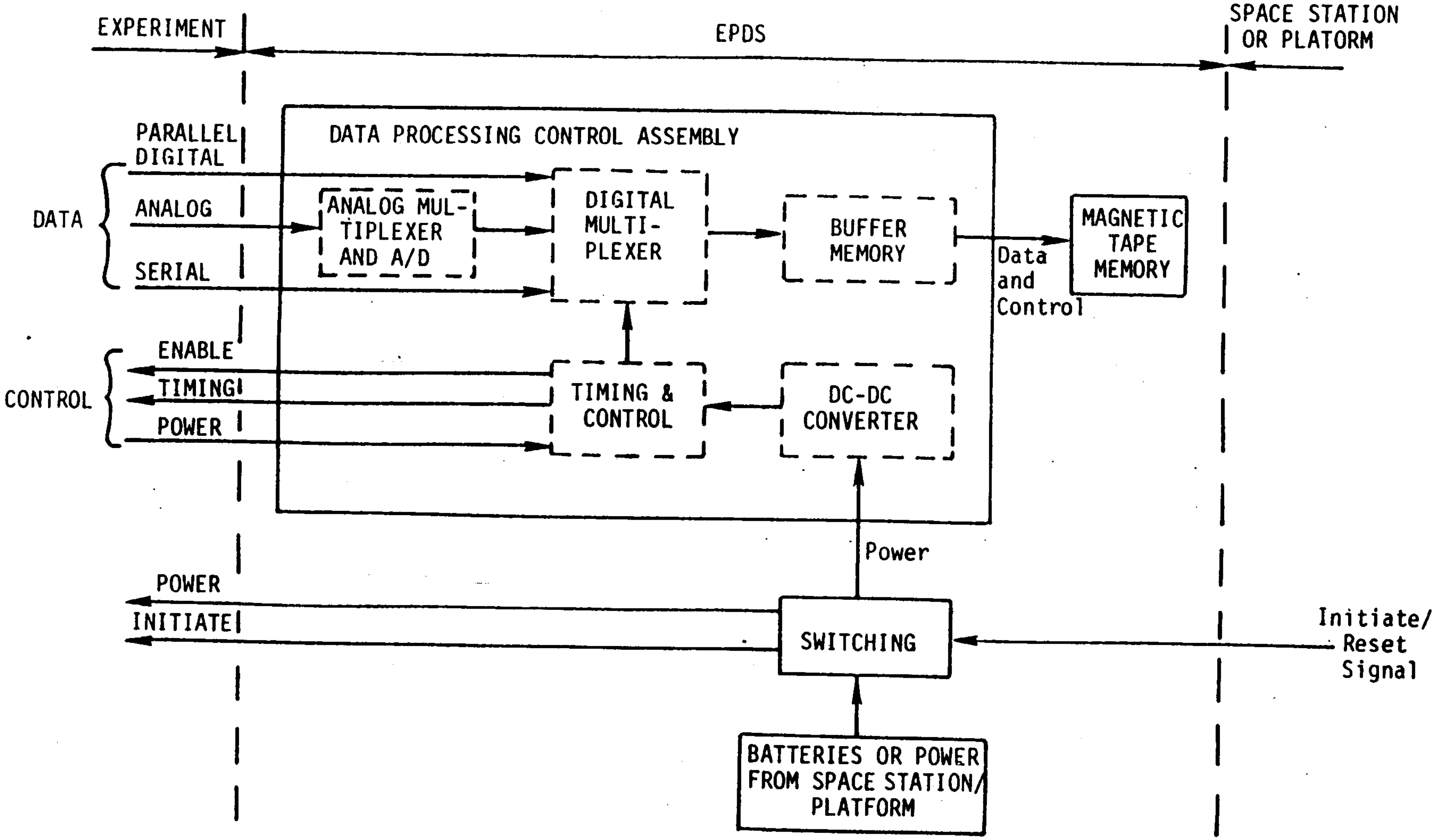


Figure 2-9. EPDS Block Diagram [11].

have a sampling rate of 1 Hz. These would be used for the high voltage monitor, the electron beam monitor, and the commutated monitors. Another set of analog monitors would have a sampling rate of 100 Hz (or other, slower rate). These would monitor the DC sweep voltage, the RF sweep voltage, the retarding voltage, and the mass spectrometer current. Other analog monitors would have a sampling rate of 1000 Hz (or other, slower rate) for monitoring the grid current amplifier. The total analog bit rate would be 11,224 bps (maximum). The sampling rate for the five faster monitors could be adjusted to match the Space Station telemetry system capabilities [2].

The two discrete telemetry monitors would have a sampling rate of 1 Hz. These would provide information on the cover closed monitor and the cover open monitor. Therefore, the total discrete bit rate would be 2 bps. Thus, the total data rate from the telemetry monitors would be approximately 11.3 kbps (maximum). The data rate could be reduced by decreasing the sampling rate if required by the Space Station telemetry system [2].

2.7.1 Additional Data Requirements

The start and stop times (with 0.1 second resolution) of each Space Station operation that could affect the exterior environment would be required for accurate assessment of environmental changes influencing material performance. Such events would include Reaction Control System (RCS) main and vernier thruster firings, Orbital Maneuvering System (OMS) firings, water dumps, fuel cell purges, flash evaporator operations, etc. [2].

2.8 CREW ACTIVITIES

Flight crew training requirements would depend strongly on the method chosen for commanding the instrumentation on the Space Station, as well as the method for reviewing the health and scientific data for each experiment during the mission. Extensive flight crew training in the operation of the mass spectrometer, for instance, would not be necessary if communication occurs directly between ground personnel and the Space Station. However, if a Mission Specialist was to be responsible for the operation and assessment of the experiment status, then an extensive training program would be required [2]. This training program would require discussions on the principals of mass spectrometer and quartz crystal microbalance operations and associated command requirements, the procedures for in-space calibration of instruments, knowledge of telemetry interpretation from the instruments, health status criteria of the various instruments, as well as trouble-shooting and maintenance procedures for the instruments. The magnitude of such an extensive training program, in addition to the concurrent requirement for development of hardware on the Space Station to enable reformatting and analysis of the data, precludes direct communication between the experiment and Space Station personnel in the initial operation of this experiment [2]. As this experiment and the Space Station mature, crew interaction may be possible if initial experiment operations show this interaction to be necessary.

2.8.1 Operational Constraints

The QINMS sensor cover should not be opened until a 30 minute time interval has elapsed after installation of an active tray into a panel and ideally, the QINMS power should not be turned on until an additional 30 minutes has elapsed after opening of the QINMS cover. These restrictions allow the mass spectrometer sufficient time to pump out to operational pressure before power application [2]. The cover of the mass spectrometer should be opened at least one hour prior to data collection, allowing the quadrupole electronics time to stabilize. QINMS, in its present configuration, has a maximum switching rate of once per minute between ion and neutral modes through the data collection period. Five periods each of ion and neutral data is the minimum acceptable amount of data [2]. Also, the front end of the sensor should have an unobstructed 2π steradian view of space in the ram direction. The QINMS pressure requirement limits use of the mass spectrometer (as currently designed) to altitudes higher than 200 km.

There are no special requirements or restrictions on Space Station (or STS) operational timings such as thruster firings or water dumps, since measuring the effects of such perturbations to the Space Station environment on materials properties is part of the mission of this experiment [2].

2.8.2 Mission Timeline

Activities to be conducted are given in the mission timeline shown in Table 2.1. Further definition of these activites is required.

The following assumptions have been made for integrating TDMX 2011 into the Space Station and, therefore, impact the mission timeline.

A. General

1. One crewmember will be located in the STS cargo bay, the Mobile Remote Manipulator System (MRMS) would have been activated and checked out, and one Manned Maneuvering Unit (MMU) will be available for use.
2. The basic Space Station structure will have been erected.
3. The panels will be installed after the Space Station is inhabited and in an operational mode.

B. Flight One - Sample Attachment Structure

1. (Install panels)
   a. The three sample attachment panels will be easily stowed/accessed in the STS cargo bay.
   b. Both a launch restraint and separate panel temporary restraint will be required.

Table 2.1. TDMX 2011 - SPACECRAFT MATERIALS AND COATINGS CREW TIMELINE EVALUATION

FUNCTION/TASK	TIME MANMINUTES	METHOD	COMMENTS
FIRST FLIGHT			
• STS docks with structure			See Assumption A.
• Deploy sample attachment panels;	90		See Assumptions B.1.a and B.1.b.
• To velocity-facing truss structure	(30)	MRMS	
• To wake-facing truss structure	(30)	MRMS	
• To solar array truss structure	(30)	MRMS	
• Install Panels	60		
• Maneuver to position panel at insertion location	(10)	EVA	See Assumption B.1.e.
• Insert panel into brackets on truss	(5)	EVA	See Assumptions B.1.c and B.1.d.
• Check/verification of automatic locking mechanism	(5)	EVA	
• Same three steps above for other two panels	(40)		
• Install Sample Trays	480		See Assumption B.2.
• Removal of shipping containers from the STS cargo bay	(25)	MRMS	
• Store shipping containers on Space Station	(40)	MRMS/IVA	
• Remove sample trays from shipping containers and inspect for damage	(60)	IVA	
• Translation of sample trays to appropriate panel locations	(20)	MRMS	
• Position trays to insert into desired panel location	(5)	MRMS	
• Connection of active trays to panel electrical/instrumentation lines	(5)	EVA/MRMS	See Assumptions B.3.a and B.3.b.
• Check/verification of operational status of active trays	(5)	EVA	See Assumptions B.3.c
• Same seven steps above for other two panels	(320)		

Table 2.1. Continued.

FUNCTION/TASK	TIME MANMINUTES	METHOD	COMMENTS
• Stow MRMS	10		
SECOND FLIGHT			
• Retrieval of sample trays prior to STS arrival	90		
• Disconnection and removal of active sample trays from panel electrical/instrumentation lines	(30)	MRMS	See Assumption B.3.b.
• Removal of passive sample trays from panel	(30)	MRMS	See Assumption B.2.d.
• Translation of sample trays to storage containers on Space Station	(30)	MRMS	
• STS docks with Space Station			
• Rotation of sample trays	535		
• Removal of shipping containers from the STS cargo bay	(25)	MRMS	See Assumption B.2.
• Store shipping containers on Space Station	(40)	MRMS/IVA	
• Remove new sample trays from shipping containers and inspect for damage	(60)	IVA	
• Translation of sample trays to appropriate panel locations	(20)	MRMS	
• Position trays to insert into desired panel location	(5)	MRMS	

Table 2.1. Concluded.

FUNCTION/TASK	TIME MANMINUTES	METHOD	COMMENTS
• Connection of active trays to panel electrical/instrumentation lines	(5)	EVA/MRMS	See Assumptions B.3.a and B.3.b.
• Check/verification of operational status of active trays	(5)	EVA	See Assumptions B.3.c.
• Same seven steps above for other two panels	(320)		
• Translation of exposed sample trays (in shipping containers) to STS	(30)	MRMS	
• Stow retrieval containers in STS cargo bay	(25)	MRMS	
• Stow MRMS			
<u>THIRD FLIGHT</u>			
• Duplicates second flight			

c. Some type of crew-operated quick-release handling aid for the panels.
d. Panel bracketry already installed on truss structures.
e. One crewmember operating MRMS for panel positioning and another crewmember performing the attachment.

2. (Install sample trays)
   a. Sample trays are not within the panels during initial setup.
   b. Two types of sample trays, active and passive.
   c. Sample trays are installed sequentially.
   d. One crewmember (EVA) operation, with crewmember in foot restraints.
3. (Power/Data-Link Connection)
   a. Power for active trays provided by Space Station.
   b. Crew would manually (EVA or MRMS) connect (or disconnect) power and data interconnections between sample trays and panels.
   c. Crew participation is limited to observation for nominal operations, possibly some EVA assistance in contingency operations.

C. Flight Two - Rotate/Retrieve Sample Trays

Same assumptions as B.2 and B.3.

2.9 GROUND STATION

Information is retrieved from the data handling (e.g., modified EPDS) recorder by transcribing to a ground-based computer compatible tape recorder (e.g., Kennedy Model 9832) [12]. Utilizing the experience gained from the LDEF mission for in-flight data collection, sensor information from the active trays on the Space Station would be collected as 10-bit data words. Telemetry data from the experiments would be stored as 8-bit characters bearing no fixed phase relationship to the original 10-bit data words. The tape record would subsequently be searched for the Synch code; the time and data words would then be reconstructed, converting them to a more useful form. These flight data tapes would be processed on a ground-based mainframe computer to analyze and manipulate the data, presenting results in appropriate formats [8]. A ground-based software program would allow: (1) sequential searching for the Synch code, (2) locating and converting the time code, and (3) reconstructing and converting the 10-bit data words to appropriate data values for each channel [8].

2.10 SUPPORT EQUIPMENT

2.10.1 Data Handling Support Equipment

The data handling system ground support equipment would consist of a magnetic tape memory controller, a data display monitor, and a computer-compatible tape recorder (with tapes and interfacing cables). During pre-flight checkout of the experiment, portable hardware will be required to assess system performance (a) during assembly, (b) before and after flight acceptance tests, and (c) during prelaunch testing [8].

The portable ground support equipment would consist of a power supply, a pulse command sequencer, dummy loads, microcomputers (e.g., TRS 80), and a channel selector probe. The command sequencer would simulate the data handling system pulses and timing sequences to the SCU. The dummy loads would simulate a transducer signal to the SCU and the channel selector probe would sample each of the SCU outputs. This portable equipment permits the SCU's to be checked out independently of the data handling system [8].

2.10.2 Ion Pump

Additional support equipment would consist of a small ion pump which would be attached directly to the front cover of the mass spectrometer (QINMS) during preflight testing. The ion pump would be removed prior to the active tray being place in the reuseable carrier in the STS cargo bay [2]. However, if the QINMS tray could be transported sealed in vacuum, then the ion pump would be redundant.

REFERENCES

1. Clark, L. G., ed., Preliminary Copy: LDEF First Mission Experiments, September, 1981.

2. Hunton, D. E., Shuttle Environment and Ionospheric Irregularity Studies Using the Tethered Satellite System, NASA Proposal submitted by AFGL (Hanscom AFB, MASS), August, 1984.

3. Carignan, G. R. and E. R. Miller, Mass Spectrometer, in "STS-2,3,4 Induced Environment Contamination Monitor (IECM), Summary Report," NASA TM-82524, E. R. Miller, Ed., February, 1983.

4. Grebowsky, J. M., M. W. Pharo, III, H. A. Taylor, and I. J. Eberstein, Measure Thermal Ion Environment of STS-3, AIAA-83-2597, AIAA Shuttle Environment and Operations Meeting, October 31-November 2, 1983, Washington, DC.

5. Shawhan, S. D. and B. G. Murphy, Plasmas Diagnostics Package Assessment of the STS-3 Orbiter Environment and Systems for Science, AIAA-83-1253, AIAA 21st Aerospace Sciences Meeting, January 10-13, 1983, Reno, Nevada.

6. Murphy, G. R., S. D. Shawhan, and J. S. Pickett, Perturbations to the Plasma Environment Induced by the Orbiter's Maneuvering Thrusters, AIAA-83-2599, AIAA Shuttle Environment and Operations Meeting, October 31-November 2, 1983, Washington, DC.

7. Narcisi, R. S., E. Trzcinski, G. Federico, L. Wlodyka, and D. Delorey, The Gaseous and Plasma Environment Around Space Shuttle, AIAA-83-2659, AIAA Shuttle Environment and Operations Meeting, October 31-November 2, 1983, Washington, DC.

8. Giguere, R. P., SD/LDEF Technical User's Manual, The Aerospace Corp. (El Segundo, CA), September, 1981, AD-A113504.

9. Clark, L. G., et al., eds., The Long Duration Exposure Facility (LDEF): Mission 1 Experiments, NASA-SP-473, 1984.

10. Operating Manual for the Mark V Type 4200 Magnetic Tape Recorder and Reproduce Unit, Manual No. 3A0177, Lockheed Electronics, Industrial Technology Division (1979).

11. Motley, W., J. Cox, and J. Bene, Experimental Power Data System for Long Duration Exposure Facility, NASA Langley Research Center, Hampton, Virginia (1 August 1978, rev 18 April 1979).

12. Operating and Maintenance Manual for Model 9832 Buffet Tape Transport and Model 9800 Digital Tape Transport, Kennedy Co., Altadena, California.

3.0 TDMX 2072 - SPACECRAFT STRAIN AND ACOUSTICS SENSORS

CONTACT: Dr. Joseph S. Heyman, MS 231
NASA LaRC (804) 865-3036

3.1 GOALS AND OBJECTIVES

The goal of this experiment is to develop and demonstrate a structural monitoring "nervous" system for the Space Station to provide crew, mission control, and ground-based contractors with real-time structural status including analysis of unexpected events such as a "hard docking bump" or a debris/meteor impact. This experiment will develop technology necessary to monitor spacecraft structures and provide long-term structural verification through advanced nondestructive evaluation (NDE). A further goal would be to test these prototype systems on spacecraft missions prior to the Space Station to validate and enhance performance so as to ensure acceptable performance of Space Station monitoring needs.

A variety of advanced sensor instrumentation will be tested, evaluated, and verified on both the ground and during early spacecraft missions. Early demonstrations of the technology will include optimization experiments over a specified range of measurements of critical sensor parameters. Correlation and calibration of sensors will be required utilizing data obtained from ground laboratory tests as well as flight tests. Early monitoring system sensor data is anticipated to be batch processed, with concomitant design characteristics for conversion to interactive real-time data processing at some future date. Additional experiments for structural monitoring of the other orbital platforms and provision for data communications are anticipated.

3.2 DESCRIPTION

The spacecraft strain and acoustic sensor experiment could be made an integral part of the Space Station structure as it is assembled, resulting in increased flexibility of experimental design. Some sensors (i.e., acoustic emission) can be fixed/imbedded within the actual structure during fabrication or placed in position subsequent to fabrication while other NDE sensors would require accurate placement after station fabrication. As previously mentioned, the specific type of NDE sensors to be utilized for structural monitoring of the Space Station will be determined primarily by the Station's construction material and geometric configuration.

Advanced acoustic emission sensors with broadband capability and fiber-optic sensors will be monitored during the mission by a preprogrammed computer, thereby minimizing crew involvement. The sensors will be developed and tested utlizing ground facilities, taking advantage of LaRC's current R&D program output to provide enhanced state-of-the-art sensors. Other state-of-the-art monitoring devices designed to assess strain which have been developed at LaRC, such as acoustic and fiber-optic interferometric sensors, will be structurally integrated into the Space Station as well.

Some redundancy in monitoring systems is anticipated. For example, an overlap of system monitoring capabilities could occur as new technological instrumentation is tested and optimized to meet increased future Space Station needs. One such system currently being developed in the Materials Characterization Instrumentation Section at LaRC, the Optical Pulsed Phased Locked Loop (OP2L2) strain sensor, would provide full geometric and dynamic characterization of the Space Station. This high resolution instrumentation has produced very encouraging results in preliminary ground testing, thus providing an extremely versatile monitoring device for the accurate assessment of Space Station geometry, strain, and dynamic response for station control or damage assessment. This instrument, combined with analytical methods such as finite element analysis, could provide accurate details of stress distributions within the Space Station structure which can be used as feedback for structural control.

In the case of an acoustic emission sensor, an "event" will be simulated in a designated area of the spacecraft. This sensor network monitoring the anamolous signal would first locate the disturbance. Note that at the present stage of sensor development, the envisioned monitoring system could only provide data on the location of this disturbance and a realtive indication of its magnitude, not its cause or form of damage. Continuing resarch in acoustic emission and details of the Space Station geometry/materials will greatly enhance the capability of this system for damage assessment. For acoustic emission events of sufficient concern, a portable NDE instrument (i.e., ultrasonic crack detection, etc.) would then be used to manually scan the region designated as the source of the disturbance for quantitative evaluation of its cause and form. The manual scanning would be accomplished by a designated crew member, thus requiring either EVA, MRMS, or IVA.

The preprogrammed computer would automatically record and transmit data on the disturbance to a temporary data storage dump for subsequent transmission to ground. On-board data processing is required, but, in its early stages of technological development, does not have to be an interactive system. It is envisioned that incorporation of the ground control data processing feedback into the on-board processsing would eventually evolve into an interactive real time monitoring system. It is still uncertain as to the absolute necessity of having an interactive system during the inception of the Space Station, but its benefits can readily be seen and is assumed to be required during later stages of development.

Correlation of the flight data with laboratory ground data is required for further evaluation. It is possible that certain acoustic emission spectrums/signatures could be qualitatively correlated with specific types of "event" based on ground test data. However, this type of failure recognition requires more investigation. The test data obtained will be displayed in real time and, as previously mentioned, stored for later processing and/or transmission to the ground via the Space Station data link. For tests where the experiments are being simulated under control of Space Station personnel, the computer will be configured for real-time data interpretation. Processed data will also be stored and transmitted to the ground for later analysis. Ground analysis (feedback) of the data could elicit action on the part of the Station crew for certain NDE tests to be performed for further evaluation of the disturbance.

This sensor has a limited frequency range, covering only one decade. Moreover, the frequency response curve typical of most commerically available transducers does not provide a "flat" voltage response over the full frequency range, essential to accurate AE data interpretation. An atypical state-of-the-art transducer manufactured by Industrial Quality, Inc. approaches this desired transducer response over its entire frequency range while maintaining high sensitivity. The IQI Model 501 transducer has the following specifications [2]:

Frequency range: 50 kHz to 1 MHz

Amplitude Response: Flat within $\pm$ 3 dB (over entire frequency range)

Dimensions (cylindrical): 4.4 cm dia. x 3.2 cm high

Weight: 283 g

Although the specifications of these sensors are excellent by today's standards, they still fall short of the additional requirements necessary for adequately monitoring the Space Station. An aggressive AE resarch program to develop transducers with extended range and sensitivity is a fundamental necessity for this sensor technology to favorably impact the Space Station.

Approximately 20 AE sensors are expected to be placed in critical high-stress areas (i.e., cross-truss junctions) during the early stages of the Space Station. As monitoring needs increase for growth configuration of the Space Station, additional sensors would be added as required to ensure adequate monitoring (See Figure 3-1).

The second type of sensor to be evaluated for this application is a pressure sensor which employs optical fibers (waveguides), typically in a grid array. This array would yield both the functional form and the amplitude of the stress distribution in structural support members [3,4]. This type of sensor offers good dynamic range, high sensitivity, excellent mechanical flexibility, and is not affected by electromagnetic interference. Use of this sensor is especially important if the choice of Space Station construction material is a composite. In this case, the sensor can be embedded in the structure during manufacture, thus providing the capability of determining internal intraply effects as well as distributed body forces [5,6]. The optical fiber acts as the transducer element in a quantitative differential fiber interferometric detection technique (See Figures 3-2 and 3-3).

 33 - 34

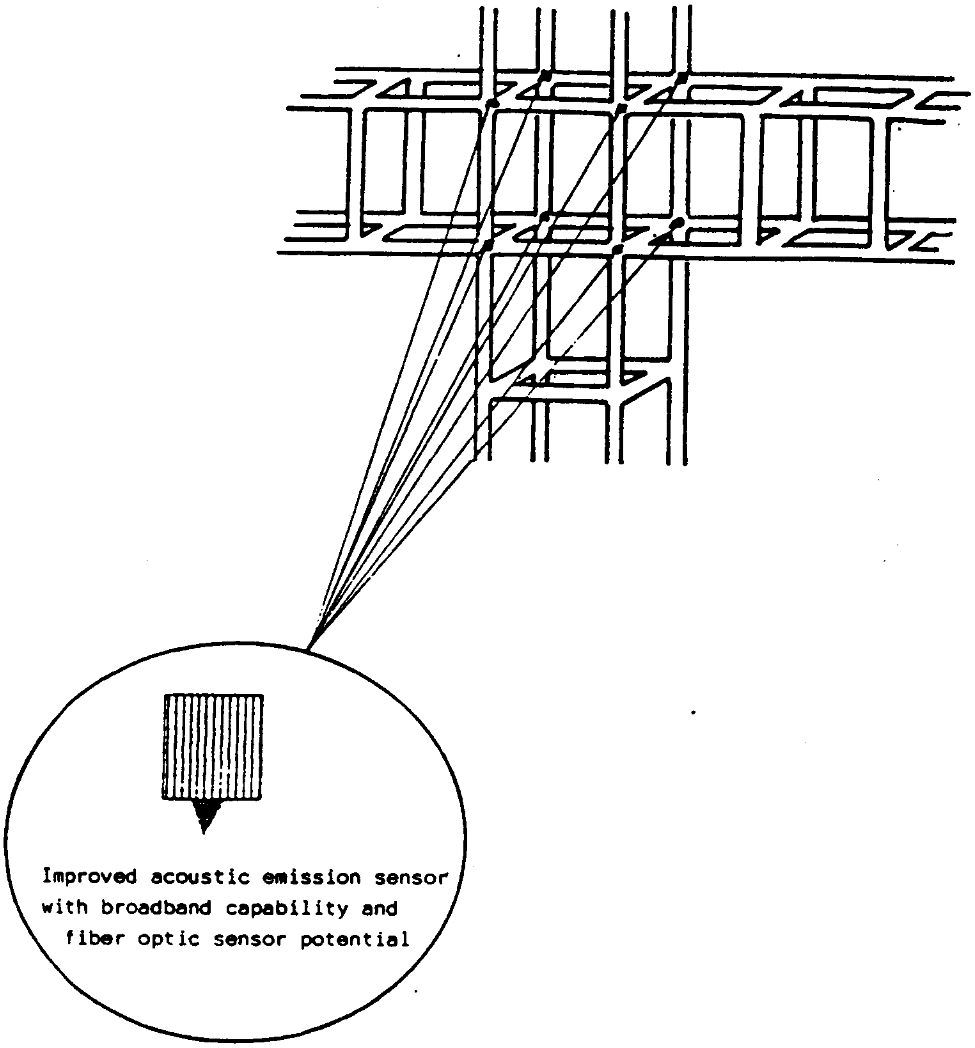


Figure 3-1. Illustrative Acoustic Emission (AE) Sensor Placement for Space Station.

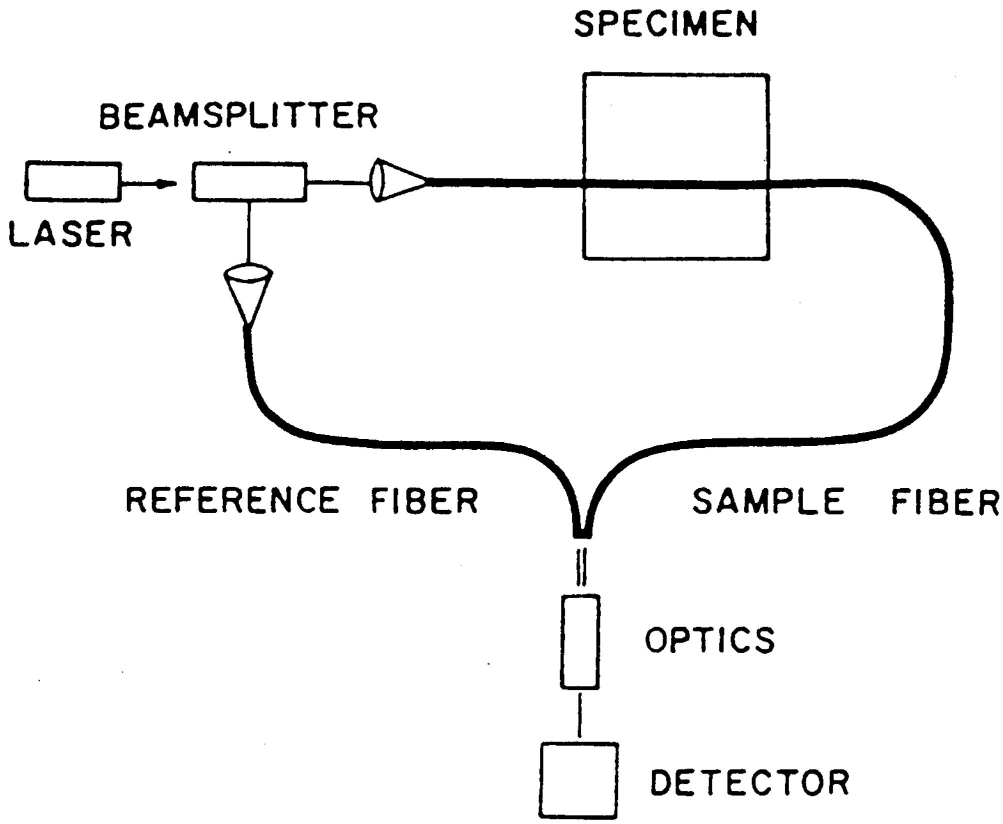


Figure 3-2. Single Mode Fiber Interoferometer System for Measurement of Loading on Specimen [4].

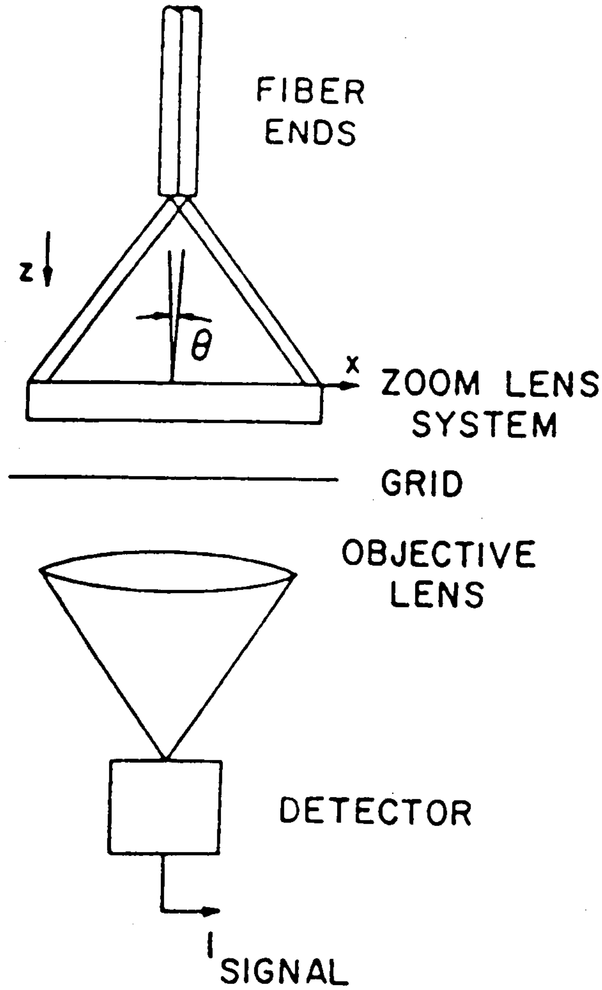


Figure 3-3. Geometry of Detection Optics [4].

The acoustic pressure field to be measured distorts the optical fiber, changing both its geometry and index of refraction, resulting in modulation of the intensity, phase, polarization, and mode of the propagating optical field. These induced changes may then be observed to determine the nature of the applied pressure field [3].

The optical fiber sensor may still be utilized, even if the construction material of the Space Station is a metal. In this case, the sensor would not be embedded in the material but fixed between or along structural support members. Several hundred meters of optical fiber would be required to sufficiently monitor critical stress areas (e.g., those areas experiencing high bending moments during orbital maneuvers).

The third type of monitoring device, the OP2L2 strain sensor, (still under development) as mentioned previously, utilizes a highly sensitive optical detector calibrated to a specific coherent wavelength of light which is provided by a modulated laser beam. The basic objective of this instrument is the extremely accurate measurement of distance (resolution of parts in 10 million) from the detector to the target (Space Station). The sensor could conceivably be mounted on a boom extending away from the Station. The detector would then view the Station, accurately measuring Space Station geometry, strain, and dynamic response for control and damage assessment (See Figures 3-4 and 3-5).

The surface area to be investigated utilizing the OP2L2 sensor can vary widely, depending on aperture size and distance from the target. Use of a coherent light source maintains a nearly parallel beam with very little divergence, even if target distance is on the order of several hundred meters. With resolution of approximately 10^{-7}m, measurement of Station geometry at a 1 km distance would still be accurate to within 10^{-4}m (or $\pm$ 0.1 mm). Combination of this geometrical measurement of the Space Station configuration with finite element analysis would provide the details of stress distribution in the structural members, which could subsequently be used as feedback for structural control. Eventually, this optical sensor system could ensure that no part of the Station would exceed its designed stress during orbital maneuvers or docking.

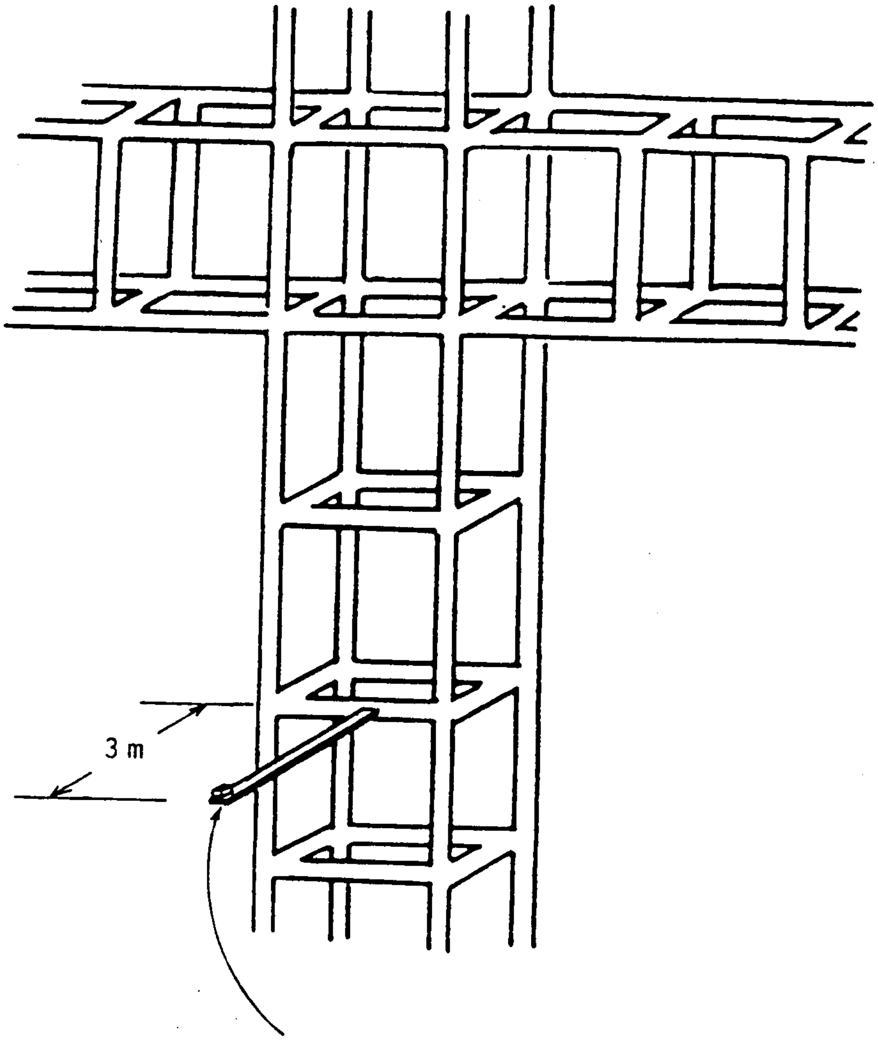


Figure 3-4. Space Station OP2L2 Strain Sensor.

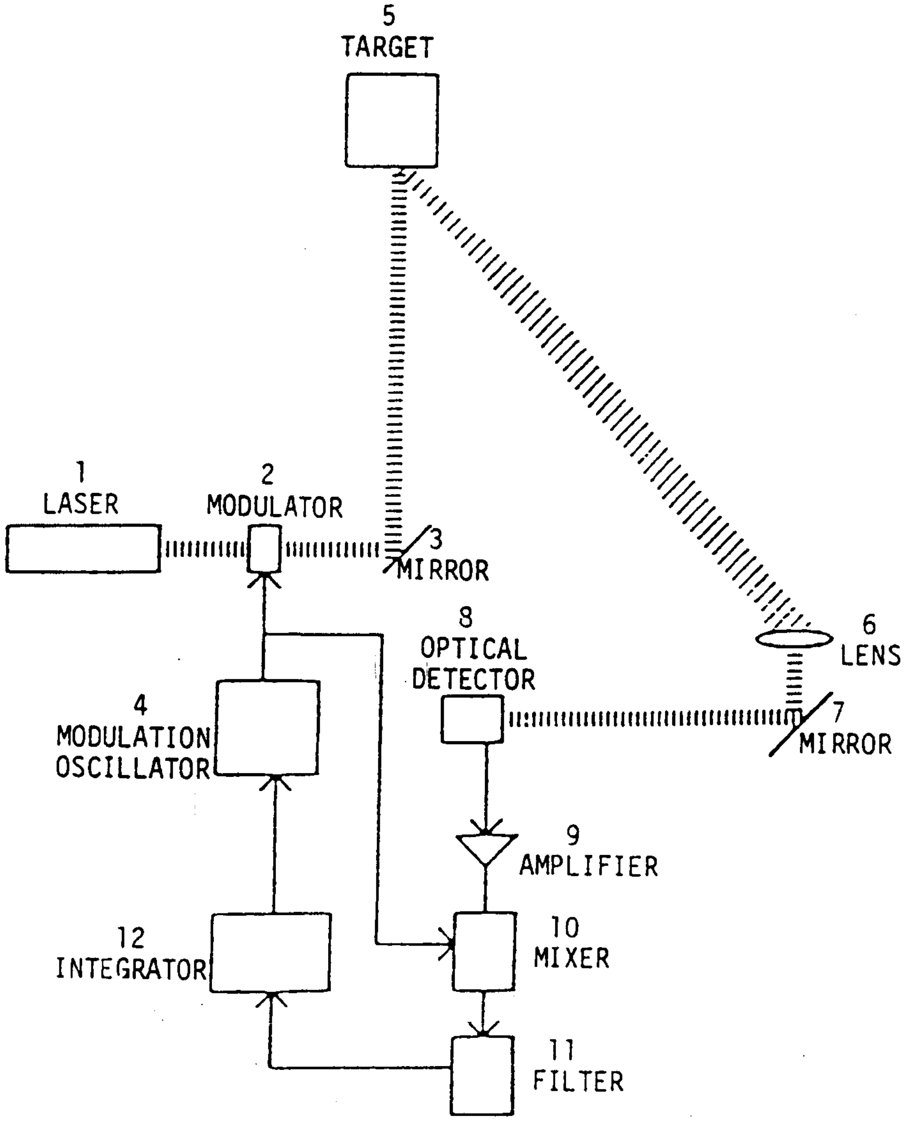


Figure 3-5. Optical Pulsed Phased Locked Loop (OP2L2) Block Diagram.

3.5 DATA ACQUISITION

The data acquisition system required for the three types of sensors mentioned in the previous section differ substantially. For AE data acquisition, the analog signal coming from the transducer must be preamplified to provide an intermediate gain stage for the acoustic emission signal. Most AE transducers do not have built-in preamplifiers. The preamplifier is necessary not only to increase signal strength, but also to provide a low impedance output enabling long output cables to be utilized. If the original transducer signal has a high impedance, this signal can only travel short distances (typically 1.0 m) without serious loss of signal, thus necessitating a preamplifier to boost the signal to the wideband conditioning amplifier/signal processor. A typical commercial preamplifier can usually provide either a 40 dB or 60 dB fixed gain with exceptionally low noise operating over a wide frequency range. A commercially available preamplifier that could be utilized with AE transducers for the Space Station might have the following specifications (e.g., model 160 B preamplifier, manufactured by Acoustic Emission Technology Corporation) [7]:

Gain: 60 dB (x1000)
Frequency Range: 1 kHz to 2 MHz
Output: 10 Vp-p into 50 ohms; typically 17 Vp-p at 175 kHz
Power: $\pm$ 12 Vdc at 20 mA ($\pm$ 9 Vdc to $\pm$ 20 Vdc nominal)
Dimensions: (Length x Width x Height) 13.6 cm x 8.4 cm x 4.8 cm
Weight: 250 gm
Environment: 0° to 40°C
0-95% relative humidity (non-condensing)

The AE data acquisition system also requires a wideband conditioning amplifier/signal processing unit. This device provides the final amplification and, depending on the model, initial digitization of the analog AE signal for real-time monitoring. The signal processing unit (SPU) consists of a postamplifier and filter(s), a threshold comparator, and circuits to monitor the signal level. The device would also supply power to the preamplifiers if they are required.

The SPU compares the fully-amplified AE signal to a reference voltage (the threshold) to produce a digital AE signal. Whenever the AE signal rises above this threshold voltage, the SPU produces a pulse which is sent to the system mainframe. Using these pulses, the mainframe can distinguish events, their duration, etc. The threshold can be either a fixed voltage, or set to automatically "float" above the peak value of the background noise (See Figure 3-6) [8].

The gain for the SPU is usually variable in two ranges, providing up to 100 dB total system gain when combined with the gain from the preamplifier. There may also be plug-in filters to constrict the bandwidth of the SPU (if required) to match the AE characteristics of the transducer/station signal signatures [8]. A fully-amplified AE signal is continuously available; it is useful for oscilloscope presentation, analog recording, transient recording, or for frequency analysis [8].

The following specifications, for an Acoustic Emission Technology Corporation Model 208 [8], are illustrative of typical AE signal processing units available commercially today that might be used aboard the Space Station:

Number of Channels: Two
Gain: 0 to 40 dB
Frequency Range: 6 kHz to 1.25 MHz
Threshold: Selectable by logic level, fixed or auto
AE Output Voltage: 10 Vp-p minimum into 50 ohms
Environment: 0^{o}-45^{o}C
0-95% relative humidity (Non-condensing)

3.6 DATA HANDLING

The inflight data collection system for the Space Station is expected to be comprised of not only the various data sensor (AE transducers, optical fiber sensors, OP2L2) and signal conditioning subsystems (i.e., analog-to-digital conversion, increased signal gain, etc.), but also a data and handling storage sytem.

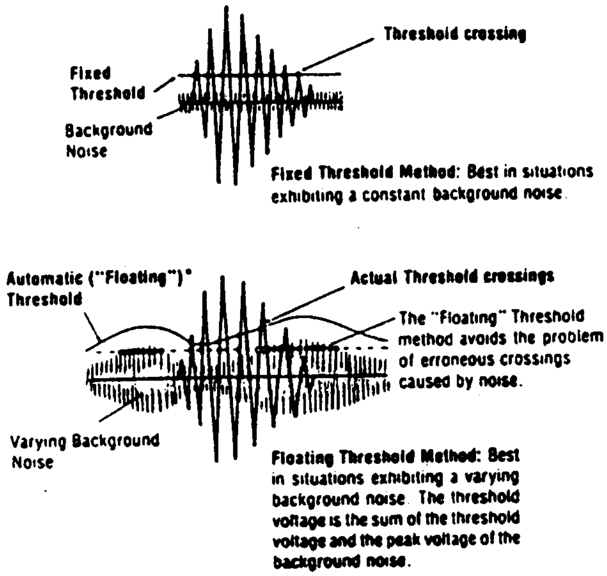


Figure 3-6. Comparison of Fixed vs. Floating Threshold Techniques [8].

The data handling and storage system must have the capabity to process and record information from many separate digital and analog data channels. The system would be user programmed to provide both the desired timing for data collection and the binary word length used in the analog-to-digital conversion of the analog data. Use of a data handling and storage system similar to that used on the LDEF mission would adequately meet inflight processing and recording requirements. LDEF utilized an Experiment Power and Data Systems (EPDS) for data collection and handling; a functional block diagram of the EPDS is shown in Figure 3-7.

Adapting the experience gained from LDEF for handling in-flight data, both a Synch and time code would be generated by the data system immediately prior to data collection/recording. The Synch code identifies the onset of a data scan, the specific environment, and the location of the data system. Data would be collected using 5 consecutive data scans (to guarantee data accuracy) which would be stored in the buffer memory of the data system. The buffer memory would then be transferred to a single-track tape recorder (e.g., Lockheed Mark V Type 4200) after the 5 data scans for eventual dumping to ground for analysis each orbital period [9].

3.7 TELEMETRY

The AE sensor instrumentation requires approximately 20 analog/digital (with various sampling rate) telemetry monitors with 8 bit resolution to carry the health status information and scientific data. The two other types of sensors, optical/fiber interferometric and the OP2L2, require no more than ten channels each. The monitors would have at least the following sampling rates:

- Sampling rate = 1 Hz
  - High voltage monitor
  - Commutated monitors
- Sampling rate = 100 Hz (or slower)
  - Low voltage monitor
  - Current monitor
- Sampling rate = 1000 Hz (or slower)
  - Data monitor
- TOTAL BIT RATE = 15 Kbps

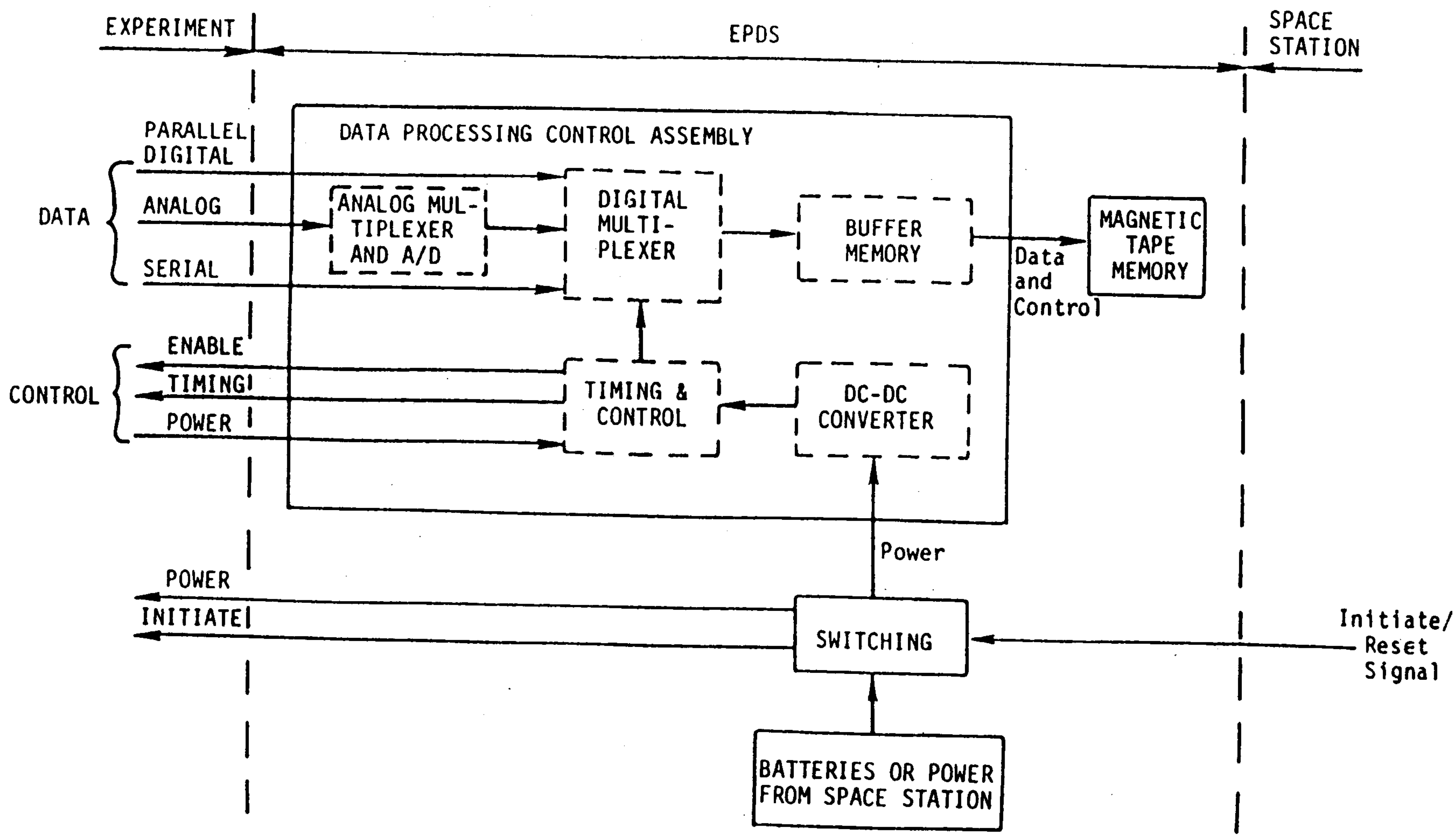


Figure 3-7. EPDS Block Diagram [10].

The sampling rate for the raster monitors can be adjusted to the telemetry system capabilities. Limiting the sampling rate to 1000 Hz or less precludes real-time monitoring for sensors having much higher bit rates during the early missions of the Space Station. Increased data transfer telemetry rates for growth configurations would provide real-time monitoring when deemed necessary.

3.8 CREW ACTIVITIES

The automated method for reviewing the health and scientific data required little (if any) crew activities during normal operation, other than "tracking down" anomalous events during information feedback (from the ground station during early missions) utilizing hand-held NDE scanning detectors. Since the autonomous instruments would be operating continuously during the mission, a timeline for only the initial instrumentation set up is provided in Table 3-1.

The following assumptions are implicit in the timeline shown in Table 3-1.

A. General

1. One EVA crewmember is located in the STS cargo bay, the MRMS has been activated and checked out, and one Manned Manuevering Unit (MMU) is available for use.
2. The basic Space Station structure has been erected.

B. Flight One - Sensor Hook-up/Calibration

(Sensor Placement)

1. The acoustic emission sensors are not imbedded in the Space Station structure and must be manually affixed.
2. Any Space Station structure that will be monitored by fiber-optic interferometric techniques that is made of a composite material will already have optical sensing fibers imbedded in them during fabrication.

Table 3-1. TDMX 2072: SPACECRAFT STRAIN AND ACOUSTICS SENSOR CREW TIMELINE EVALUATION

FUNCTION/TASK	TIME MANMINUTES	METHOD	FEASIBILITY*	COMMENTS/ASSUMPTIONS
FIRST FLIGHT				
• Dock with structure				See Assumptions - General
• Checkout Mobile Remote Manipulator System (MRMS)	30	EVA	1	
• Deploy transfer containers	110	MRMS/EVA		
• Translation from STS Cargo Bay to appropriate sites on Space Station	(75)	MRMS	1	See Assumptions - B.4
• Install Monitoring Sensors	750	EVA/MRMS/IVA	4	No Data on Space Installation
• Acoustic Emission	[540]	EVA/IVA		See Assumptions - B.1
• AE sensor placement at high stress/strain structural areas (X20)	(400)	EVA	3	See Assumptions - B.3
• Connect electrical/instrumentation cables (X20)	(2)	EVA/IVA	2	See Assumptions - B.6
• Check/verification of operational status of AE sensors (X20)	(100)	EVA/IVA	1	See Assumptions - B.7
• Fiber-Optic Interferometer	[70]	EVA/IVA		See Assumptions - B.2
• Connection of fiber-optic interferometer cables with instrumentation cables	(40)	EVA/IVA	2	See Assumptions - B.6
• Check/verification of operational status of fiber-optic sensors	(30)	EVA/IVA	1	See Assumptions - B.7
• OP2L2	[160]	EVA/IVA/MRMS	4	Still in Development
• Positioning of laser/modulator/optical detector housing OP2L2	(60)	EVA/MRMS	1	See Assumptions - B.5
• Affix OP2L2 to mounting boom	(50)	EVA/MRMS	3	See Assumptions - B.3
• Connection of electrical/instrumentation cables	(20)	EVA/MRMS	2	See Assumptions - B.6
• Check/verification of operational status of Op2L2	(30)	EVA/IVA	1	See Assumptions - B.7
• Stow MRMS	10	MRMS	1	

*1-Proven 2-Easy 3-Hard, 4-Questionable, 5-Not Possible

3. Utilization of the OP2L2 sensor will require a mounting boom extending outward from the main station structure at a predetermined optimum viewing site and that this mounting structure will have been erected on a previous Space Station construction flight.
4. Sensors are transported within an easily maneuvered, detachable container stowed in the STS cargo bay.
5. One crewmember operating MRMS for container translation and another EVA crewmember performing attachment/positioning.
6. Crew will manually (EVA or MRMS) connect electrical and instrumentation lines from sensor to external/internal Space Station connectors.
7. Crew participation is limited to observation for nominal monitoring operations, possibly some EVA or MRMS assistance in contingency operations requiring manual NDE scanning of a suspect area.

3.9 GROUND STATION

Information is retrieved from the data handling (e.g., modified EPDS) recorder by transcribing to a ground-based computer compatible tape recoder (e.g., Kennedy Model 9832). Utilizing the experience gained from the LDEF mission for in-flight data collection, sensor information from the instrumentation on the Space Station would be collected as 10-bit data words. Telemetry data from the experiments would be stored in 8-bit characters, bearing no fixed phase relationship to the original 10-bit data words. The tape record would be searched for the Synch code; the time and data words would be reconstructed, converting them to a more useful form. These flight data tapes would be processed on a ground-based mainframe computer to analyze and massage the data, presenting results in appropriate formats. A ground-based software program would allow: (1) sequential searching for the Synch code; (2) locating and converting the time code, and (3) reconstructing and converting the 10-bit data words to appropriate values for each channel [9].

3.10 SUPPORT EQUIPMENT

3.10.1 Data Handling Support Equipment

The data handling system ground support equipment would consist of a magnetic tape memory controller, a data display monitor, a computer compatible tape recorder (with tapes), and interfacing cables. However, during preflight checkout of the experiment, portable hardware is required to check system performance (a) during assembly, (b) before and after flight acceptance tests, and (c) during prelaunch testing [9]. The portable ground support equipment would consist of a power supply, a pulse command sequencer, dummy loads, microcomputers (e.g., TRS 80), and a channel selector probe. The command sequencer would simulate the data handling system command pulses and timing sequences to the signal conditioning unit (SCU). The dummy loads would simulate a transducer signal to the SCU; the channel selector probe would sample each of the SCU outputs. This portable equipment permits checking of the SCU's independently of the data handling system [9].

3.10.2 AE Calibrator

An acoustic emission (AE) calibrator is required to check/calibrate the AE instrumentation. The following specifications are for a commercially available model (PAC No. C-101), manufactured by Physical Acoustics Corporation, that might be utilized for this experiment:

Broad Pulse Repetition Rates: 1 Hz - 1 kHz
Energy Levels: 30, 60, 90 V
Power: 115 Vac, 100 mA
Temperature: $25^{o}C \pm 5^{o}C$
Dimensions (Length x Width x Height): 17.2 cm x 14 cm x 6.4 cm
Weight: 1.14 kg

3.10.3 AE Analysis

Once the telemetry data from the acoustic emission (AE) sensors has been (ground) recorded, AE analysis would be required; note that this could be extended to real-time monitoring if the analytical hardware were aboard the Space Station. Several commercial products/systems are available for complete AE analysis (compatible with both real-time and post analysis) of this Space Station experiment [11-16]. Many of these have features such as:

- Multi-channel data acquisition (e.g., up to 128)
- Complete AE parameter data set
- User programmability
- Self-diagnostics
- Stand alone system
- Expandability
- Portable/self-contained
- High data rates
- Parametric filters

Use of this analytical hardware aboard the Space Station would not only provide enhanced AE detection, display, and analysis, but would also serve to locate any AE events.

REFERENCES

1. Acoustic Emission Technology Corporation Data Sheet, "MC500 Sensor-Microminiature."

2. IQI Product Brochure, "A Flat Response AE/UT Transducer."

3. Claus, R.O. and J.C. Wade, "Distributed Stress Measurement in a Rectangular Plate Using an Array of Optical Fiber Sensors," in Analysis and Acoustooptical Measurements of Bulk and Surface Acoustic Wave Fields, VPI Technical Report, Sept. 30, 1983, pp. 200-218.

4. Wade, J.C., et. al, "Optical Fiber Systems for the Non-Destructive Evaluation of Materials," presented at VACON '82, Hampton, VA, October 1982.

5. Crane, R., "Fiber Optics for a Damage Assessment System for Fiber Reinforced Plastic Composite Naval Structures," ARO/DARPA Workshop on Nondestructive Evaluation of Polymers and Polymer-based Composites, (Wrightsville Beach, NC), April 1982.

6. Wade, J.C., P.S. Zerwekh, R.O. Claus, "Detection of Acoustic Emission in Composites by Optical Fiber Interferometry," Proc. IEEE Ultrasonics Symp. 2, 849, (1981).

7. Acoustic Emission Technology Corporation Data Sheet, "Model 140B/160B Preamplifiers."

8. Acoustic Emission Technology Corporation Data Sheet, "Signal Processor Units, Model 200 and 208."

9. Giguere, R.P., SD/LDEF Technical User's Manual, The Aerospace Corp. (El Segundo, CA), September 1981, AD-A113504.

10. Motley, W., J. Cox, and J. Bene, Experimental Power Data System for Long Duration Exposure Facility, NASA Langley Research Center, Hampton, Virginia (1 August 1978, rev. 18 April, 1979).

11. Bruel and Kjaer Instrumentation 1985 Catalogue, "Acoustic Emission Measuring Equipment," p. 44.

12. Physical Acoustics Corporation Data Sheet, "FRP-1, 16 Channel Industrial Acoustic Emission System."

13. Physical Acoustics Corporation New Product Data Sheet, "Atlas 7016."

14. Acoustic Emission Technology Corporation New Product Announcement Data Sheet, "AET 5000 A."

15. Physical Acoustics Corporation New Product Data Sheet, "3000 Series AE Analyzer/Locator."

16. Physical Acoustics Corporation New Product Data Sheet,"Spartan AE System."

4.0 TDMX 2111 - DEPLOY AND TEST LARGE SOLAR CONCENTRATOR

CONTACT: Dr. Edmund J. Conway, MS 160
NASA LaRC (804) 865-3781

This TDM is the first in a series of four which develop and evaluate the capability of transmitting power through space using laser generated light beams. In this mission, a solar concentrator is to be fabricated and evaluated. In the second mission (TDMX 2121), candidate solar-pumped laser systems are evaluated by placing them near the focal point of the concentrator. In the last two experiments, two different mechanisms for using the energy transmitted along the laser beam are evaluated: (1) photovoltaic devices for direct conversion of optical power to electric power (TDMX 2122), and (2) propulsion devices which produce thrust directly from a concentrated laser beam (TDMX 2322).

With the exception that the last two missions may be interchanged, the missions must be completed in order, as each experiment relies on capabilities implemented in previous missions. Because the later experiments rely on capabilities developed in earlier missions, the later missions also establish the requirements of the earlier missions. The entire group must therefore be designed as a coordinated unit.

4.1 GOALS AND OBJECTIVES

For successful solar pumping of lasing materials, energy must be collected from a relatively large area and focused or concentrated on the lasing material. The goal of this mission is to develop and demonstrate the technology necessary for the fabrication of practical solar concentrators for space applications. In this mission, a reflective concentrator would be implemented which is capable of collecting sufficent energy to satisfy the requirements of the related experiments TDMX 2121, 2122, and 2322. The structure required would be large and would challenge technology because of the strict requirements for mechanical stability to maintain proper focusing of the energy. The objective of this experiment is to implement a large concentrator in space and evaluate its performance over an operational period, assessing both its performance upon initial setup and its ability to maintain that performance without continuing attention. A secondary objective is to determine the required operating procedures for a large solar concentrator on the Space Station, e.g., determine parameters which must be continuously controlled versus those which, once set, remain sufficiently stable to eliminate the need for adjustment.

4.2 METHODOLOGY

The design of the solar concentrator is driven by the requirements of the three experiments which derive their power from it. Specifically, the concentrator must focus sufficient light into a region where it may be efficiently coupled to various lasers including a blackbody absorption CO_2 laser and a laser using iodine compounds. After the evaluation of their performance, the lasers will be used to supply energy to the laser-to-electric-energy conversion experiment and the laser propulsion experiment. The laser propulsion experiment will not operate unless the input power exceeds a threshold level presently estimated to be 1 kW, however, it can accept light over a broad range of wavelengths including the infrared light emitted by the more efficient blackbody laser. Therefore, the concentrator design analyzed here was configured for compatibility with the blackbody laser and the laser propulsion experiment. The other lasers and the laser-to-electric-energy system could then be operated from the same concentrator but with possibly lower efficiencies or power levels. Please note that the design is preliminary and is based on anticipated requirements. The ultimate implementation may differ from the present design for many reasons including:

1. Changes in laser requirements during their development possibly may reduce input radiation requirements or change the required shape of the concentration region.

2. Development of concentration technology may produce a concentrator with improved optical characteristics which might better satisfy the requirements of both the blackbody and the iodine compound lasers.

4.2.1 Description

This mission would be implemented by assembling the large solar concentrator system adjacent to the Space Station and connecting it to a single point to the keel extension. An orientation is selected so that the system is operational for a minimum of one-half of every orbit (the Space Station daytime) but will continue to track even during earth shading. This coverage is available for six months out of any year but, as will be discussed shortly, can be extended to full year coverage by periodic reorientation.

The solar concentrating system's tracking motions are simplified by orienting the tracking axis (i.e., the heliostat to concentrator axis) perpendicular to both the Space Station's flight path and nadir. The sun can then be tracked by the heliostat mirror, the motions of which are similar to the motions of the Space Station's solar panels.

An alternate orientation was considered which placed the solar concentrator system atop the Space Station using the same tracking position. Of the two orientations, the latter would be the most desirable because there would be no shading by the Space Station during the operational period each orbit. However, due to the size of the structure, shading of the Space Station's solar panels is a possibility that must be studied. At this point in the experiment definition, the most practical orientation for discussion would be the former.

4.2.2 Design Considerations

Figure 4-1 is a drawing of the physical layout of the solar concentrator system. The mount to the station would consist of a truss extension arm extending from the Space Station some 10 to 20 meters. A single connection to the station's main structure is made and power, data, and coolant lines are tapped at that point. This extension arm is connected to the solar concentrator base through a joint that allows connection of primary functions (power, data, and coolant) and allows reorientation of the entire system. This reorientation involves rotation of the structure 180 degress about the horizontal once every six months to

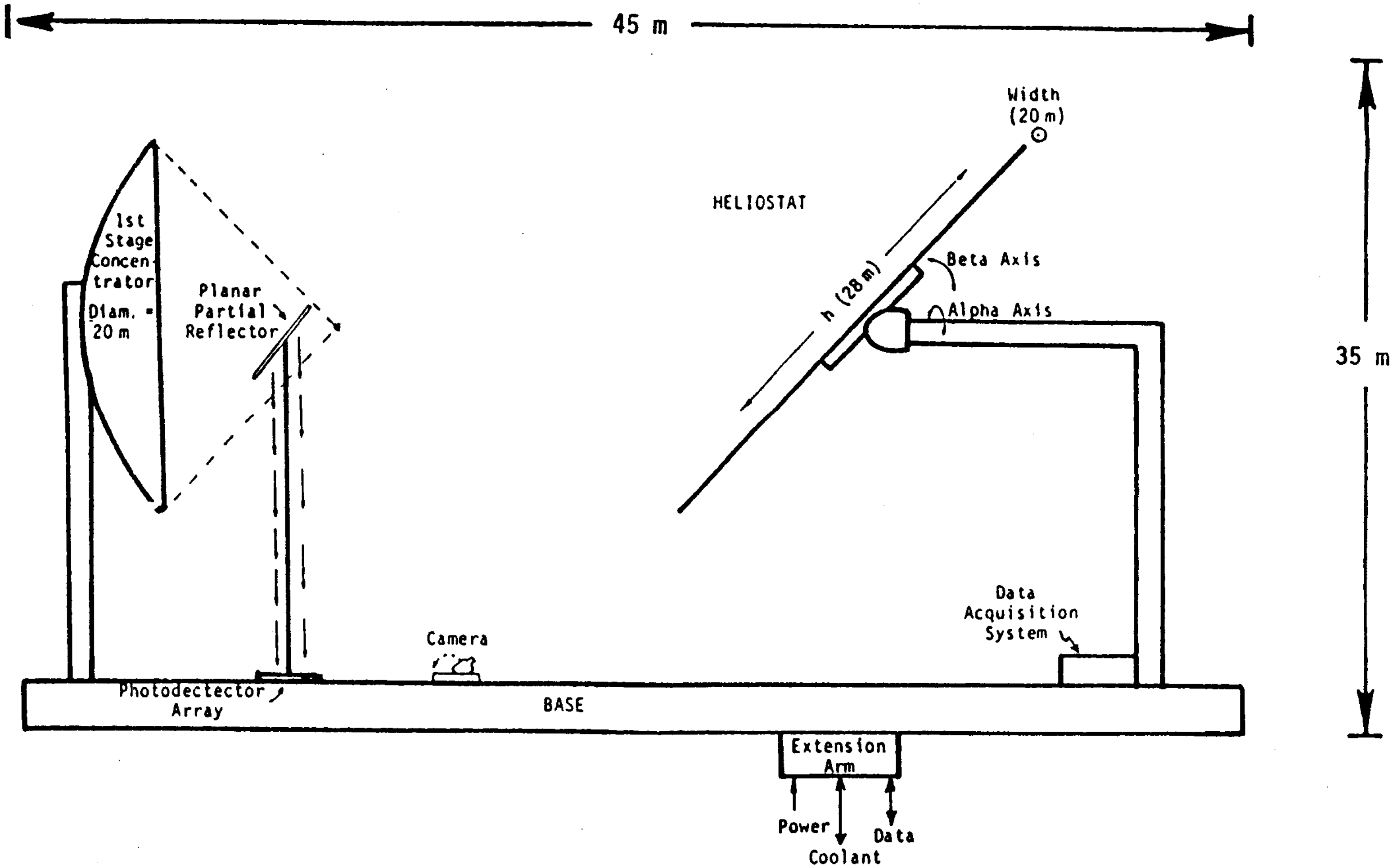


Figure 4-1. Physical Layout of Solar Concetrator System.

allow year round solar tracking. It is this rotation that determines the minimum length of the extension arm (i.e., the solar concentrator must be mounted far enough from the Space Station to prevent interference during this reorientation). The extension arm joint also allows for fine alignment of the structure after assembly.

The solar concentrator system consists of a base, heliostat mirror, parabolic mirror, and support instrumentation. The base would be a rigid trusslike structure similar to the Space Station's keel and would span approximately 45 meters. It would provide mounting for the other features and allow internal routing of signal lines and coolant and for experiments TDMX 2121, 2122, and 2322, would provide room for plumbing, the laser beam, and related optics. The heliostat structure consists of a right-angle truss mount, a tracking axis with two degrees of motion, a deployable flat mirror frame with individually positioning cells, and 140 replaceable flat mirror panels each mounted in a frame cell. The overall size of the mirror face is 28 meters by 20 meters which will provide continuous full irradiation of the parabolic concentrator during the operational period. The parabolic mirror was patterned after the TRW advanced sunflower mirror that was proposed for the solar dyanamic system (that would replace the Space Station's solar panels). This mirror would be assembled and folded on the ground and deployed once it is in place. It would have a diameter of 20 meters and an F stop of f/0.6 (focal length of 12 meters).

The heliostat flat panel mirrors would be approximately 1 meter by 4 meters and due to weight considerations would probably be constructed out of a lightweight graphite-strengthened fibrous material similar to the material used in the parabolic mirror. The reflecting surface would be covered with an aluminum alloy, such as with aluminized mylar or vaporized aluminum and top coated with magnesium fluoride to protect against scratching and oxidation. The aluminum surface was selected for its lightweight and adequate reflectance at ultraviolet wavelengths. The overall reflecting surface of the heliostat, including mounting errors and surface irregularities in each mirror panel is expected to achieve an average surface pointing error of less than 2.5 milliradians.

The heliostat mirror would have two degrees of rotation referred to as the Alpha and Beta axes (See Figure 4-1). The Alpha axis allows continuous tracking of the sun's motion by revolving 360 degrees during each orbit. The Beta axis provides the small adjustment needed to track the sun's motion derived by the Earth's rotation about the sun. The drive mechanism for these two axes would be controlled by an intelligent tracking system and would be capable of making small tracking corrections during any orbit. The tracking accuracy should be 180 arc seconds with a tolerance of ± 90 arc seconds. Finally, overall pointing inaccuracies due to tracking and structural vibrations is expected to add less than 3.0 milliradians to the surface pointing error. It was found that this pointing error criteria could be met under the present design if the long-duration vibrations from the Space Station, that are felt by the heliostat mount, are less than 0.075 Gs. Additional strengthening would be needed in the heliostat's mount for larger accelerations to be tolerated and while maintaining the required pointing stability.

The parabolic mirror is a deployable structure of petal type design. An appropriate space applicable mirror has already been designed, namely TRW's Advanced Sunflower parabolic reflector. The solar concentrator experiment could provide the means for proving technologies such as this. Thus, in this preliminary study, the TRW parabolic mirror design was used to help define the structural requirements. It is recommended that the area where this parabolic mirror is mounted be considered a testbed that would be available to other experiments involving solar concentration and analysis.

The parabolic mirror would have a reflective coating of an aluminum alloy similar to the heliostat. The expected slope errors and the resulting average pointing errors should be less than 5.0 milliradians. For a 20 meter diameter mirror, this pointing error translates into a maximum concentration of 2500 (a 40 centimeter spot size) when observing the sun. Combined with the heliostat pointing errors mentioned above, the total system provides a concentration of about 1150 or a 60 centimeter spot moving within a 60 centimeter envelope, the significance being that a second stage concentrator placed at the focal point could reduce the envelope again to 40 centimeters (effectively eliminating the tracking and collector pointing errors). Finally, assuming the solar energy flux of 1350 watts/m^2 and a 90% reflection of the total system, a figure for total energy delivered to the focal point is 382K watts or an energy flux of 300 watts/cm^2 assuming a 40 centimeter spot size.

4.2.3 Sensors

The solar concentrating system would have various sensors for monitoring system performance. Specifically, these sensors would provide feedback for the tracking controls, help monitor collector and concentrator efficiencies, analyze thermal effects, help study the stresses and vibrations in the structure and provide system diagnostics. Table 4-1 is a list of the type, amount, and function of the sensors that would be used.

The luminance sensor group would consist of a ring of photodetectors about the perimeter of both the heliostat and the concentrator, a small array of photodetectors set at the focal point during setup, a large array mounted at right angles to the focal plane, and many other photodetectors placed strategically about the experiment (see Figure 4-1). The photodetectors on the mirror perimeter sense the relative intensity of solar radiation and help analyze direction and coverage. Both the small and large photodetector arrays are used as tracking and focal point analyzers with the large array observing only a small reflected portion of the concentrated radiation. The small array will be installed only during setup when only a small amount of radiation will be focused (such as from the diagnostic laser). These arrays determine the size, position, and motion of the spot size of the concentrated light by analyzing the location as well as the intensity of the image falling on the array. Finally, there will be a group of photodetectors placed at various locations on the experiment to measure solar radiation due to reflections.

The luminance sensor group would be scanned periodically at a rate sufficient to allow the necessary tracking adjustments in the heliostat system. Focusing irregularities that change with time arise from two

4.2.4 Data Acquisition and Control System

One of the objectives of the solar concentrator experiment is to study the capability of long term unattended operation. To achieve this objective a minimum amount of control from the Space Station would be required. This points to the need for a stand alone Data Acquisition and Control system that could perform the necessary sampling and tracking functions. Such a system would communicate with the Space Station data network and could pass all data available to it for performance verification. In addition, real-time control of the stand alone system would be possible either from the Space Station or the ground.

A diagram of the signal flow within the solar concentrator experiment is shown in Figure 4-2. The diagram demonstrates how control of the system is achieved by feedback from the local sensors. The three blocks that make up the stand alone system are: the data acquisition block, the control block, and the Subsystem Data Processor block (SDP). These blocks are interconnected with the system "intelligence" residing in the data acquisition area.

A list of electronic hardware associated with the stand alone system is given in Table 4-2. Since the solar concentrator experiment provides a testbed for other experiments, the use of common resources within the data system for these experiments would be very desirable. Growth in the way of electronic modules to both the acquisition and control systems to perform other functions would also be possible.

The Subsystem Data Processor system (SDP) has been designated to provide the communication link between the Space Station data network and any payload experiment such as the solar concentrator. Its job is to format transmittable information into a standard form for passage through the Interface Devices (ID) to the Space Station. It also receives and buffers data for use by the payload experiments. Finally, the SDP will have standardized modules (slices) available that can perform various processing, computation and storage functions. It is not clear at this

PRECEDING PAGE BLANK NOT FILMED

61-62

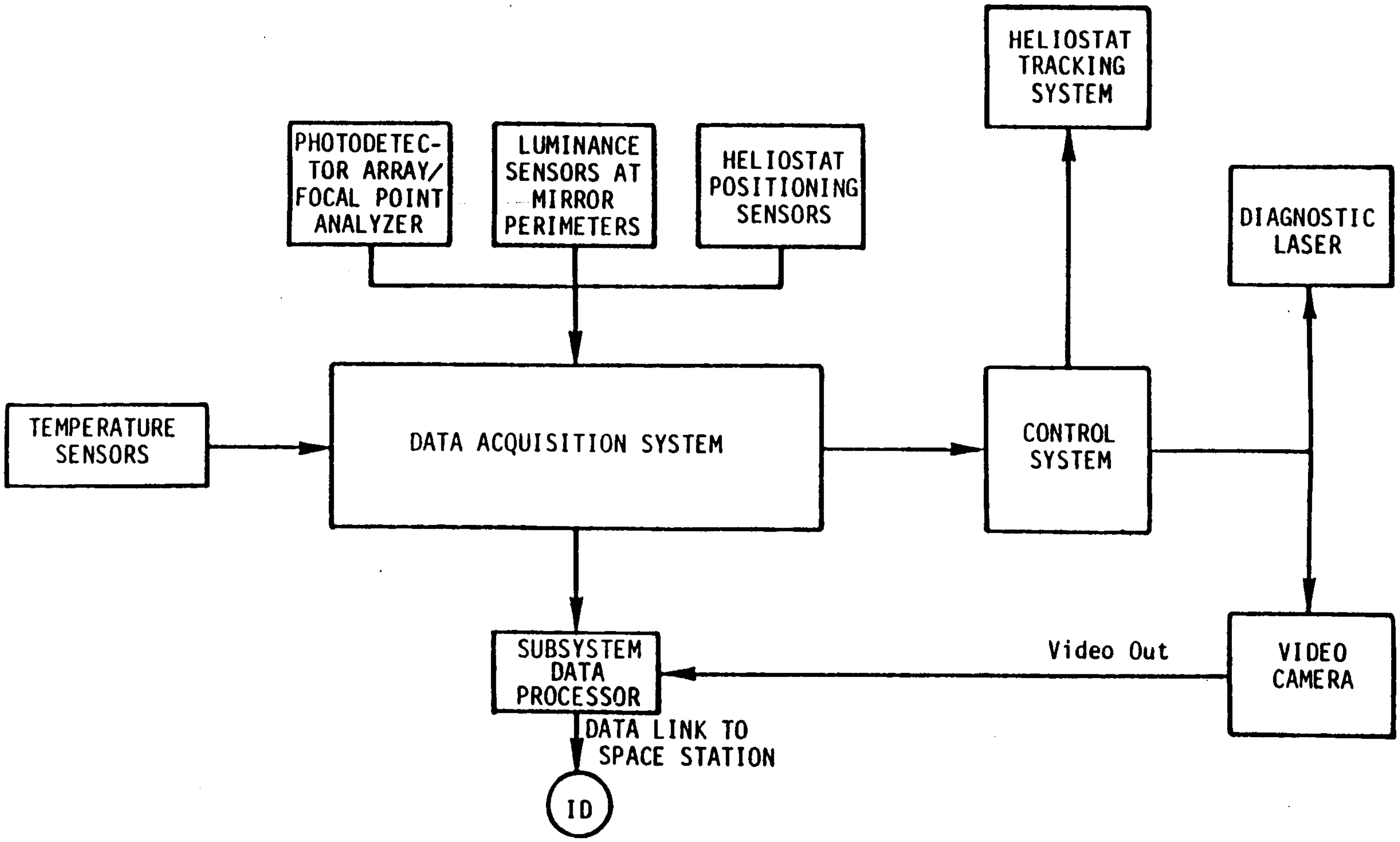


Figure 4-2. Signal Flow Diagram of the Solar Concentrator Experiment.

Table 4-2. List of Electronic Equipment in Stand Alone Data System

Equipment	Function Block	Description
Microprocessor	Data Acquisition	Provide system "intelligence." Includes resident tracking software.
Signal Conditioning	Data Acquisition	Includes multiplexed sensor interfaces and analog to digital conversion.
Mass Storage (1 Mbyte)	Data Acquisition	Short term data storage for tracking and status checking.
Motor Drivers	Control	Driving electronics to control heliostat rotational motors and cell positioners.
Communications	SDP	Reformatter for two-way communications with Space Station.
Expansion Modules	Data Acquisition	Additional memory and programs for shared experiments.
Expansion Modules	Control	Additional control electronics for shared experiments.

point how many of the stand alone data system functions could actually be done by the SDP system, but any that can would help simplify the final design.

4.2.5 Data Communications and Handling

An information network will be available between the solar concentrator data system and the Space Station resources. This network goes by way of the Subsystem Data Processor (SDP) and the Interface Devices (ID) through the optical data distribution payload network and finally to the Space Station's Optical Data Distribution NETwork (ODDNET). This two way network will allow operators (both Space Station and ground) to monitor and control the concentrator experiment. However, operator control will be at a minimum since the purpose of the experiment will be to achieve long periods of unattended operation. Monitoring the system performance without intervention is therefore the main goal.

The stand alone data acquisition system is discussed in the previous section. Its function is to sample large amounts of sensor data and make control decisions based on that information. This accounts for its large short-term data storage requirements. A periodic dump of a portion of this data along with a summary of command decisions and system status would be the typical information passed to the Space Station. By request, a real-time sample of all the sensors could be sent or a full dump of all the data systems short-term memory could be passed. In addition, real-time observations with the video camera would be available.

The data received by the Space Station will be processed, stored, and periodically sent to a ground facility. The processing capabilities needed by the Space Station for the solar concentrating experiment include data compression and storage, analysis and response to the experiment status, and control of real-time communications between operator and the experiment. Data compression and storage functions require periodic use of the Space Station's computer resources. The other functions require resources only if and when they are needed. A summary of the processing functions and the effect on system resources is given in Table 4-3. Disk

Table 4-3. TDMX 2111 Space Station Data Processing Functions

Function	Resources Needed
Data Compression and storage	CPU Time Network Time 4 MBits Disk Storage
Performance Verification	CPU Time
Alarm on Failure	CPU Time Crew Time
Real-time Control	CPU Time Crew Time Network Time
Program Code Allocation	1 MBits Disk Storage 1/2 MBits Memory (RAM)

storage is based on the worst case where all the data is transmitted during the non-operational period or any specific period of each orbit (such as over a certain receiving station). If real-time telemetry is performed at some point, then no disk storage resources would be needed except for program storage.

4.2.6 Telemetry Requirements

The observation and control of this experiment will require both station to ground and ground to station data communications. In the early phase of the experiment a continuous link is needed for station to ground so that system status and performance can be observed in real-time. During that same testing period, voice and video would also be transmitted. Once the system is operational, voice and video transmissions would rarely be needed and digital data would be sent in short bursts during each orbit. Ground to station communications would be infrequent and would require minimal voice and digital data.

The telemetry requirements are summarized in Table 4-4. The generation rate, duration, and frequency are for the worst case which would be during setup and testing of the experiment. Some two-way real-time interaction will be necessary throughout the experiment so that experimenters located on the ground could override any of the system functions.

4.2.7 Ground Processing

Data received at the ground station would need a certain amount of processing before distribution to experimenters. During the setup phase an operator would be available onsite and would require fast turn around of received data to complete setup functions. At that time, online storage and moderate processing time would be required. Once operational, data would be formatted and stored on magnetic tape for distribution within twenty-four hours to the designated major experimenter (NASA facility). It should be noted that, once operational, the success of the mission is not greatly affected by the speed of the distribution of the data.

Table 4-4. Telemetry Requirements

Station to Ground

Function	Digital	Video	Voice
Generation Rate (kbps)	5.0	0.0	NA
Duration (hours)	24.0	.25	0.25
Frequency (per day)	1.0	1.0	1.0
Delivery Time (hours)	24.0	0.0	0.0
Reliability (%)	99.0	0.0	100.0
Interaction (y/n)	yes	no	yes

Ground to Station

Function	Digital	Voice
Generation Rate (kbps)	1.0	NA
Duration	0.25	0.25
Frequency (per day)	1.0	1.0
Reliability (%)	100.0	100.0
Interaction (y/n)	yes	yes

4.2.8 Crew Activities

Construction and setup of such a large space structure as this will require a well coordinated effort among Space Station personnel. A large amount of EVA time will be needed for material handling and assembly. Personnel skilled in spacecraft systems would be best suited for performing the many EVA functions. Once operational, only a minimal amount of crew time will be needed to perform the periodic and servicing operations. It is expected that various configuration changes will be performed during the mission, such as reorientation of the structure or adding experiments, and would require a certain amount of skilled crew activities.

The solar concentrator's construction will require both deployment and assembly. It is possible that as many as three crew members at the technician skill level would be needed during the most demanding construction phases. Use of the mobile manipulator is very likely and an operator skilled in its use will be needed. His time will account for much of the IVA time needed for this mission. Table 4-5 is a summary of crew activities during contruction and setup.

During normal operations, crew activity would be limited to periodic observation and servicing of the system. The functions would include realignment and calibration of the equipment, locating damaged materials, and removal and replacement of defective heliostat mirror panels. Minor calibration would be done through communications with the data acquisition system. Any hard realignment would require some EVA time as would any repair servicing.

Every six months a major configuration change will be performed that will require a moderate amount of crew time. This change would involve rotation of the entire solar concentrator structure 180 degrees about the extension arm joint. This would be a delicate maneuver requiring the mobile manipulator and possibly two skilled technicians.

Other configuration changes would involve the setup of interrelated experiments, in particular experiments TDMX 2121, 2122, and 2322. Crew activities relative to the solar concentrator experiment would include any

Table 4-5. Summary of Crew Activities

Functions	Estimated EVA Time manhours
1. Unpack Shuttle Bay	4
2. Deploy Extension Arm and Attach to Station	12
3. Deploy Base and Attach to Extension Arm	16
4. Deploy Heliostat Mirror Mount and Install	12
5. Deploy Heliostat Mirror Frame and Install	20
6. Install Individual Mirror Panels	20
7. Deploy and Mount Parabolic Mirror	12
8. Install Partial Planer Reflector	2
9. Install Data Acquisition System	2
10. Connect Power and Signal Lines	8
11. Install Diagnostic Laser and Video Camera	2
12. Install Photodetector Arrays	2
13. Fine Align all Optics	16
TOTAL ESTIMATED EVA	128

modifications needed for compatibility with these additional experiments. Crew operations for these additional experiments are discussed in their respective sections.

4.2.9 Design Specifications

The section summarizes the specifications for system setup and operation, in particular the power, weight, volume, and thermal requirements. The setup requirements include both launch weight and volume and the final operational dimensions. A summary of the physical requirements are given in Table 4-6. Operational requirements include operating, standby, and peak power usage and thermal requirements imposed on the station's thermal control system. A summary of the operation requirements is given in Table 4-7. Finally, a list of equipment and a breakdown of weight, volume, and power is given in Table 4-8.

The launch volume involving the truss pieces and heliostat frame is based on an estimation of their packaged girth and does not take into account the unused internal space. More efficient packaging would surely be possible. As is, the launch volume and launch weight are small enough to be included in one shuttle launch. Operational volume is defined by a surface totally enclosing the experiment and is considerably more than the sum of the parts it surrounds.

The power required by the system includes power needed for the data acquisition and control system and for the heliostat tracking motors. Peak power usage will be short in duration and is defined by the maximum power requirement for the data system and startup surge needed by the tracking motors to bring the heliostat mirror quickly up to rotational speeds.

4.2.10 Safety

There are two areas of safety that must be addressed in the solar concentrator experiment. Those areas are, 1) the handling of large structures and how they might affect crew and Space Station safety and 2) the safety and interference considerations associated with collecting, redirecting, and concentrating solar energy near the Space Station. There

Table 4-6. Physical Specifications of Experiment

Packaged Volume:	250 cubic meters
Packaged Dimensions:	16 m by 4.5 m diameter
Launch Weight:	5,700 kg
Operational Volume:	56,000 cubic meters
Operational Dimensions	45 m long by 40 m diameter
Maximum Acceleration During Operation:	0.075 g's with no pointing loss

Table 4-7. Operational Requirements

POWER		
	Power Usage (ave):	0.2 kW DC
	Peak Power:	0.25 kW
	Operating Power Period:	24 hours
	Voltage:	12 volts DC
THERMAL (Active Cooling)		
	Temperature Range:	0 to 40^{o}C
	Heat Rejection:	0 to 1 kW

Table 4-8. Equipment List for the Solar Concentrator Experiment TDMX 2111

EQUIPMENT	WEIGHT (kg)	PACKAGED VOLUME (m^3)	PEAK POWER (watts)
Data Acquisition and Control System	25	0.2	200
Parabolic Solar Concentrating Mirror	1100	137	--
Parabolic Mirror Mount	50	6.8	--
Heliostat Mirror Frame	100	13.6	--
140 Individual Heliostat Mirror Panels	3900	56.	--
Heliostat Mirror Mount	150	8.0	--
Heliostat Tracking Drives	5	0.1	40
Solar Concentrator Truss Base	218	18.7	--
Truss Extension Arm	97	8.3	--
Planar Partial-Reflecting Mirror	5	0.1	--
Photodetector Array Modules	20	0.7	--
Diagnostic Laser	5	0.05	5
Video Camera	5	0.05	5
Sensor Package	5	0.05	--
Signal Cables	5	0.05	--
Coolant Lines	10	0.3	--
TOTALS	5700	250	250

is little experience gathered at this point in large material handling in zero gravity except from past shuttle operations and neutral-buoyancy test tank experiments. However, considerable experience in large space structures will be obtained from building the Space Station. This knowledge can be immediately applied towards the safe construction of the solar concentrator experiment.

The geometry of the solar concentrator experiment defines a hazardous zone in which solar concentrations greater than 1 are found and could be dangerous to the crew and materials during EVAs. This hazardous zone has a cylindrical shape that runs the length of the base and entirely encompasses both mirrors in any tracking position. Outside this zone, the reflectance of one or less solar constants in any direction is possible and can be controlled within the limitations of the tracking system. The shutdown or failsafe alignment of the experiment involves rapidly pointing and holding the heliostat mirror in the sunrise position (i.e., pointing away from the Space Station) which would restrict any reflection of light beyond the vertical plane of the experiment. Finally, the heliostat positioning mirrors can be adjusted to give a maximum concentration of 28 solar constants which could be focused in a direction that could affect Space Station operations. Should any problems arise during tracking that might cause focusing by the heliostat outside the hazardous zone, then the mirror panels could quickly be returned to the planar position which would reduce the solar concentration to one.

5.0 TDMX 2121 - TEST SOLAR PUMPED LASERS

CONTACT: Dr. Edmund J. Conway, MS 160
NASA LaRC (804) 865-3781

5.1 GOALS AND OBJECTIVES

This mission is the second in a coordinated four experiment sequence. Its objective is the evaluation of candidate solar pumped lasers for operation in the Space Station environment. It is this environment where these lasers have their most promising application - the transmission of power through space - typically from a large central platform with a large solar concentrator to smaller, more mobile platforms. Prior to this mission, all development and testing of solar pumped lasers has been done in the laboratory environment using solar simulators to produce light simulating the intensity and spectral characteristics of concentrated solar radiation. The goal of this experiment is twofold: (1) the assessment of efficiency, stability, and operating characteristics of candidate laser systems, and (2) the optimization of operating characteristics to achieve maximum efficiency and lifetime of state-of-the-art, solar pumped lasers.

5.2 METHODOLOGY

5.2.1 Description

Solar pumped laser systems would be evaluated in this experiment by placing them near the focal point of the solar concentrator installed on the Space Station under TDMX 2111 and by analyzing the characteristics of the laser beams produced by them. The laser systems will be installed one at a time on a support along the axis between the heliostat collecting mirror and the parabolic concentrating mirror. The resulting laser beam will be routed inside the truss-like base to the far end of the experiment where the far-field beam properties would then be studied. A drawing of the physical layout of the CO_2 laser system and its connection to the solar concentrator experiment is given in Figure 5-1. Other candidate laser systems to be tested are the Iodine direct-pumped photodissociation laser and a Neodymium/liquid laser.

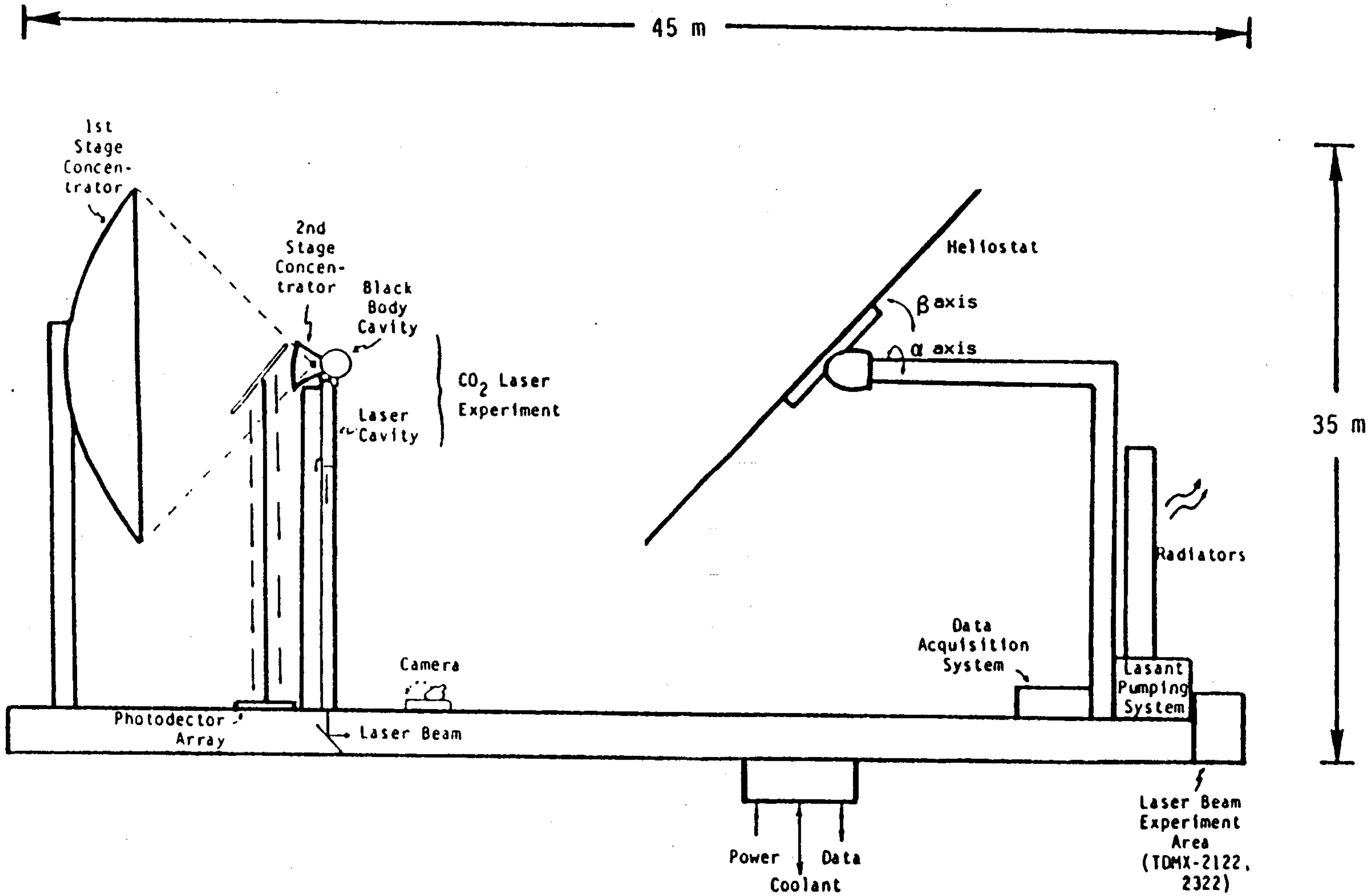


Figure 5-1. The CO_2 Laser System As Seen Integrated With The Solar Concetrator Experiment.

The daily operational period of the lasers is determined by the solar concentrator system which is continuously tracking the sun, but due to earth shading, would be operational a minimum of one half each orbit or at least 12 hours per day. The solar concentrator experiment is operating throughout the year, and at certain times of the year the daily coverage will be greater than 16 hours. During these operational periods, a constant amount of solar energy is present at the focal point except when partially shaded by the Space Station. This partial shading would always be restricted to the second quarter orbit from sunrise. The laser system could, therefore, be operated during any orbit or time of year with peak operating periods twice each year. Also, since the instantaneous solar radiation available to the laser system is relatively constant, laser operation at maximum power would be relatively independent of the time of year.

5.2.2 Design Considerations

The general laser system is divided into three physical blocks which are the laser block, the lasant handling block, and the cooling block. The laser block would include the laser cavity, a second stage concentrator or beam shaper, and depending on the particular system, a blackbody cavity and associated plumbing. These items would be located near the focal point for direct conversion of solar power to laser power (see Figure 5-1). The two remaining blocks would both be located behind the heliostat mount with plumbing to and from the laser block being routed through the trusslike solar concentrator base. Their location was selected so that the cooling radiators would be continuously shaded from the sun by the heliostat mirror.

Figures 5-2 and 5-3 are block diagrams of the two most likely candidate laser systems; the CO/CO_2 laser and the Iodine gas laser. The three physical blocks are defined for each laser type. Common elements such as plumbing and cooling would be adapted from one system to the next to reduce cost and setup time. Other types of laser systems that might be chosen would in all likelihood share the same features as these two. Not shown in the diagrams is the availability of the Space Station thermal

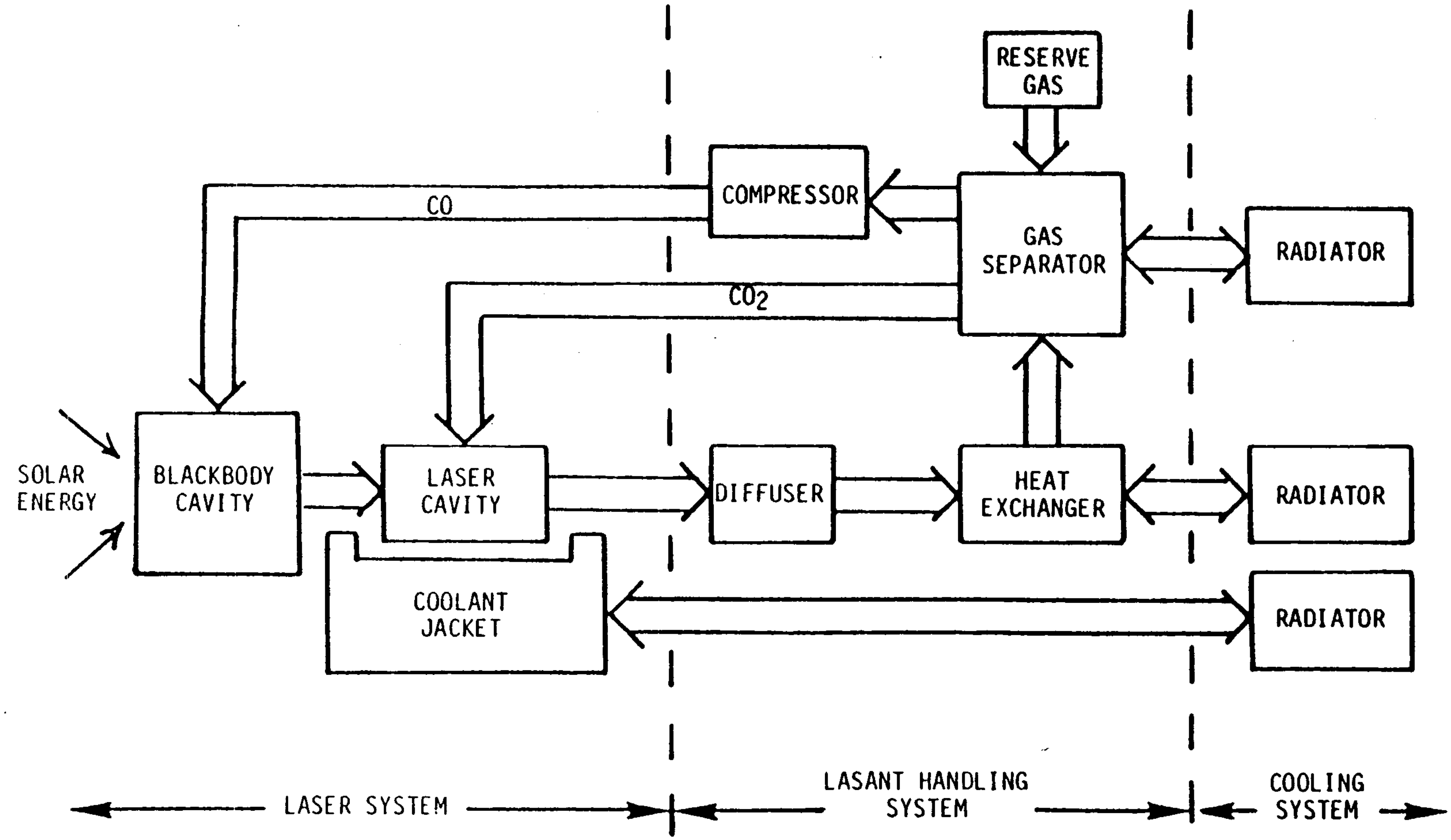


Figure 5-2. Block Diagram of Solar Pumped CO_2 Laser.

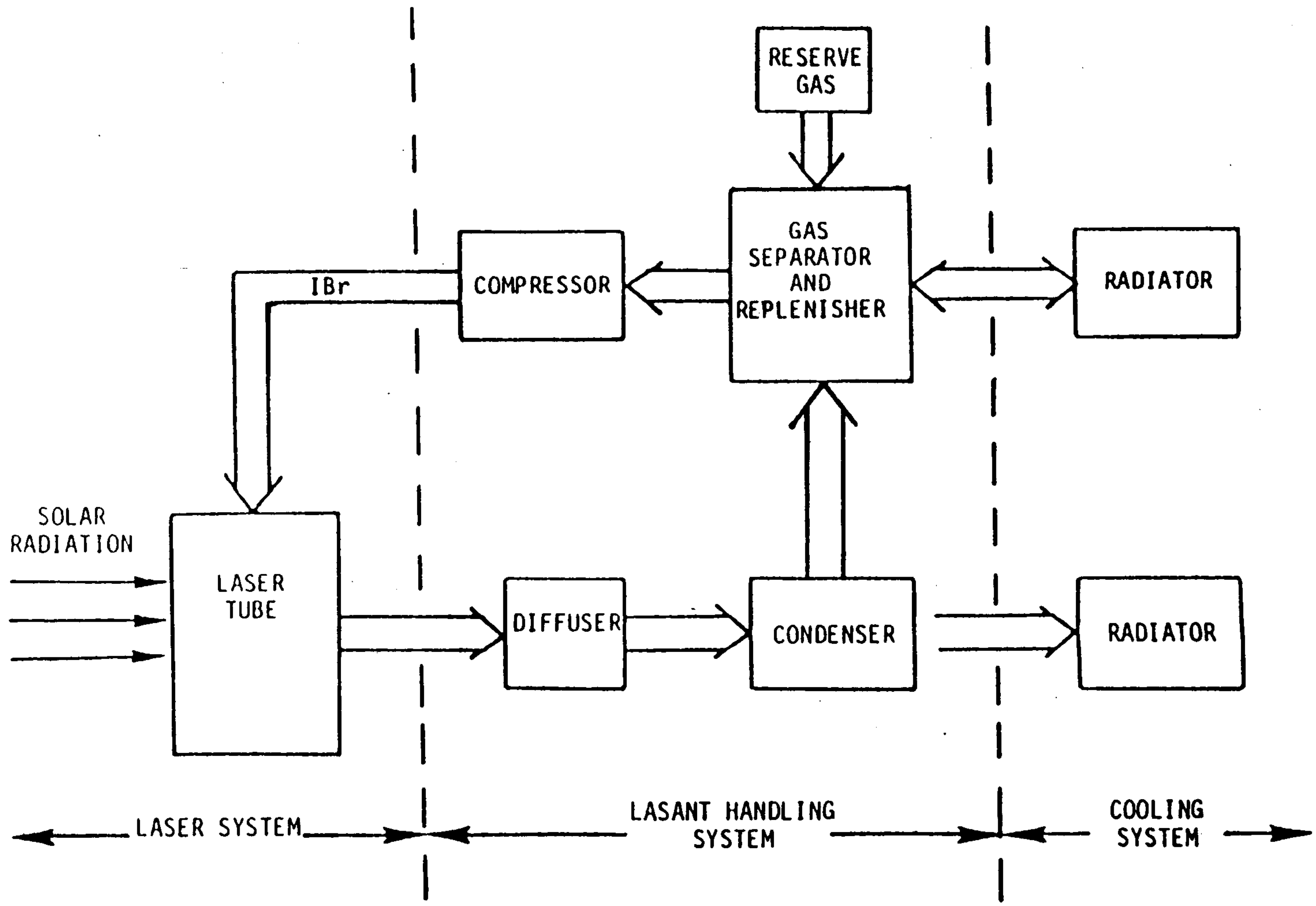


Figure 5-3. Block Diagram of Solar Pumped Iodine Laser System.

system to payload experiments. It is suggested that this system might be used effectively in handling part of the thermal burden, but that a maximum usage must be set and additional cooling would be done by the experiment's cooling equipment.

The goal of the CO_2 system was to reach a peak power output of at least 1.0 kW. Since this peak power is important to related experiments, the duration of the peak should be significant. At present, a laser system of this size represents the most currently available technology. For instance, the blackbody oven might reach a temperature of 2000 degrees Kelvin, and must reradiate the same amount at the incoming energy (estimated at 328 kW. Materials must handle these extremes and the thermal shock of turning the system on and off and yet be practical for space applications. Also the interior of the blackbody oven should be composed of a material that is very reflective to 4 micrometer wavelengths (the desired CO absorbing band) because the incoming radiation has a larger percentage of radiation at this wavelength than the emissions from the hot oven.

The second stage concentrator used in the CO_2 laser system helps remove pointing errors and provide additional concentration into the blackbody oven. Based on concentration limits, it is thought that the typical aperture into the blackbody chamber would be 40 centimeters and that the chamber would have approximately a one meter diameter. It is estimated that 4 kW of power will be absorbed by the CO gas passing through the blackbody chamber and carried to the laser cavity. About fifty percent of this energy is translated to the walls of the laser cavity and must be removed by surrounding the laser with a coolant jacket. Twenty-five percent of the incoming energy is translated to laser energy while the final twenty-five percent remains in the gases and must be rejected by the cooling system. The overall system efficiency is expected to be about 0.3%.

The Iodine laser has much the same design considerations as the CO_2 laser system. The overall system efficiency will be less than the CO_2 laser and will result in a laser power of somewhat less than 1.0 kW. This lower efficiency is due in part to less energy at the desired absorbing spectrum which is in the ultraviolet region and losses in the reflecting surfaces at those wavelengths. The incoming radiation is shaped by the second stage concentrator into a shape appropriate for the laser rodlike cavity. No blackbody chamber is needed since absorption is done directly in the laser cavity. Energy not absorbed by the laser is transmitted out the laser cavity and back into space. If the laser cavity has a coolant jacket, then the coolant material must be transparent to the desired wavelength. The cooling system must be able to handle any additional heating due to the coolant material absorbing solar energy. The Iodine laser gases are expected to reach much higher temperatures (possibly 500 degress Kelvin) than the CO_2 laser and thus will require a higher capacity heat transfer system to cool the gases to approximately 250 degrees Kelvin. Finally, a certain amount of waste gases will be formed during lasing, in particular Iodine molecules, that tend to reduce the efficiency of the laser. To remedy this effect, these waste gases will be condensed and separated.

5.2.3 Sensors

The objective of this mission is to study and characterize various candidate laser systems. In particular, we want to measure various efficiencies and lasant handling capabilities must be evaluated at different points in the system. To do this, several sensor types will be incorporated into the experiment. The following is a list of the sensors that will most likely be used.

: Flow meters
: Pressure sensors
: Thermocouples
: Electrical power meters
: Optical power meter
: Luminance sensors
: Gas analyzers

Certain elements in the laser system are critical for continuous high power operation. For instance, gas flow rates in the Iodine laser are critical in maintaining the desired gas temperature and pressure in the laser cavity. Potentially damaging temperatures can be reached if this system does not recognize and respond quickly. Therefore, certain sensors such as flow rates and temperature must be monitored at a high data rate. Also, the optical properties of the laser beam during operation would also need to be sampled at high rates to help characterize the entire system performance. During peak testing, these data rates could be as high as hundreds of samples per second. Of course, the response time of any particular sensor determines its maximum practical data rate and for sensors such as thermocouples sample rates may be considerably less.

5.2.4 Data Acquisition and Control

The data acquisition functions used to sample and control the laser system will be performed by the stand-alone data system used in the solar concentrator experiment. This system utilizes a sensor feedback loop which allows real-time analysis of data for control applications. This is of particular importance to the laser systems where quick response to changing parameters is essential. Software modules will be incorporated in the data system that will perform the desired control and monitoring functions. For a more general discussion on the data acquisition system, refer to Chapter 4.

A diagram of the signal flow for the laser experiment is given in Figure 5-4. The various sensors are interfaced to the data acquisition system where their signals are conditioned and digitized. Data is then analyzed and stored in onboard memory. Control decisions would be made on this data and instructions sent to the control block. System functions such as flow rates would then be adjusted by the control electronics. Finally, communications between the Space Station operators and the laser experiment are handled by the Subsystem Data Processor (SDP). This network would allow periodic data dumps and provide real-time control to the operators if desired.

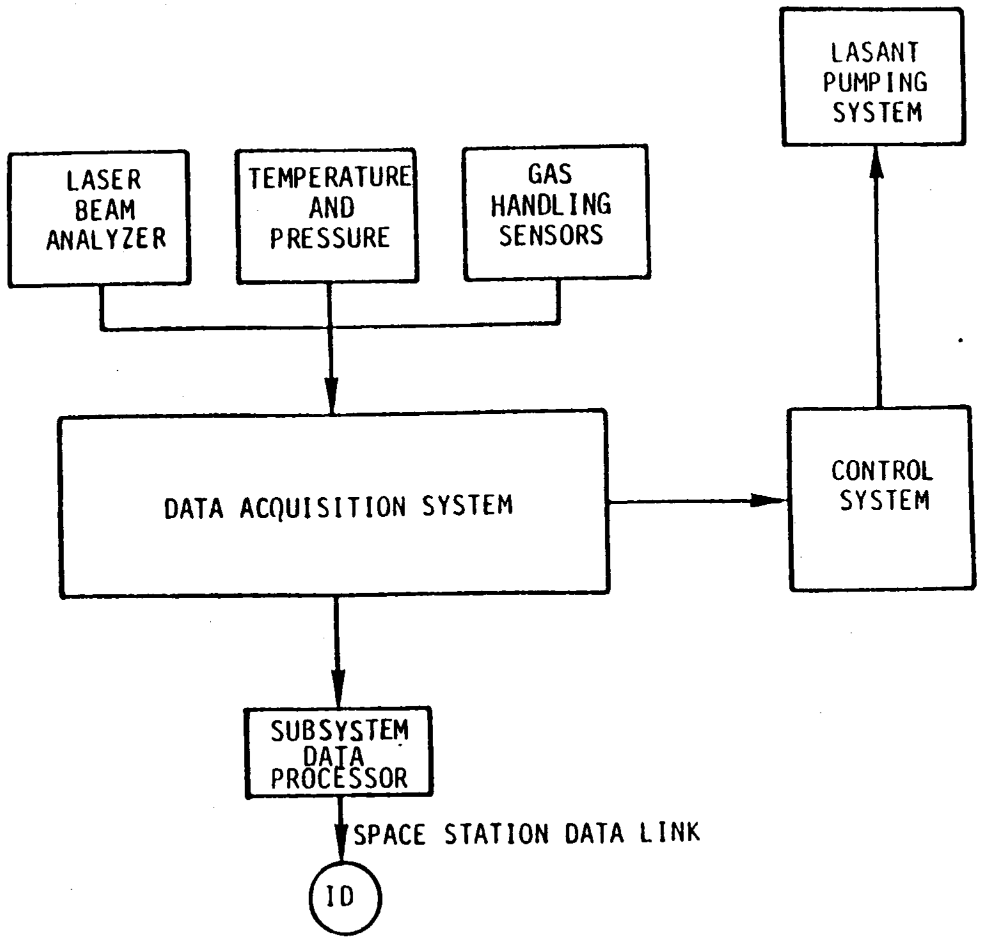


Figure 5-4. Signal Flow Diagram of the Solar Pumped Laser Experiment.

5.2.5 Data Communications and Handling

Data communications are handled through the ODDNET which allows two way communications between the payload and most of the Space Station resources. Periodic dumps of sensor data, status, and control decisions will be passed over this network. Processing resources will be needed at the Space Station to aid in the storage and analysis of this data. If problems are recognized at this level, then Space Station operators might be notified allowing them to interact with the experiments and correct problems that the data acquisition system could not handle. In general though, this data would be stored temporarily on disk and transmitted to the ground periodically as desired. This temporary disk storage is expected to be no more than 20M bits if this amount of data is transmitted at least once each orbit. An additional 1/2M bits of permanent disk space will be needed for program storage and 1 to 2M bits of random access memory will be used during operation. For additional information on data communications refer to Chapter 4.

5.2.6 Telemetry and Ground Operations

Both station to ground and ground to station communications will be used during this experiment. Experimenters at the ground site will require fast turn around of incoming data so that they may communicate system modifications to the Space Station operators. Some real-time telemetry will be needed during the critical phases of the experiments.

Digital data, voice, and perhaps video information will be handled by the telemetry system for this experiment. The digital data would be sent at about 10 kbps either continuously for real-time applications or in periodic bursts at higher data rates during each orbit. Voice communications would be restricted to short periods of real-time control applications. Video data could provide ground experimenters with visual observations of the laser system during setup.

The data received by the ground station will be processed, reformatted, and stored on magnetic tape for distribution to NASA experimenters. Limited analysis of the data will be done by the ground base facilities and would include unit conversion of the data and visual presentation of the data on screen or through printouts. Onsite experimenters would also make use of the telemetry network to send data and/or voice to the Space Station operators when necessary.

5.2.7 Crew Activities

Setup and operation of the solar pumped laser systems represent configuration modifications to the solar concentrating experiment as described in Chapter 4. Each candidate laser and support instrumentation will be installed and tested and after successful completion, it will be replaced by the next laser type. Personnel skilled in spacecraft systems at the technician level or higher would be needed to perform the many EVA functions.

The laser systems are defined by three physical blocks as discussed in Section 5.2.2 on design considerations. These three blocks or modules would be assembled and tested on the ground and launched as individual packages. Once in place, these modules would be deployed requiring only a minimum of crew time to complete setup. Each block would be installed in its proper location such as the laser cavity being placed near the solar concentrator focal point. The most extensive EVA time will be routing and connecting the plumbing and signal lines that go between these modules. The laser beam pointing mirrors and beam analyzer sensors would also require much crew time due to the delicacy of their placement and alignment. Finally, data and control lines would need to be connected to the data acquisition and control system. It is estimated that two crew members could perform the necessary EVA setup functions in one to two days using about 16 manhours.

During normal operation of each laser experiment, crew activity would be limited to periodic inspection of the instrumentation and performance of those servicing functions needed for the successful operation of the experiment. During non-operating periods, the crew would perform functions such as realignment of the laser or blackbody cavity or adjusting valves in the lasant plumbing system. Modifications in control of the lasant materials such as changing gas flow rates would be made by a Space Station operator through communications with the data acquisition system.

After completion of each particular laser experiment, crew activities would involve disassembly of those parts of the laser system that cannot be shared by the next experiment. These parts would be stowed and returned on a future space shuttle so that they may be studied. After completion of the final laser experiment, other components such as the laser beam pointing mirror might also be returned for inspection.

There are two other interrelated experiments requiring the laser beam produced in this experiment. They are, the power conversion experiment TDMX 2122 and the laser propulsion experiment TDMX 2322. Crew activities for these experiments will be intermixed with the operating activities of the laser experiment. Modifications in the laser system may be needed for the successful operation of these experiments. Advanced planning would keep these modifications to a minimum.

5.2.8 Design Specifications

This section gives a brief discussion of launch, setup, and operating specifications for solar pumped laser experiments. Launch and setup specifications include weight and volume both before and after assembly. Operating specifications include resources needed from the Space Station in the form of electrical power and maximum thermal loads.

It would be practical to launch all laser systems in one shuttle flight since launch weight and volume allow it. However, it might be possible that some candidate laser systems may not be launched until design problems are solved in tests with the first laser systems. The weights and packaging volumes given are best estimates of the total package regardless of the number of launches required. Setup volume includes the largest volume any one experiment would require. The setup dimensions include the laser cavity, the lasant handling and cooling system, and the plumbing and laser beam routing volume. The launch and setup specifications are given in Table 5-1.

Table 5-1. Launch and Setup Specifications

Launch volume:	18.0 cubic meters
Launch dimensions:	3 x 3 x 2 meters
Launch weight:	800 kilograms
Setup volume:	38.0 cubic meters (overall envelope: 1836 cubic meters)
Setup dimensions:	17x1x1 + 4x3x1 + 36x0.5x0.5 meters
Acceleration max:	0.1 g (defined by solar concentrator limits)

The lasant handling system for each laser experiment will require a certain amount of power to operate pumps and valve actuators as well as additional power for growth accommodation in the data acquisition and control system. These power requirements would be somewhat less during the non-operation periods during each orbit. The heat rejection requirements of the system would only be handled in part by the Space Station's thermal control system. This is referred to as the active thermal requirements of the experiment and would probably include cooling of the instrumentation and the electronics and not the gas cooling. A stand-alone heat exchanger and radiator system would handle a large majority of the cooling loads and is referred to as the passive thermal requirements. The operating specifications are given in Table 5-2. A list of equipment and corresponding weight, volume, and power is given in Table 5-3.

5.2.9 Safety

Operating a high powered laser system in such close proximity to the Space Station requires that the effects on crew and structural safety be addressed. When EVA functions are being performed, particularly those involving servicing and observation of the laser experiment when it is operating, the possibility of interferences or contact with the laser beam or heated surfaces and the hazards involved in inadvertantly reflecting the beam must be recognized and prevented.

In practice, the laser beam energy will be contained and dissipated after being analyzed by experiments mounted in the path of the beam on the solar collector structure. Power densities of the order of 1.0 kW/cm^2

Table 5-2. Operating Specifications

Power		
Operating power:	0.1 kilowatts (dc)	
Operating Period:	14 hours/day typical	17 hours/day max.
Peak power:	0.15 kilowatts	
Peak period:	less than 1 hour/day	
Standby power:	0.05 kilowatts	
Thermal (active)		
Operating temperatures:	0.0 to 40.0 degrees C	
Heat rejection:	4.0 kilowatts max operating and non-operating	
Thermal (passive)		
Operating temperatures:	20 to 100 degrees C	
Heat rejection:	20 kilowatts	

Table 5-3. Equipment List for the Solar Pumped Laser Experiment TDMX 2121

EQUIPMENT	WEIGHT (kg)	PACKAGED VOLUME (m^3)	PEAK POWER (watts)
Blackbody Chamber	150	1.5	--
Second Stage Concentrator/ Beam Shaper	50	3.0	--
Laser Tubes	100	3.0	--
Lasant Gas Plumbing	50	1.5	--
Coolant Fluid Plumbing	50	1.0	--
Lasant Pumping System	50	1.5	50
Gas Separating System	100	2.5	50
Gas Cylinders	50	0.6	--
Heat Exchanger and Radiators	150	3.0	25
Laser Beam Pointing Mirrors	5	0.05	--
Laser Beam Analyzer Pad	25	0.1	--
Sensor Package	5	0.1	--
Signal Cables	5	0.1	--
Data Acquisition and Control	10	0.05	25
TOTALS	800	18	150

could be produced and temperatures in the range of 1000 degrees Kelvin on the surfaces absorbing this power might be found. The orientation of the solar concentrator system and the laser system prevents a direct beaming of laser light towards the Space Station should the beam not be contained. However, obstacles in the long range line-of-sight would be affected.

It is probably best that crew activites be restricted to non-operating periods of the laser experiment, particularly when contact with the solar concentrator structure is required. In view of the safety hazards imposed by the solar concentrator experiment and those described by the laser experiment, remote monitoring of the experiment during operations should be the rule-of-thumb. Finally, it is practical to assume a small portion of the laser beam would be reflected in random directions due to imperfections in mirror and absorbing surfaces. Though perhaps not a safety hazard, these reflections might cause interference with other sensitive experiments and represents an incompatibility between these experiment types.

6.0 TDMX 2122 - LASER-TO-ELECTRIC ENERGY CONVERSION

CONTACT: Dr. Edmund J. Conway, MS 160
NASA LaRC (804) 865-3781

6.1 GOALS AND OBJECTIVES

The solar concentrator described in TDMX 2111 and the solar pumped laser described in TDMX 2121 form the transmitting portion of a system for transmitting power through space using a laser beam. The purpose of this TDM is to evaluate candidate devices which may serve as efficient receivers of that energy and thereby assess the feasibility of transmitting power through space using laser beams. Data from this experiment can be used to determine operational characteristics of energy conversion devices to allow analytical modeling and the subsequent design of energy transmission systems.

6.2 METHODOLOGY

6.2.1 Description

Candidate laser-to-electric energy converters will be evaluated by installing them, one at a time, in the beam of the solar pumped laser and monitoring the electrical output of the device. Each device will be mounted on the solar concentrator experiment base (TDMX 2111) and would replace the sensors used to measure the far-field laser characteristics in TDMX 2121 (see Figure 5-1). The primary candidate for the energy conversion experiment is the photovoltaic cell for which various semiconductor materials and etching technologies will be investigated. Other potentially useful devices are being developed and may be tested as well.

As discussed in Chapter 5, the laser beam system could be operational as much as one half each orbit (46 minutes) or 12 hours each day throughout the year. These solar pumped lasers would normally emit a continuous laser beam during this period. Therefore, experiments which use the laser beam, such as the energy converter, would need to be functional during the long operating periods of the laser system. Otherwise, some means of diverting or shuttering the beam would be needed. Most likely in the energy conversion experiments the effects of continuous long-duration laser radiation on power output and material degradation would be an area of importance and not a problem to be avoided.

6.2.2 Design Considerations

There are several photovoltaic designs being developed right now that would prove practical for high energy (greater than 10 watts/sq cm) laser-to-energy conversion. The current research is considering the effects on cost, efficiency and lifetime of a photovoltaic cell based on the choice of semiconductor materials, fabrication procedures and the frequencies of the irradiant beams. It has been found that to achieve satisfactory efficiencies from cells subjected to laser light, the selected materials must be carefully matched to the laser type. In addition, certain etching technologies provide higher efficiencies than others at large energy densities such as those that would be encountered in this experiment.

There are several common materials used to make photovoltaic cells; silicon, germanium and gallium arsenic to name a few. Each semiconductor has a bandgap energy associated with it which the energy of the incoming photons must exceed in order for current to flow. In monochromatic laser light most all the photons contain the same energy which is a function of laser frequency. Selecting a semiconducting material whose bandgap energy is less than this energy would allow laser-to-electric power conversion to occur. To achieve the highest efficiencies from a particular laser, a material should be selected whose bandgap energy is closest to but less than the photon energy. This is true because the higher the bandgap energy of a material the lower the dark current density would be, resulting in a higher open collector voltage. Similarly, the absolute temperature of the material during conversion also determines its efficiency since lowering the temperature decreases the dark current density and results in increased cell output.

The three laser systems that will be used in TDMX 2121 are the CO_2 laser, the Iodine laser and the Neodymium liquid laser. The CO_2 laser emits light at a wavelength of 10.6 micrometers (or equivalently, photons at an energy level of 0.18 electron volts). At present, there is no practical semiconductor or semimetal material that will directly convert these low energy photons into electrical current. For the CO_2 laser to participate in the energy conversion experiment, then some new type for photovoltaic material must be developed or some other type of energy conversion system must be used.

The Iodine family of lasers emits light at similar wavelengths in the near infrared range. For the C_3F_7I laser, germanium would be a good choice for the photovoltaic cell material. The IBr laser has lower photon energies than the bandgap energy of germanium, therefore some other material such as Indium Arsenic (InAs) would be needed. The Neodymium/YAG laser has the most promise for energy conversion because the emitted photon energy is the highest of the three. Silicon is a very good match for this laser type. Germanium also is good but has a lower efficiency.

Several manufacturing methods are presently used to build photovoltaic cells. The most common types are the Schottky barrier converter, the conventional P-N junction converter, the planar multijunction converters and the vertical P-N single and multijunction converters. The most promising of these technologies is the vertical multijunction converters (VMJ). Due to the geometry of this cell type, the series resistance in each junction is minimal and the cell can tolerate very high power densities. Several methods of etching the VMJ cells and connecting junctions in series and parallel are being studied at this time, particularly in silicon. These studies will need to be directed towards germanium and perhaps InAs in order to fully test the VMJ technology in this laser-to-electric power conversion experiment.

Figure 6-1 is a very general drawing of the laser power conversion system. The photovoltaic array would be contained in a square plate with the effective conversion area larger than the expected laser spot size (perhaps 5 cm in diameter) so that pointing errors would not be a factor. The conversion efficiency increases with decreasing absolute temperature of the array, therefore the effects on heating from absorption and the equilibrium temperature that the array obtains would need to be studied. Using the back and outer edges of the plate as a blackbody radiator would be one means of passively removing heat so a low absolute temperature can be obtained. Finally, the photovoltaic cells should absorb as much of the laser beam as possible to reduce beam scattering which might disturb other Space Station experiments. A shield might be needed to keep any reflections to a minimum.

6.2.3 Sensors

There are three basic parameters that will be monitored during the energy conversion experiment. They are the magnitude of the incident laser beam power, the temperatures of the photovoltaic array and the power/voltage output of the device. Various sensor types will be used to monitor these parameters and would include photodetectors, thermocouples and a simple electrical circuit to measure the voltage and current generated by the cell.

Laser beam power would be monitored by placing a small mirror in the beams path to direct a small sample of the incident energy onto a photodetector. Similarly, the back side of this mirror would direct a sample of any backscattered laser light onto a second photodetector to help monitor the reflection properties of the surface of the photovoltaic array (see Figure 6-1). These two detectors would be tuned to the wavelength band corresponding to the laser type.

Temperatures in the device would be monitored by embedding thermocouples in the substrate. Their outputs would be representative of the absolute temperature of the photovoltaic array and would help

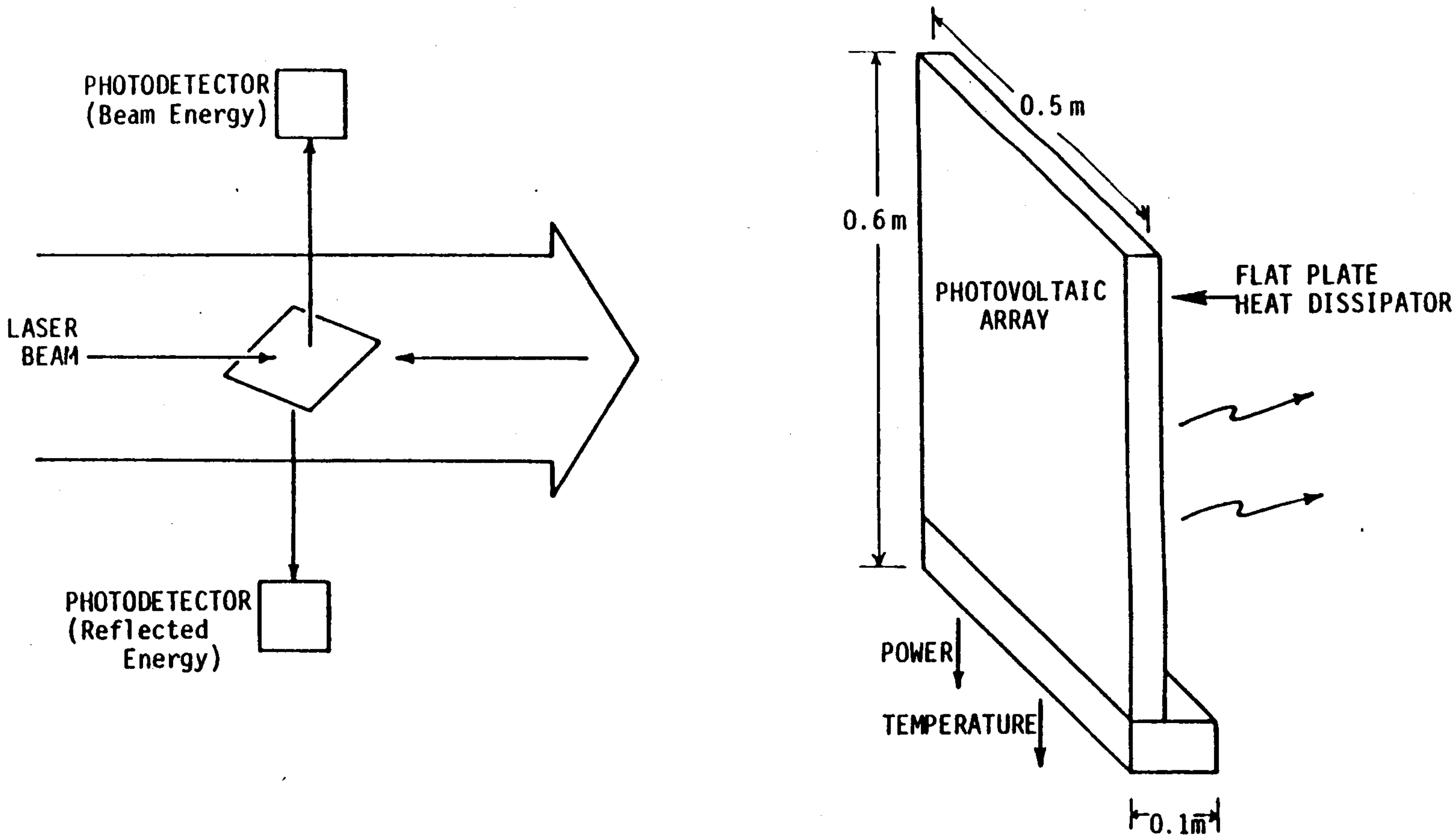


Figure 6-1. Layout of the Photovoltaic Laser-to-Energy Conversion Experiment.

characterize the heat transfer mechanism which is important in defining the overall efficiency of the converter.

The individual junctions in the photovoltaic array would be arranged in series and parallel to provide particular voltage and current levels. These voltages and currents would be monitored to verify cell performance and to give a better understanding of the effects of series-connected verses parallel-connected junctions. The overall power of the cell would be monitored to determine the efficiency of laser-to-electric power conversion.

6.2.4 Data Management

Data acquisition for the laser converter experiment will be done by the stand alone data system which is part of the solar concentrator experiment TDMX 2111 discussed in Chapter 4. The sensors will be interfaced to the signal conditioning electronics which are a part of this system. The desired parameters will be monitored periodically during the operational period and stored in short-term memory in the data system. Figure 6-2 is a diagram of the signal flow in the power conversion experiment.

The photovoltaic device is passive and no control over the experiment will be required by either the data system or Space Station operators. Data will simply be gathered and transmitted over the ODDNET to the Space Station. No additional processing will be required by the Space Station computing resources except for some reformatting and storage of data before transmission to the ground. Storage resources would be expected to be less than 1 Mbit depending on the frequency of telemetry.

Only station to ground telemetry will be required for the operation of this experiment. Some limited voice data might be sent by Space Station operators to confirm setup. Continuous transmission requirements would be small, probably less than 1 kbps. It might be more practical, however, to store a block of data and transmit that block in a short burst each orbit. Once on the ground this data would be reformatted and recorded on magnetic

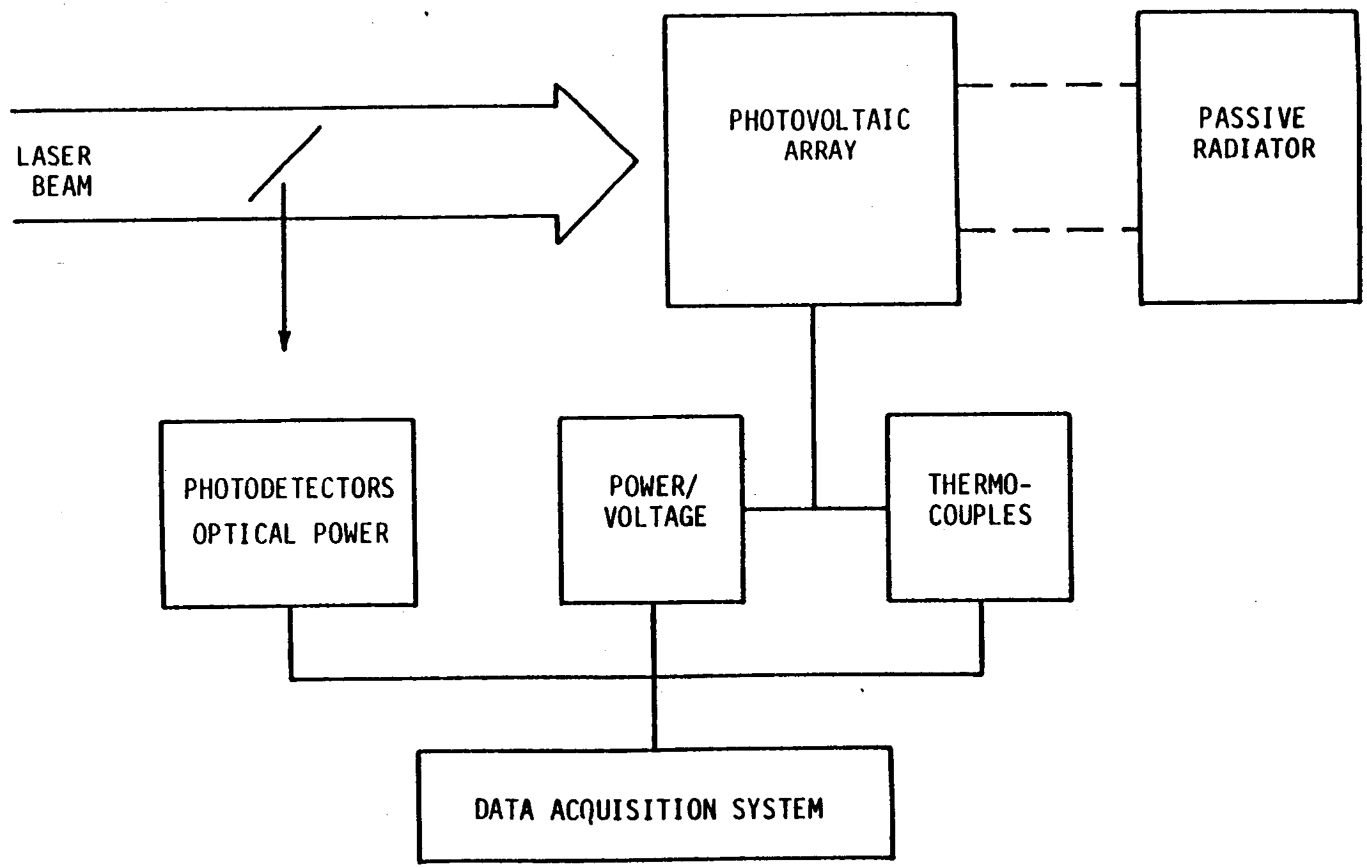


Figure 6-2. Signal Flow Diagram for Photovoltaic Energy Conversion Experiment.

tape for distribution to the experimenters. No real-time turn around of the data is required beyond those necessary for the related solar concentrator and laser experiments.

6.2.5 Crew Activities

Setup and operation of the power conversion experiments should require only minimal crew EVA time. The only activities involved in setup would be to unpack the experiment, mount it onto the end of the trusslike base of the solar concentrator and connect the signal lines to the data acquisition system. Each photovoltaic array type would be tested and replaced by the next type. This installation could be handled by one technician with probably no more than 2 manhours per experiment.

Once operational each experiment might require periodic crew activity to perform servicing of the experiment such as sensor realignment and inspection of the device's surface to look for visual damage. The EVA time estimated for servicing is about 1 manhour. After completion of the power conversion experiment, a crew member will be responsible for stowing all device types for return by the space shuttle to ground investigators.

6.2.6 Design Specifications

All the power conversion devices could be packaged and launched on one shuttle flight. The total launch volume would be on the order of 1 cubic meter and would have an estimated mass of 20 kilograms. The operational volume would also be about one cubic meter. Since the conversion devices will be studied after completion of the experiment, they represent returnables of similar mass and volume for a future shuttle flight. Table 6-1 is a list of equipment and a breakdown of weights and volume.

The photovoltaic system is a passive system and would require no external power or cooling. The specification for power usage in the data collection for this experiment is contained in the data acquisition

Table 6-1. Equipment List for the Laser-To-Electric Energy Conversion Experiment TDMX 2122.

EQUIPMENT	WEIGHT (kg)	PACKAGED VOLUME (m^3)
Photovoltaic modules	15	.95
Laser Beam Partial-Reflecting Mirror	2	0.02
Photodetectors	1	0.01
Signal Cables	2	0.02
TOTALS	20	1.0

system's power specifications as given in Chapter 4. No additional cooling is required since, by design, the device acts as its own blackbody radiator.

6.2.7 Safety

Crew safety during operation of the laser power converter has been partially addressed in the laser experiment in Chapter 5. As discussed there, the hazards of beaming a high intensity laser beam in proximity of the Space Station must be studied. This experiment has few additional safety considerations except that the experiment would, hopefully, be setup and inspected by the Space Station crew during non-operational periods of the laser. Finally, there will be a certain amount of randomly reflected laser light that might not be contained by the conversion apparatus. This stray light could interfere with other sensitive experiments on the Space Station and make their simultaneous operation incompatible. This area should be addressed before a firm scheduling of experiments is attempted.

7.0 TDMX 2261 - SENSOR SYSTEMS TECHNOLOGY

CONTACT: Richard R. Nelms/Arthur L. Newcomb, Jr., MS 486/474
NASA/LRC (804) 865-4947/3761

7.1 OBJECTIVE

The objective of this experiment is to develop and demonstrate a manned laboratory capability on Space Station for the calibration, characterization, and modification of electro-optical instrumentation for remote measurements of the solar system, with emphasis on measurements of the earth and its environment. The availability of a manned laboratory in space will allow experimental optimization of sensor technology resulting in improved accuracy of remote measurements of the Earth and its' atmosphere. Such optimization of sensors planned for the Space Station platforms will allow "final" design refinements and modifications of instrumentation modules to be done on the ground prior to shipment to the station for use on the platforms. Future evolution of this laboratory technology can provide a service and calibration facility in support of the station orbital platforms presently planned for operational measurements of the Earth's environment, e.g., EOS. Such a facility would also allow free-flyer platform instrumentation to be repaired, or even upgraded, without return to Earth thus enhancing the usefulness and lifetime of such platforms.

7.2 DESCRIPTION

The Sensor Systems Technology Laboratory will be designed for experimental research on passive remote sensors which view several targets, including Earth-nadir, Earth Limb, the Solar Disc, and cold space. For the present Dual Keel Space Station configuration, such a range of view angles cannot be obtained for sensors directly attached to the modules which comprise the Manned Core. A concept for the laboratory could thus consist of a Sensor Pointing Platform (SPP) mounted on the Lower Keel or Lower Boom as shown on Figure 7-1, which carries the Electro-Optical Sensing Head, attitude determination and pointing control sensors, and specialized sensor equipment which must be closely coupled to the sensing head (e.g., pre-amplifiers, detector cryogenics, bias power supplies, etc.). The sensing head and pointing platform are remotely controlled from the Sensor Systems Technology Laboratory (SSTL) located in one of the manned laboratory modules mounted on the Space Station Transverse Boom using the station SISS Core Network and Payload Accomodation System. The SSTML will contain the electronics systems for sensor system control and data processing and for control of the SPP. It will also serve as the control and data interface between the experiment and the Space Station. The SSTML will also be equipped for a number of optical assembly, test, alignment, characterization, and calibration tasks, and will include calibration reference and primary standards for optical wavelengths. It is planned that research sensors can be assembled and tested in the SSTL, then moved to the SPP using EVA for final checkout and alignment prior to flight testing.

7.3 ENVIRONMENT

7.3.1 Platform and Orbit

The Sensor Systems Technology Laboratory will be a part of the growth Space Station Manned Core. The 500 km altitude and 28.5 degree inclination orbit is suitable for the sensor technology tasks. The orbit should be nearly circular, and orbital ephemeris data will be required for data interpretation for a number of sensor types. The ephemeris accuracy requirements vary widely depending on the research being performed. Typical values for most sensors will be data on orbital altitude to 1.5 km, 1σ, on orbital position to 10 km, 1σ and on orbital velocity to 100 m/sec 1σ.

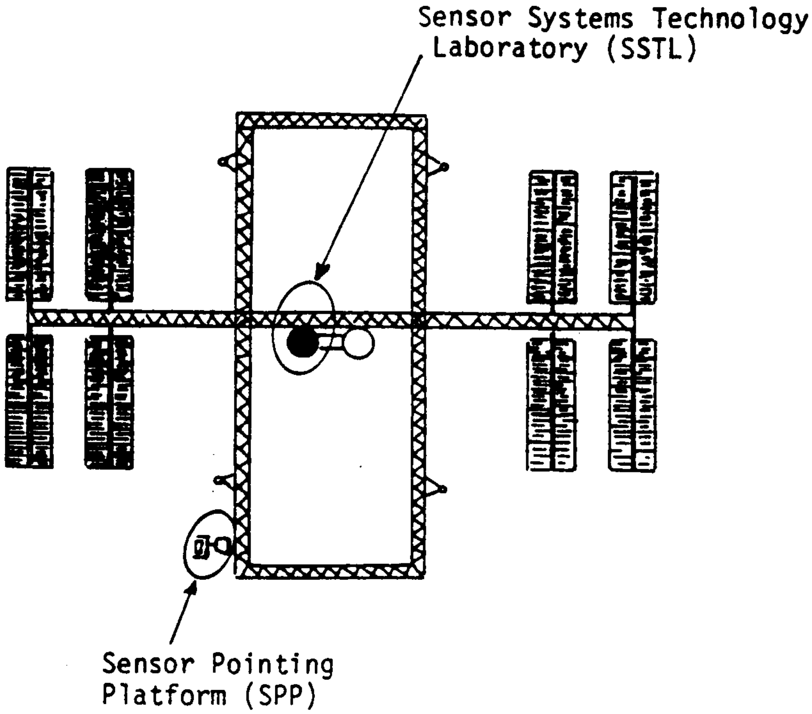


Figure 7-1. Sensory Systems Technology Laboratory.

7.3.2 Viewing Requirements

The Sensor Systems Technology Laboratory will be used for research on instruments with a wide range of targets and associated viewing requirements. These include:

7.3.2.1 Solar Measurements: Clear field-of-view (FOV) of ± 30 arcmin in azimuth to the sun's direction as it traverses from the horizon to a zenith angle of 15 degrees. It is also desirable to have the same FOV from a zenith angle of 15 degrees to the horizon as the sun sets.

7.3.2.2 Earth-Limb Measurements: Clear FOV across the earth's horizon over 180° of azimuth (orbit plane forward to orbit plane aft) is required. Complete 360 degrees azimuthal clearance is desirable. The vertical clear FOV over this azimuthal range must be between nadir angles of 62 to 72 degrees to cover the Earth's rim and atmosphere.

7.3.2.3 Earth-Nadir Measurements: Clear FOV of ± 10 degrees along orbital plane, and ± 30 degrees transverse to orbital plane is required for atmospheric sounders and earth scanners.

7.3.2.4 Space Look: A number of sensors require Clear FOV's to "cold space" as a calibration and/or anchor point. This is defined to be in the anti-solar-direction quadrant and must be at least 5 degrees above the earth's horizon and at least 15 degrees from the sun's direction. It is required that at least twenty minutes of clear space viewing be available per orbit, made up of single clear windows of 5 minutes or more each.

7.3.3 Pointing, Pointing Stability Rates, Pointing Determination

The pointing requirements for the various sensor types involved in Sensor Systems Technology are listed in Table 7-1.

7.3.4 Structural Considerations

As previously described in Sections 7.2 and 7.3, accurate determination of the sensor pointing angles relative to viewed target (e.g., Earth or Sun) is required for interpretation of experimental data. This implies a

Table 7-1. TDMX 2261 - Pointing Accuracy, Stability, Pointing Determination Accuracy Requirements for Sensor Systems Research

Measurement Target/Type	Pointing Accuracy 1σ arc secs	Pointing Accuracy 1σ degrees	Pointing Stability degrees per sec	Pointing Determination, 1σ arc secs	Pointing Determination, 1σ degrees
Solar	58	0.016	0.006	36	0.010
Earth-Limb	100	0.030	0.004	36	0.010
Earth-Nadir	360	0.100	0.008	180	0.050
Space Look	3600	1.000	0.015	1800	0.500

requirement to relate the sensor line-of-sight not only to the Sensor Pointing Platform, but to the Space Station Attitude Control System (ACS) and Guidance, Navigation, and Control (GN&C) System as well. Location of the Sensor Pointing Platform on the Space Station to provide the sensor viewing fields-of- view (See Sect. 7.3.2) will require careful consideration to avoid problems because of the Space Station structure, e.g., Lower Boom, Radiators, other payloads. The pointing stability requirements (Sect. 7.3.3) must consider the dynamics of the Space Station Structure as well as those of the Attitude Control System.

7.4 SENSORS AND HARDWARE

The Sensor Systems Technology Experiment has a wide range of objectives resulting in an extensive list of equipment and instrumentation requirements. Table 7-2 includes a summary listing of these requirements.

The Sensor Pointing Platform will be mounted remotely from the Sensor Systems Technology Laboratory so that the required clear-fields-of-view specified in Section 7.3.2 can be obtained. A design concept for such a platform is illustrated on Figure 7-2. The pointing platform will provide two degrees of freedom, having $\pm$180 degrees about the SS Alpha axis and approximately 240 degrees about the Beta axis. The sensor mounting table should be a minimum of 1 meter by 1 meter, and will be removable from the pointing gimbal so that a flight test sensor can be mounted, aligned, and tested on the table while within the manned laboratory, then carried by EVA for mounting on the pointing gimbal. Provisions for coupling sensor power, control, data acquisition, thermal control, etc. through the pointing gimbals must be included, in addition to provisions for attitude determination/control sensors which must also be mounted on the table and aligned with the sensor under test. The Sensor Pointing Platform, and its research sensor payload, will be connected to the manned Sensor Systems Technology Laboratory through the Space Station SISS Core Network and the Payload Accomodations Systems, as illustrated on Figure 7-3. The Sensor Systems Technology Laboratory (SSTL) will be a portion of a laboratory

Table 7-2. TDMX 2261 - Summary of Equipment/Instrumentation Needs

A. Equipment

- Two-Axis, remotely controlled sensor pointing platform
- Solar Tracking Heliostat directing Solar Flux into Laboratory
- Optical relay transferring solar flux from heliostat to sensors under test
- Optical-bench equipment, tooling, and components, including mounting hardware, stand, etc., which can be disassembled for compact storage
- Cooling system and components for thermal stabilization of sensors, electronics, calibration, and test equipment
- Cryogenic Coolers for detectors and sensor components
- Equipment and materials for cleaning of optical and instrumentation components

B. Sensors

- Solar Aspect Sensors
- Star Tracker
- Instrument Type Photographic Camera

C. Optical System Assembly and Alignment Instrumentation

- Alignment Laser Systems
- Autocollimator/angle Measure
- Optical Components: Mirrors, Lenses, etc.
- Alignment Telescope
- Visible/IR Imagers with Video Display, Data Storage, and Playback
- Machinist's Microscope
- Microposition Sensors
- Microposition Manipulators

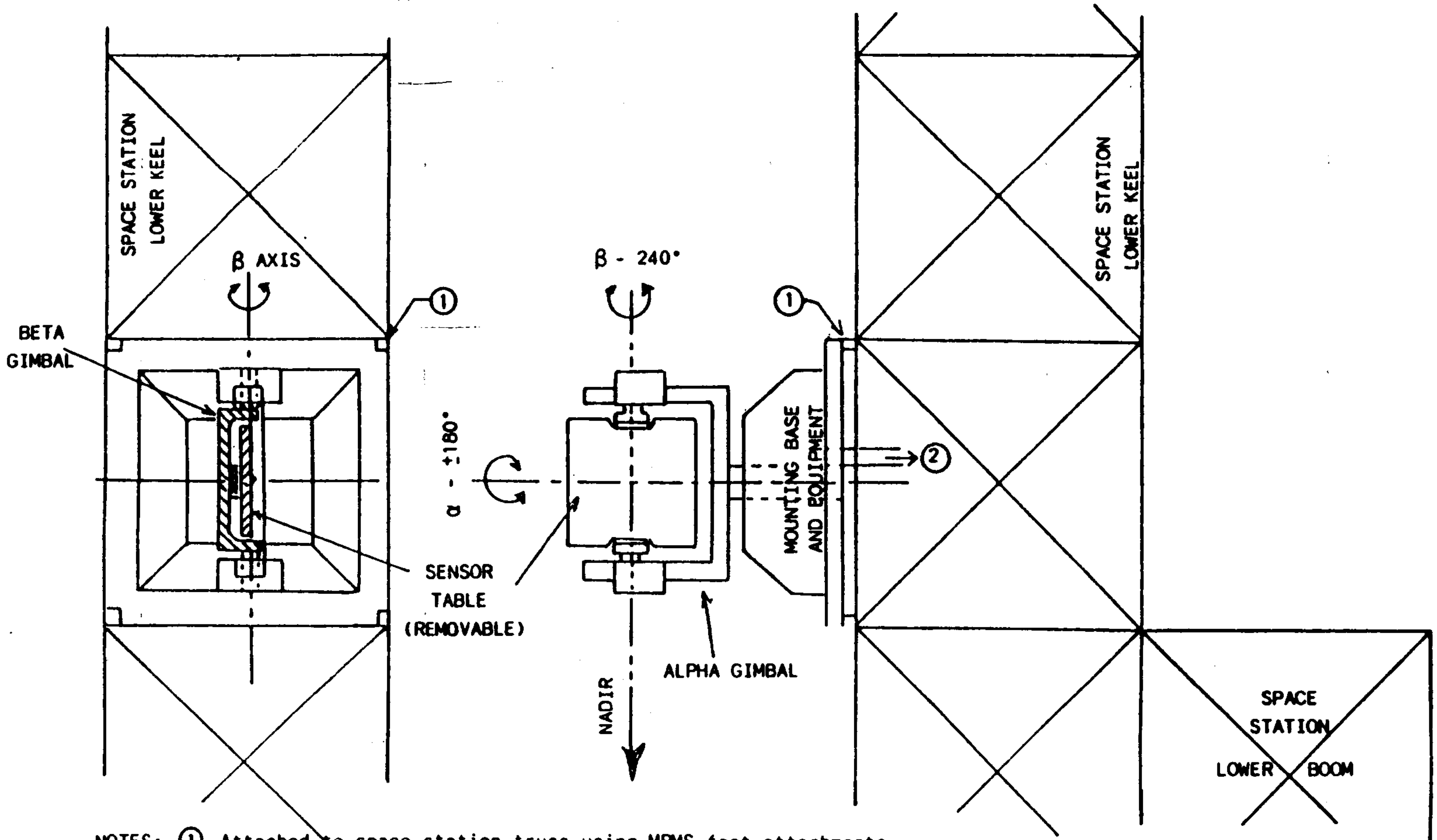
Table 7-2. (Continued)

D. Calibration Instrumentation and Standards

- Monochrometer (Sun as source)
- Integrating Sphere (Sun as source)
- Master Reference Blackbody (Traceable to IPTS-68)
- World Radiometric Reference Transfer Instrument
- Transfer Standard Radiometer(s)
- Silicon Photodiodes and Electronics for use as Absolute or Transfer Standards

E. Electronic Instrumentation for

- Power Conditioning, Clock, and Command
- Programmable Signal Conditioning, Processing, and Recording
- Computer Real-time Control, Data Processing and Display, and Storage
- Computer Off-line Data Processing, Data Analysis with Records Storage/Recall
- Up-link and Down-link Telemetry Transfer
- Electronics Laboratory System Testing



NOTES: (1) Attached to space station truss using MRMS foot attachments
(2) Power, electronic, and cooling connections through space station payload accomodations system to SSTL.

Figure 7-2. Sensor Pointing Platform: A Design Concept.

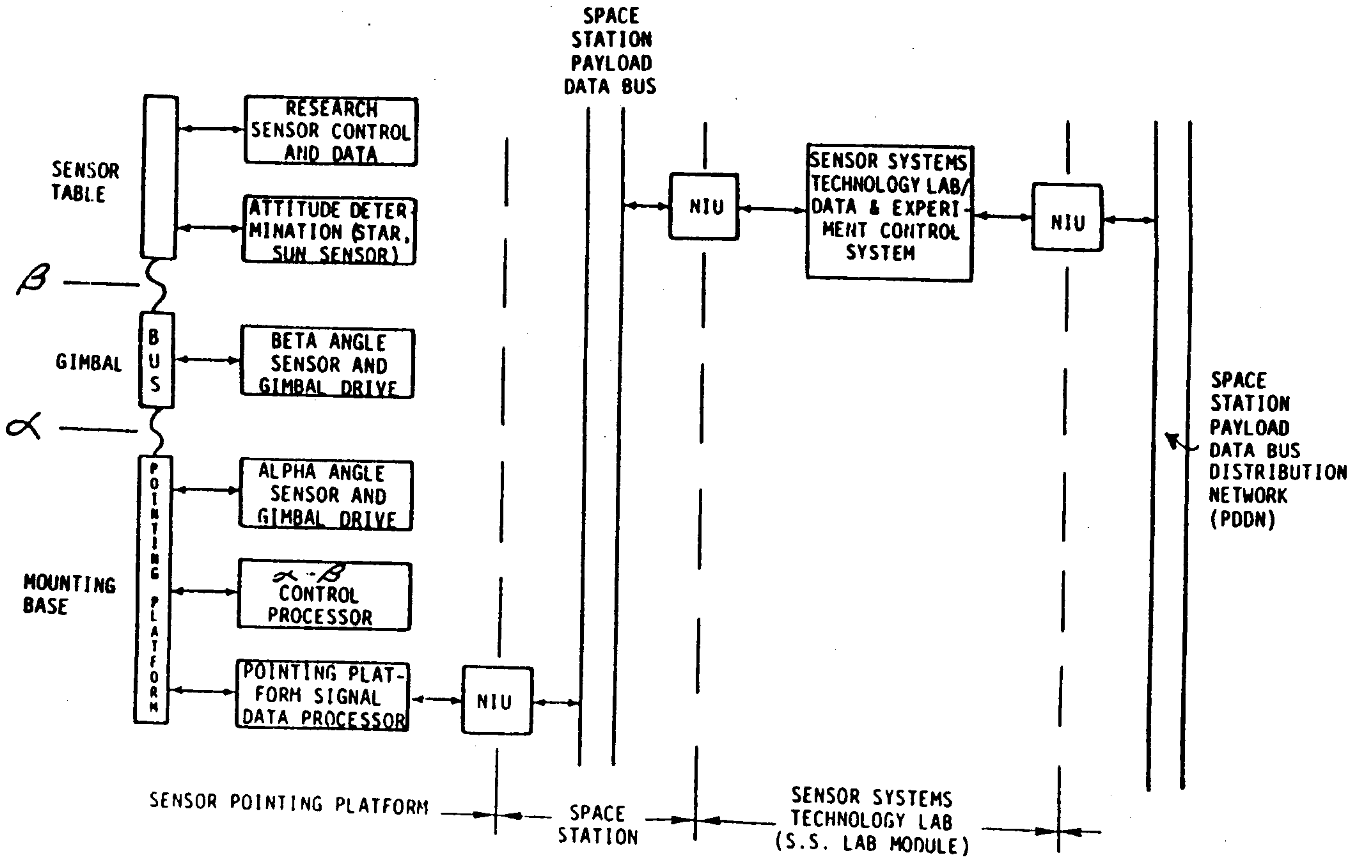


Figure 7-3. Sensor Systems Technology Experiment - Signal Data Flow Concept.

module in the Space Station Manned Core. The lab will provide the work areas for assembly and test of the research sensors, and will contain the electronics for control of the pointing platform and the research sensor systems as well as for data acquisition, handling, and reduction. The laboratory module also requires an externally mounted solar tracker/heliostat which relays solar energy into the lab through an optical quality window to permit sensor testing using the sun as a source. The solar tracker must also have an unobstructed view of the Sun for extended periods of time, and should have the same field-of-view as for any solar sensor (Section 7.3.2.1). The laboratory must be designed with multiple internal attachment points for optical bench stands required for sensor testing; to allow the stands to be designed, assembled, and securely anchored to "build" optical setups as required. These stands and other optical bench components must be suitable for disassembly and compact storage when not in use. Generally, all electronics for control, signal, and data processing and display will be rack mounted and accessible through data buses. A schematic representation of the sensor system data and control electronics is shown on Figure 7-4. A similar, but smaller dedicated system will be provided for control of the Sensor Pointing Platform.

7.5 DATA ACQUISITION, HANDLING, AND CONTROL

The Sensor Systems Technology Laboratory will generally operate as an autonomous unit as regards real-time data acquisition and control for the various experiments performed. The fact that the Space Station SISS Core Network and the Payload Accommodation System are used to connect the Sensor Pointing Platform to the SSTL makes the preceding statement technically incorrect, and will normally require transmission of 20 to 30 kbps of data which is actually internal to the Sensor Systems Technology Experiment. The laboratory will also depend on the Space Station for real-time voice communications with ground personnel for perhaps 2 hours at the time, approximately once per 24 hours. Data which is collected by the laboratory will periodically be relayed through the Space Station to ground personnel

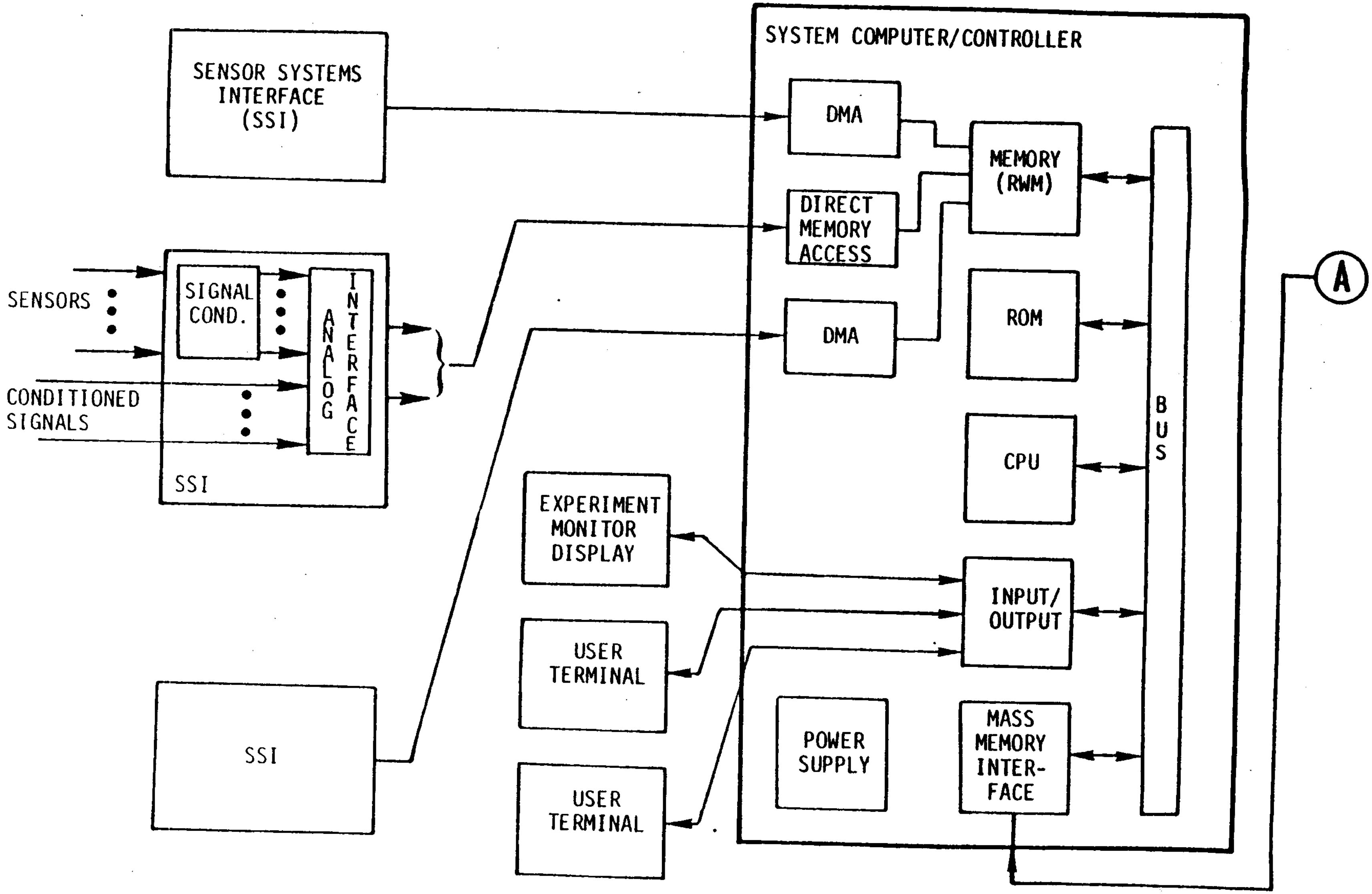


Figure 7-4(a). Sensor Systems Technology Laboratory - Data and Experiment Control System.

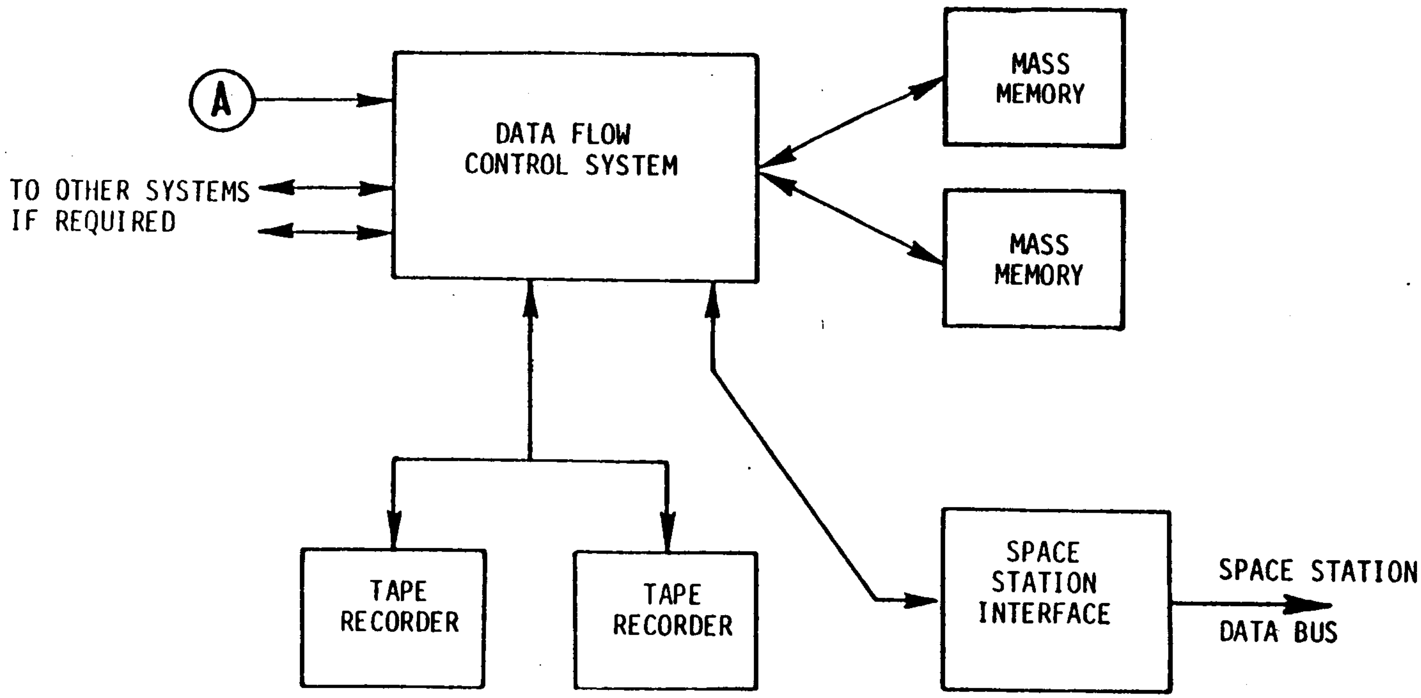


Figure 7-4(b). Sensor Systems Technology Laboratory - Data and Experiment Control System - Memory and Space Station Interface.

for data interpretation/validation. It is anticipated that most of this data will be at relatively low rates (12-20 kbps) for limited time periods (approximately 10 minutes) and most tests will require transmission of $10^5 - 10^7$ bits of data. For experiments involving imaging sensors, however, the laboratory can be collecting data at rates of 1.25 to 35 Mbps for several minutes, therefore, real-time transmission of these data becomes more difficult since it consumes a significant fraction of the 300 Mbps available in the SS-TDRS Ku-Band link. It is not anticipated that these rates will be required very often, however, and the total data transferred will be approximately 10^{10} bits. As illustrated in Figure 7-4, it will contain an internal command and control capability used by the scientist-astronauts for sensor pointing, and for sensor system operational control, including real-time data processing. The laboratory will depend on the space station voice and digital data up-links and down-links to obtain and verify data and programs to conduct the various tasks, in addition to transferring experimental data for ground evaluation. The laboratory electronic data storage will also include an information and data file which maintains an "instrument log" on the sensors as modifications and/or tests are performed. In a sense, the laboratory will utilize its own centralized data system to accomplish its tasks, and will utilize Space Station data and voice links for communication with the remotely located Sensor Pointing Platform and with other station and ground activities.

7.6 CREW ACTIVITIES

The Sensor Systems Technology Laboratory is planned to ultimately evolve into a continuous applied research and development facility including operation and maintenance which will probably require almost full time activities of a staff of two persons, occasionally supplemented by a third earth-trained specialist who accompanies instrumentation and/or performs specific tests. The laboratory staff will initially be a team of two led by an Engineering professional with specialized training in the Physical Sciences and Earth and Ocean Sciences, and include a highly

Sciences, and include a highly qualified Engineering technician. All team members must have training and hands-on experience with optical systems and testing, and radiometric calibration, in addition to electronic systems and data processing electronics.

When the SSTL is integrated into the Space Station, extensive EVA will be required to install the laboratory equipment into the laboratory module. The equipment will be designed into racks and storage cabinets suitable for transport into, and mounting in, the module. However, it is anticipated that approximately 15 racks must be moved into and mounted in the laboratory module. Assuming two hours EVA for 2 persons per equipment rack, this would require approximately 60 man hours of EVA. Installation and checkout of the externally mounted solar tracker on the laboratory module is estimated to require approximately 6-8 hours EVA for a 2 man team, assuming the optical quality window and tracker mounting provisions had been designed and fabricated into the module prior to its launch. Subsequently, internal integration and checkout of the SSTL equipment is estimated to require approximately 60 hours of IVA for a two-man team (120 man hours). When the SPP is installed and integrated with the Space Station, it is estimated for a two-man team will be required (35-40 man hours) to mount the unit, align it as required, and complete service connections to the SSTL. During and after integration of the SPP, it is estimated that checkout and testing of the unified system will require approximately 60 man hours of IVA. Once the SSTL is operational, it is planned that EVA will be limited to maintenance of the SPP and maintenance or modification to the research sensor under test. It is anticipated such activities will be limited to 4 hours of EVA for a two-man team (8 man hours) every 30 days early in the program and decrease to once every 45 to 90 days as experience is gained. Early task assignments will be directed to assembly, test, and/or calibration of small instrumentation/sensors within the SSTL as a learning experience for the crew and to verify laboratory standards and equipment by comparisons with pre-launch data obtained on the ground. When the crew gains confidence and the internal laboratory systems have been verified, the laboratory tasks will be expanded to include work on remote sensor systems which utilize the SPP. Because of the breadth of the types and complexity of work required, it will not be possible to provide detailed descriptions or timelines for the tasks; rather it would be more properly described as a "Nine-to-Five" laboratory operation, where a series of tasks to assemble, checkout, test, and calibrate sensor systems are done, and specific crew-controlled flight experiments are performed in coordinated fashion with ground personnel to accomplish the R&D objectives. For later research programs, PERT type charts and timelines will be developed based on the work required and the work-rate experience developed during early Sensor Systems Technology experiment activites. Currently, it is anticipated that one research program on a remote sensor system will be accomplished at the time and that a specific program requiring sensor assembly/modification with flight data evaluation will require approximately 90 days to complete. It is anticipated the SSTL will be resupplied on a 90 day cycle, and it is assumed that each will require 6 to 8 hours EVA for a two man team (12-16 man hours) per resupply event. To illustrate crew requirements during normal research Table 7-3 provides an activity timeline for a typical flight test day where data is to be obtained for about 3 orbits using a sensor system mounted on a previously deployed pointing platform.

7.7 GROUND PROCESSING

The ground activities associated with the Sensor Systems Technology Lab will be intermittent and require that digital data and video information be exchanged between the SSTL and ground personnel, and that voice communications be maintained for 2 or more hours at a time. The digital data and video information to be up-linked to the SSTL will include data on recommended sensor system programming for system control and data handling to perform research and testing of instrumentation in the Laboratory and will provide reference data including detailed sensor design and test criteria. The video will be test plans and schematics and include data for discussion by the combined lab and ground test team prior to and after Space Station testing. The down-linked data will include sensor system programming and

Table 7-3. TDMX 2261: Perform Operational Test of Remote Sensor Systems

FUNCTION/TASK	TIME MANMINUTES	METHOD	FEASIBILITY	COMMENTS/ASSUMPTIONS
1. All systems turn on/warm up	30	IVA	2	Assumptions A-1, A-2, A-3, A-4
2. Sensor system check-out and performance verification	45	IVA	2	
3. Pointing platform checkout/ verification	15	IVA	2	
4. Procedure verification for data taking	15	IVA	2	
5. Initialize sensor, platform, data display, transmission, recording	15	IVA	2	
6. 3 orbits or intermittent data taken according to plan with data processing in between	300	IVA	2	Assumption A-5
7. Sensor system put into standby mode for data check	10	IVA	2	

Table 7-3. (Continued)

FUNCTION/TASK	TIME MANMINUTES	METHOD	FEASIBILITY	COMMENTS/ASSUMPTIONS
8. Recordings removed from system	5	IVA	2	
9. Remote sensor powered down port	10	IVA	2	
10. Pointing system into stow position and powered down	10	IVA	2	
TOTAL ELAPSED TIME	(7.6 hours)			

Total Manhours -IVA 18.00

3 Crew Members Involved

NOTE: Feasibility Assessment: 1-Proven, 2-Easy, 3-Hard, 4-Questionable

GENERAL ASSUMPTIONS

1. Sensor on previously deployed pointing platform and has been previously operational.
2. Test plan for data taking previously established and studied by team members.
3. Required supporting data from Space Station subsystems has been previously "wired" into remote sensor data system.
4. Two man team required for entire operation. Third man provides system monitoring support during actual data taking.

test parameters for ground review and evaluation prior to tests. As the tests are conducted in the SSTL, real-time output data will be transmitted to earth as digital data and by video link from the onboard system displays. For a number of tests, more detailed output data from the sensor system may be down-linked post-test based on information stored or recorded in the SSTL. It is anticipated that the data down-linked from SSTL may require distribution to several locations (e.g., two NASA centers and an aerospace company) for complete analysis and interpretation. The end products of the SSTL activities will include refined analytical models of remote sensors, optimized remote sensor design parameters, refined design information, and criteria for advanced remote sensors.

7.8 SUPPORT EQUIPMENT

The detailed support equipment to conduct the Sensor Systems Technology experiments and technical activities have been included as payload equipment, including specialized liquid coolant loops to thermally stabilize rack mounted electronics, computers, and display equipment, in addition to cryogenic equipment to maintain radiation detector temperatures. However, the Space Station Thermal System will be required to transport and dissipate the collected energy from both the SSTL located in a laboratory module and from the SPP located remotely (e.g., on the SS Lower Keel or Lower Beam). The laboratory module which houses the SSTL must include an optical quality window to allow relaying of solar energy from the externally located solar tracker into the laboratory. This window, and its mounting into the module wall, should be designed to facilitate the external mounting of the solar tracker itself, and include provisions for signal and power leads required for solar tracker operations. As mentioned previously, it is planned that the power, command, and data connections between the SSTL and SPP will utilize the Space Station Payload Data Bus and the Payload Accommodations System. In the event this cannot be done, special provisions to accomplish these purposes will have to be developed.

7.9 SPACE STATION RESOURCE REQUIREMENTS

The requirements given in this section were derived by estimating payload equipment inventory, then using available data from brochures and other reports, as well as the knowledge of persons experienced in performing similar tasks, to estimate size, weight, and power requirements for the equipment. The data were compiled into two groups, i.e., that located on the remote Sensor Pointing Platform and that located in the Sensor Systems Technology Laboratory. Accordingly, the following sections will present the resource requirements by location assuming operational status of the SSTL/SPP. These data are summarized on Tables 7-4 and 7-5 for each location.

7.9.1 Weight

Total experiment weight is estimated to be 4660 kg.

7.9.1.1 Sensor Systems Technology Laboratory: The weight of payload equipment contained in the SSTL is estimated to be approximately 1933 kg, of which approximately 1000 kg consists of mounting racks and cooling provisions.

7.9.1.2 Sensor Pointing Platform: The weight of the SPP is estimated to be 2727 kg. The Pointing Platform structure is the primary contributor and is estimated to be 2509 kg.

7.9.2 Power

The total power requirements for the Sensor Systems Technology Experiment are estimated to be 5.4 kW peak power, with an operating power of 4.7 kW average for 6 hours per 24 hour period. The standby experiment power for the remaining 18 hours per 24 hour period is estimated to be approximately 2.1 kW. The power is consumed in two (2) locations, i.e., the SSTL and the remotely located SPP, as detailed in Sections 7.9.2.1 and 7.9.2.2 and in Tables 7-4 and 7-5.

Table 7-4. TDMX 2261: Space Station Resource Requirements
Sensor System Technology Manned Laboratory

ELEMENT/SUBSYSTEM	VOLUME** m^3	WEIGHT kg	POWER WATTS	THERMAL DISSIPATION WATTS
1. Solar Tracker/Optical Relay	0.1	73	110	100
2. Optical Test Equipment	0.6	251	1175*	1175*
3. Calibration Equipment	0.2	62	300*	300*
4. Electronics Test Equip./Power Sup.	0.2	110	5710*	5710*
5. Electronic Command, Data Display System	0.7	283	1200	1200
6. Sensors	0.1	27	90	90
7. Miscellaneous Supporting Equipment	0.4	127	260	260
SSTL Equipment Totals	2.3	933	4055*	4055*
Mounting Racks/Cold Plates	0.2	1000	--	--
SSTL TOTALS	2.5	1933	4055*	4055*

* Assumed 1/3 use rate for special purpose equipment (items 2, 3, and 4).

** Volume does not include approximately 20 cubic meters of working space within laboratory module.

Table 7-5. TDMX 2261: Space Station Resource Requirements Sensor Pointing Platform

ELEMENT/SUBSYSTEM	VOLUME** m^3	WEIGHT kg	POWER WATTS	THERMAL DISSIPATION WATTS
Pointing Platform Structure	260/405*	2509	--	--
α-β Gimbal Drive System	7.1	84	300	300
Command and Data Electronics	2.9	126	280	280
Platform Signal Bus	--	4	--	--
Research Sensor (Under Test)	2.0	110	130	130
Attitude Determination Sensors	2.0	53	130	130
Research Sensor Electronics	1.5	22	100	100
Supporting Equipment	4.5	80	360	360
Equipment Mounting Hardware	1.0	150	--	--
SPP TOTALS	7.9/11.4*	3138	1300	1300

* SPP volume required to allow $\pm 180^{o}$ of alpha angle freedom.

7.9.2.1 Sensor Systems Technology Laboratory: The estimated peak power for equipment in the SSTL is 4.1. kW and the estimated average power utilized is 3.6 kW for 6 hours each 24 hour period. The standby power for the remaining 18 hours per 24 hour period is estimated to be 1.5 kW or less.

7.9.2.2 Sensor Pointing Platform: The estimated peak power for the SPP is 1.3 kW with operating average power of 1.1 kW for a period of 6 hours each 24 hour period. The standby power for the remaining 18 hours per 24 hour period is estimated to be approximately 0.6 kW.

7.9.3 Volume

The Sensor Systems Technology experiment requires an internal volume of approximately 2.5 cubic meters in a laboratory module for equipment mounting and storage. An additional 20 cubic meters of internal volume must be available for working space to permit optical and calibration setups. An external volume is also required on the laboratory module of 1.5 cubic meters when the Solar Tracker/Heliostat is deployed. An additional external volume of approximately 11.4 cubic meters is required for the remotely located Sensor Pointing Platform.

7.9.4 Thermal Requirements

7.9.4.1 Sensor System Technology Laboratory: The estimated thermal dissipation of equipment located in the SSTL module is approximately 3.6 kW for periods of 6 hours duration. Over a 24 hour period, the thermal dissipation within the SSTL will be approximately 48.5 kWh.

7.9.4.2 Sensor Pointing Platform: While the maximum peak payload thermal dissipation on the SPP may approach 1.3 kW for one-half hour or less, the normal level during operation is estimated to be 1.1 kW for periods of 6 hours each. Over a normal 24 hour period, the required thermal dissipation will be approximately 18 kWh.

7.10 INTER-EXPERIMENT COMPATIBILITY

The Sensor System Technology Experiment - TDMX 2261, will contribute to the success of several other proposed TDMX experiments, e.g., TDMX 2111, 2121, 2263, 2264, 2265 since it contains electronics and test equipment which supplements that required by those experiments. TDMX 2261 will also contribute strongly to the future development and maintenance of the Space Station Platforms intended for Earth Observations in an operational sense. The contributions to these platforms will begin at the remote sensor design level by experimental determination of optimum sensor designs and concepts, including signal processing and data processing. The contributions can continue if SSTL grows to become a check-out way station as the platform sensor systems are transported from Earth to Space Station enroute to the platforms. Such an expanded SSTL could also serve as a check out/calibration/maybe repair laboratory for platform sensors during the periodic servicing of the platforms by the station. The SSTL will be sensitive to other experiments which could cause shading of the lab and thus adversely affect the thermal stability of the equipment and perhaps the required clear-fields-of-view from the lab. The SSTL requirements for pointing control and determination accuracy, and for pointing stability, during those time when flight tests of remote sensor are underway, will be sensitive to attitude perturbations and induced structural dynamics resonances. Thus experiments and station operations where large space structures are being moved about may adversely affect the success of SSTL operations.

8.0 TDMX 2263 - CO_2 DOPPLER LIDAR WIND SENSOR

CONTACT: Robert V. Hess, MS 283
NASA LaRC (804) 865-3761

8.1 OBJECTIVE

The objective of this experiment is to develop design criteria, instrumentation, signal and data processing algorithms for LIDAR remote sensors to measure vertically resolved two dimensional atmospheric winds from space platforms. The technology thus developed and demonstrated will provide the means for further operational measurements of winds from the Polar Orbiting Platform Earth Observation System (EOS); as an extension of the Space Station Manned Core capabilities. The inital emphasis will be on lidars using high powered CO_2 lasers as transmitters. As additional solid state or gaseous lasers which have suitable power, pulse fidelity, and wavelength characteristic's become available, the Space Station lidar facility developed for this experiment can be adapted to other transmitter technology and for other atmospheric parameter measurements.

8.2 DESCRIPTION

The experiment requires a Lidar Facility attached to the Space Station Manned Core to permit interactive research and development for optimizing lidar sensor systems hardware and software designs for remote measurements of atmospheric parameters. The lidar shown schematically on Figure 8-1 consists of a telescope having a circular aperture of 1.25 meters, a pulsed CO_2 laser transmitter, and a coherent heterodyne detector. The telescope must have an optical performance consistent with the heterodyne detection of laser energy backscattered from the earth's atmosphere. The atmospheric parcel being measured must be viewed from two or more angles in order to resolve two-dimensional winds, therefore, the telescope must be scanned or pointed as the Space Station moves along its orbit. For the doppler measurement of winds, the CO_2 laser output pulse will be transmitted through the same telescope that receives the atmospherically backscattered signal. In this way, boresighting of the laser transmitter beam and

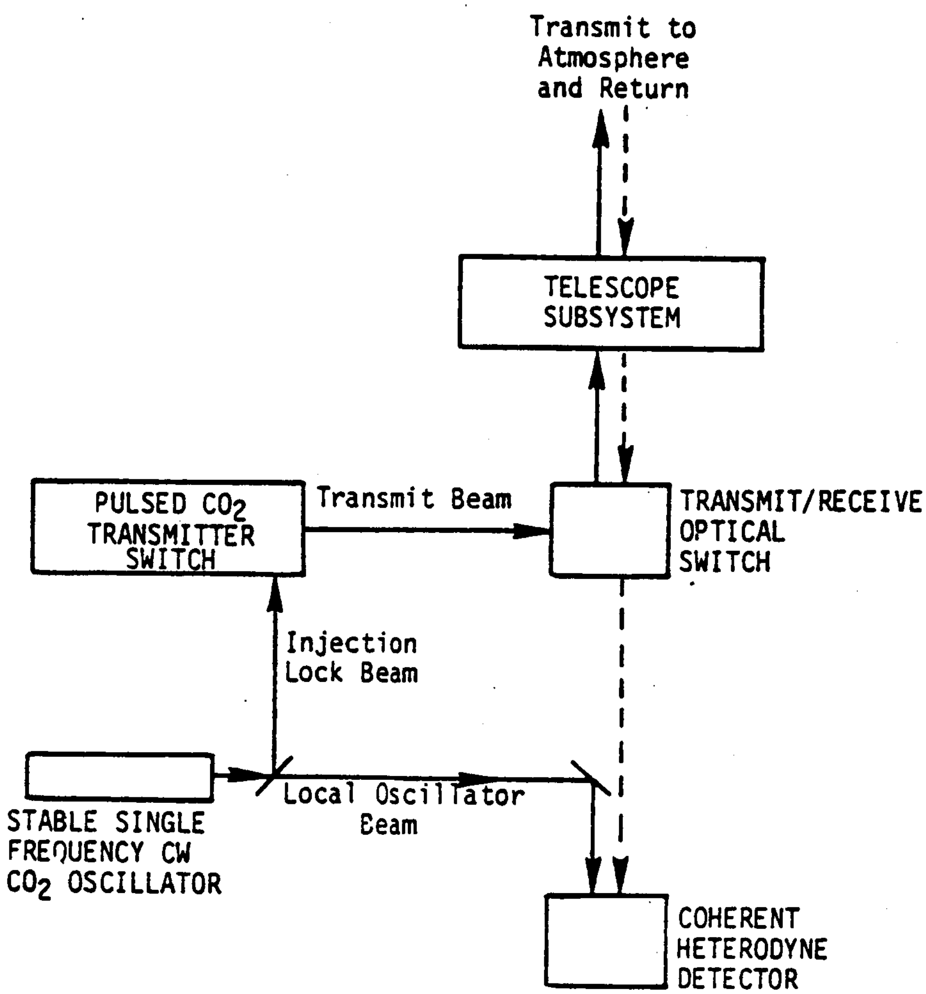


Figure 8-1. Block Diagram of CO_2 Laser and Wind Lidar.

receiver is maintained as the sensor is scanned. The telescope and its pointing and scanning pedestal (shown in Figure 8-2), will be mounted outside the pressurized area, with the laser transmitter and the telescope receiver detector modules mounted internally in the Station for improved monitoring and serviceability. This arrangement will also facilitate configuration changes as required. An optical relay through the mounting pedestal couples the laser transmitter and receiver detectors to the telescope optics. The electronics for the system control and data processing are internally rack mounted within the Station. The laser transmitter for this experiment can be modified and/or controlled to allow parametric investigations of lidar design characteristics including laser wavelength, pulse power, pulse duration, repetition rate, etc. The electronics for system control, signal and data processing will be programmable where possible to permit maximum research flexibility as the experiments progress. In addition to early determination of optimized system design concepts and parameters for the future EOS lidar, it is anticipated the Space Station system can actually emulate the design chosen for EOS and verify system performance predictions as the design progresses.

8.3 ENVIRONMENT

8.3.1 Platform and Orbit

The lidar facility will be a part of the Space Station Manned Core. It can operate successfully in the planned Space Station orbit at 500 km altitude and 28.5 inclination. The orbit should be nearly circular, and the lidar needs information from the Space Station yielding orbital information on altitude to 1/2 km, 1σ, on orbital position to 10 km, 1σ, on orbital velocity to 1 meter per sec., 1σ.

8.3.2 Viewing and Pointing Requirements

The planned applications for the lidar facility present two requirements for Earth viewing. For the initial Doppler lidar to measure wind, a conical field-of-view (FOV) is required, i.e., the small instantaneous field-of-view (IFOV) must be adjustable between nadir angles of 52 to 62 degrees and then scanned in azimuth over 360 degrees. In the event

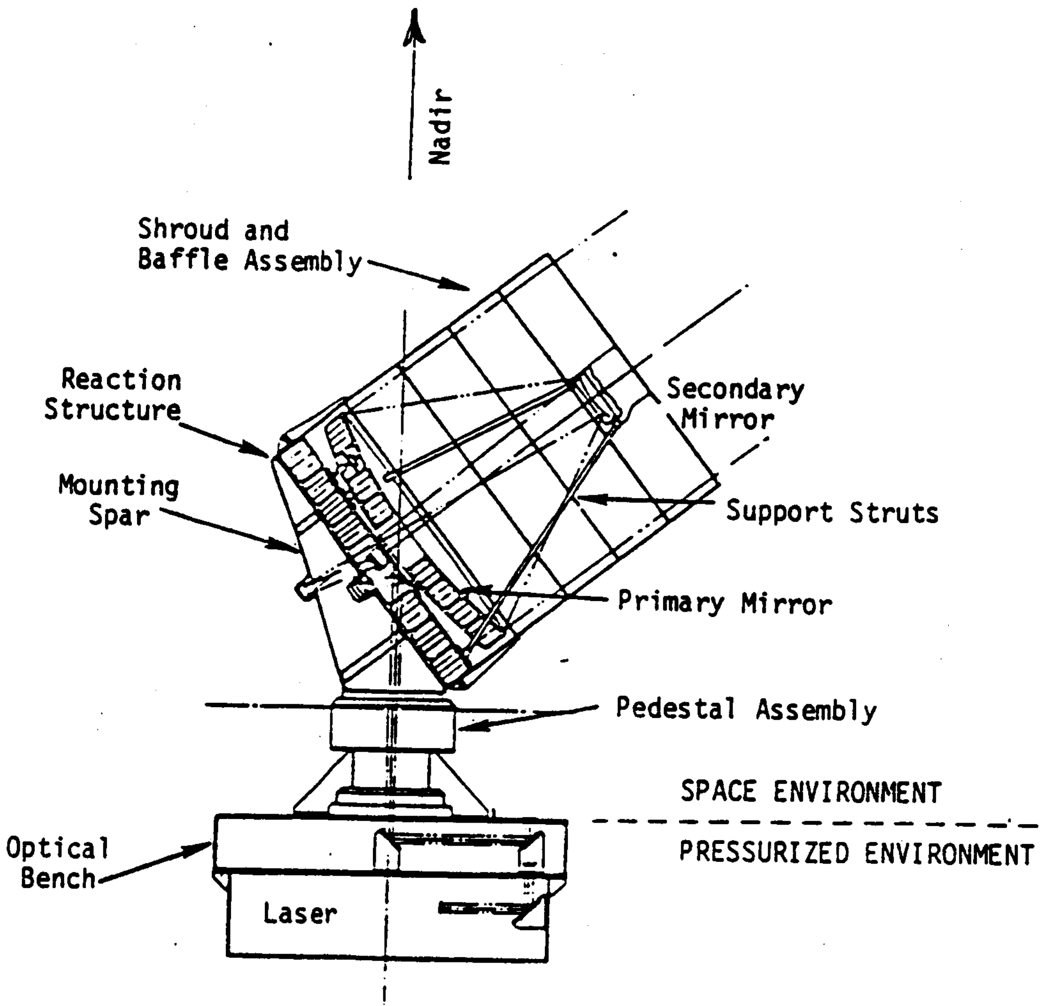


Figure 8-2. CO_2 Lidar Telescope System Concept [1].

complete azimuthal clearance cannot be obtained, a minimum of 180 degrees (from orbit plane forward to orbit plane aft) is required so that the same atmospheric parcel can be viewed from two azimuth angles as the Space Station moves along its orbit. The nadir angle setting, i.e., 52-62, degrees and the conical scan are needed to allow the Space Station facility to emulate an operational wind lidar so that hardware and software designs can be optimized. For the winds measurement, the pointing accuracy of the telescope must be 2.10 μrad's (0.18 degrees), 1σ. Knowledge of the instantaneous FOV pointing direction must be to an accuracy of 250 μrad (0.014 degrees), 1σ for each laser firing. The IFOV pointing stability must be better than 100 μrad/sec (0.006 degrees/sec) so that IFOV does not "jitter" during the approximate 9.1 msec round trip time for a laser pulse.

8.3.3 Structural Considerations

The primary requirements on structure supporting this experiment are imposed by the need to maintain the optical alignment between the externally mounted telescope/pointing platform assembly and the internally mounted laser transmitter and the receiver detector assemblies. Other potential problems could be the pointing stability requirement and the pointing determination. For the complex relatively light-weight Space Station structural dynamics, the pointing stability may require a lidar pedestal isolation/stabilization system. The pointing determination using Space Station data alone may also not be feasible and thus require special instrumentation.

8.4 SENSOR(S)

As previously described, the experiment utilizes a Light Detection and Ranging (LIDAR) System for remote measurement of atmospheric winds. This experiment will take advantage of preliminary design studies previously performed to study the feasibility of space platform mounted lidar to measure atmospheric winds [1,2,3]. The stringent requirements for knowledge of the telescope transmitter/receiver pointing angle and for receiver pointing stability will require that the internally mounted laser transmitter and the receiver electro-optics be rigidly attached to the telescope pedestal to maintain system optical alignment. The Space Station dynamics may exceed experiment requirements, and thus require that the telescope pedestal and attached subsystems be isolated from station structure. The telescope lag angle compensation system used to correct for scan movement during the laser pulse round trip time must be redesigned for the nadir angle(s) and scan rate(s) selected for the Space Station wind measuring lidar; however, previous analyses indicate the proposed design concept is suitable for the expected range of angles, scan rates, and for Space Station orbital parameters. It is possible the laser subsystem proposed in [1, 2] could be used for this experiment, however, recent technology improvements and the desire for research flexibility will probably dictate several changes. For the Space Station experiment, the 10 Joule pulse energy is still required and the pulse duration should be adjustable between 2 and 6 microseconds rather than a fixed value. Rather than a fixed pulse repetition frequency (PRF), it is desired the PRF be adjustable from 2 to 10 Hz. For the early experiments, the wavelength requirement will be 9.11μm ($^{12}C^{18}CO_2$ isotope). The UV-preionized self-sustained discharge approach proposed in [1] will be retained over the e-beam stabilized discharge to reduce the required voltages from 80-100 kV to 14-20 kV and to reduce required radiation shielding for the internally mounted laser. The Pulse Modulator Circuit proposed in [1] is also expected to be retained. The desire for adjustable laser pulse "characteristics" and PRF, coupled with recent work on Master Oscillator Pulse Amplifiers (MOPA) [4], make MOPA an attractive alternative to the Unstable Resonator/Injection Locked system proposed in [1] and the MOPA approach will be studied in detail for use on the Space Station Lidar. With the exception of the changes recommended here, the design approach for the sensor hardware proposed in [1] appears viable for the Space Station. Additional work to verify the performance of the $^{12}C^{18}CO_2$ isotope in systems of this power, and to verify sealed-off closed-loop operational lifetime of this isotope with available catalysts, is strongly recommended [4].

The system just described will obtain measurements of laser pulse energy backscattered by the earth's atmosphere which can be processed to measure Doppler shifts of the laser wavelength caused by wind velocities within twenty altitude regimes (range cells) of the atmosphere. For each laser pulse, therefore, processing must provide at least three parameter values for each of the 20 altitude regimes (range cells); namely, total power of the signal scattering within range cell, mean frequency of the return signal (to determine Doppler due to wind), and the frequency spread about the mean (to determine atmospheric variability within range cell). To determine these parameters, a number of additional measurements are required for each laser pulse including laser transmitter frequency (wavelength), laser pulse power and duration, Space Station orbital velocity, altitude, and latitude, Space Station attitude angle and rates, telescope pointing angles and scan rates. The basic requirement is to measure atmospheric winds over a range of $\pm$ 100 m/sec with a resolution of 1 m/sec. At a 9.1μm laser wavelength, this corresponds to a doppler frequency range of approximately $\pm$ 22 MHz, with a frequency resolution of 220 kHz. The additional atmospheric velocity "background" imparted by Space Station orbital motion and by rotation of the earth itself is much larger than that of the wind. Platform orbital motion introduces Doppler shift of approximately $\pm$1.4 GHz which varies sinusoidally with telescope pointing angle. The earth's rotation also introduces an additional doppler of up to 80 MHz depending on orbital position and telescope azimuth angle. It is necessary to determine all of the factors involved (orbital parameters, pointing angles, geocentric position, etc.) to an accuracy which allows compensating the approximately 1.41 GHz background bias with an uncertainty of 200 kHz or less. The preceding requirement led to the values indicated earlier in Sections 8.3.1 and 8.3.2. For the wind measurement, the transmitter pulse is sent toward the earth, and backscatter data from the atmosphere returns approximately 9 milliseconds later as an approximate 300 microsecond pulse. This return pulse is processed as 20 atmospheric layers, each approximately 15 μs long in time. As shown in Figure 8-3, the signal from the detector is amplified, then mixed with a

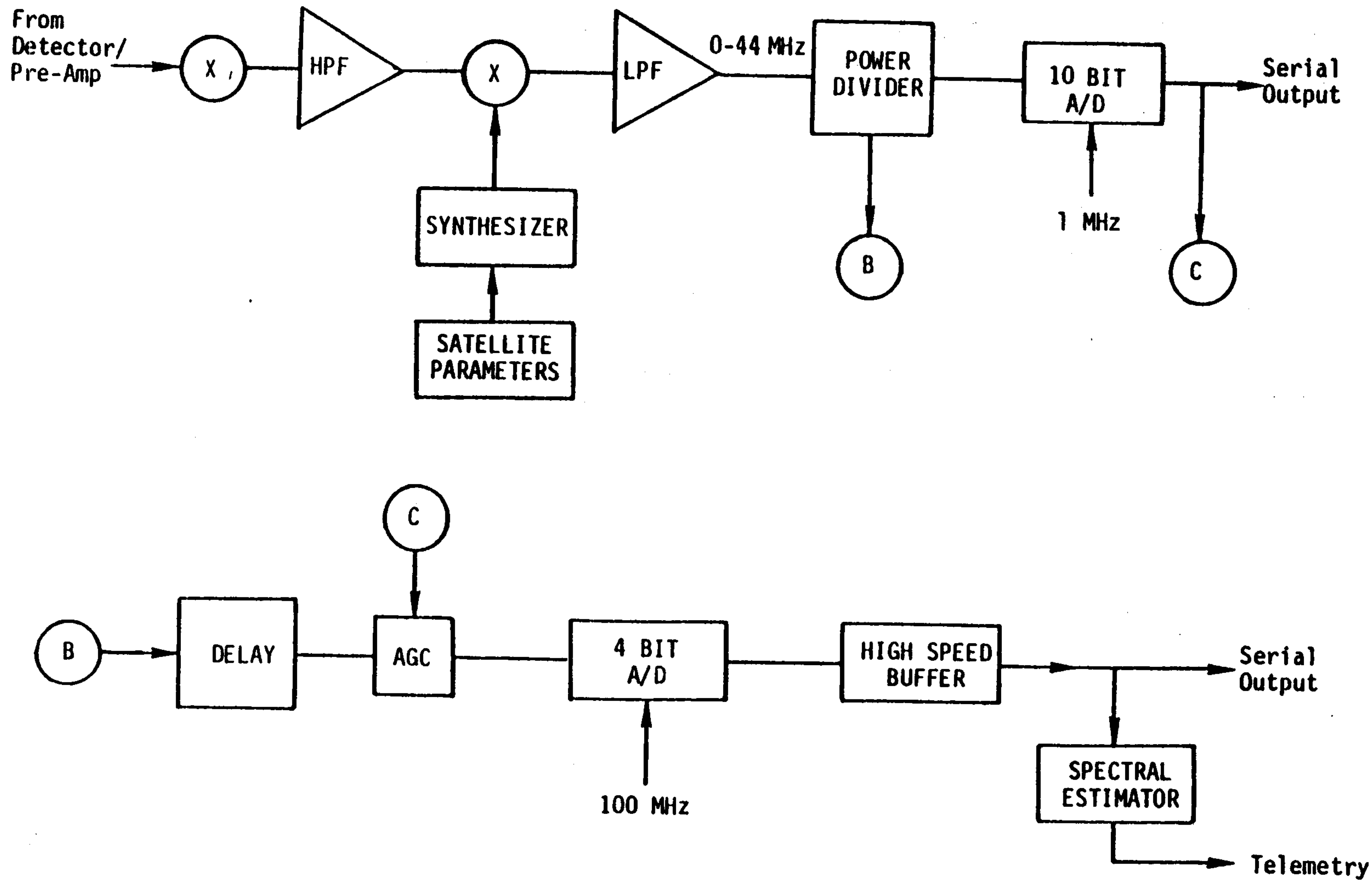


Figure 8-3. Signal Processor Block Diagram.

fixed frequency of several GHz to upconvert the baseband signal so that the $\pm$ 22 MHz of signal can ultimately be maintained within a 0 to 44 MHz baseband. The upper sideband from the mixer is selected by a High-Pass-Filter, and is then mixed with the output of a variable frequency synthesizer. The output of this synthesizer corrects for the background doppler effects of satellite motion and earth's rotation. The corrected doppler windsignal is then Low-Pass filtered to assure the baseband range of 0 to 44 MHz. The dynamic range of the backscattered return is approximately 30 dB, primarily because of backscatter coefficient variations, and must be digitized to 10 bits or more. To maintain 1 m/sec out of $\pm$ 100 m/sec, the determination of the mean frequency must be digitized to 8 bits or more, and the frequency spread about the mean requires digitizing to 6 or more bits. For 20 range cells, therefore, the minimum science data per pulse which must be output is 480 bits. Figure 8-3 shows how the concept in [1] was implemented to obtain the minimum data rate and reduce equipment requirements. The 0-44 MHz RF signal was power divided into two series of digitization states. The high quantization (10 bit) A/D is sampled at a low rate (1 MHz) to measure peak amplitude in a range cell. This information is used to control the variable gain amplifier in the second digitization stage and reduce the dynamic range requirements for the phase sampling digitizer. The input to the phase sampling digitizer is time delayed the equivalent of one range cell to allow setting the AGC prior to digitizing the data at the 100 MHz rate to 4 bits. Four (4) bits have been shown to be adequate for accurate spectral estimation [5] and the 100 MHz rate meets the Nyquist criteria for the 44 MHz analog bandwidth. The output of this digitizer must be stored in a high-speed buffer memory for post-processing. This buffer must have a storage capability of approximately 31,000 4 bit words for the total return signal. The concept in [1] then performed the spectral estimation using a Fast Fourier Transform (FFT) algorithm. If this were applied to the Space Station Lidar at a PRF of 10 Hz, the computer would have approximately 100 ms between pulses to complete this FFT. For the Space Station lidar, the system should probably output occasional samples of the raw data in addition to the compressed pre-processed data. This would require around 121 kbits/pulse, with the data sampling rate for the 300 microsecond data interval being around 410 MHz. If a laser repetition frequency of 10 Hz is assumed, the minimum (preprocessed) data rate would be about 10.0 kbps (assuming 4-5 kbps for housekeeping, etc.). The raw data output rates for the occasional samples would be approximately 1.22 Mbps to allow flexibility of data post-processing and research on data processing algorithms. The 4-5 kbps ancillary data rate includes not only the usual housekeeping type data, i.e., temperatures, voltages, etc., but also must provide the various measurements to determine telescope pointing, laser operating characteristics, and other lidar parameters on a pulse by pulse basis. Operation of the lidar would be anticipated for about 8 hours per 24 hour period when tests are underway, probably averaging 120 hours of operation over a 90 day period.

8.5 DATA ACQUISITON, PROCESSING AND CONTROL

As described in the preceding section, the raw data rates obtained from a doppler lidar are relatively high, i.e., approximately 121 kb per pulse, and require a high throughput consistent with a sampling rate of approximately 400 MHz during the 300 microsecond pulse return period. For most tests where the Space Station Lidar is operating as an autonomous unit, the output data rate to the Space Station may be as low as 10 kbps since the raw data is internally processed and displayed to onboard specialists who control the experiment. It is anticipated that the lidar will be outputting data for less than 20 minutes per orbit, and that operations will be limited to three orbits during a 24 hour period. The lidar data handling system will utilize available general purpose programmable electronic units to the maximum extent possible to provide flexibility of signal processing, data handling, and onboard display as well as data formatting for transmission through the Space Station. The lidar will be designed for control by Space Station personnel and therefore, will not require command and control through the Space Station data link. It is anticipated that a real-time voice and video link to the ground will be required during occasional tests so that ground scientists can support the

Space Station personnel who are performing the tests. The lidar experiment system will include data storage for several minutes of sustained operation when outputting raw data, approximately 10^9 bits. This storage will permit several hours operations with data preprocessing. Based on the information provided in the Space Station Reference Configuration Document - JSC 19989 [6], the outputs of the lidar facility appear compatible with the SS-TDRSS data links and should, therefore, not present a serious problem in view of the low lidar duty cycle.

8.6 CREW ACTIVITIES

The crew will be expected to integrate the lidar facility into the Space Station Laboratory, to set up and adjust the equipment for each experiment, and to operate the equipment and perform the tests as scheduled. Since a number of the experiments will be reasonably autonomous, the crew will also evaluate the performance of the equipment and make judgements on experiment success based on displays and recorded data. To carry out the experiments, it is anticipated that a minimum work crew of two will be required, with a third member assisting for some experiments. The crew should have a strong engineering background, and it is believed should be led by a professional supported by one or two highly qualified technicians. It is anticipated that flight data will be taken for approximately three orbits per day for 10 days per month requiring 15 manhours of crew time per 24 hour period and 150 manhours per month.

Initial setup of the lidar facility will require EVA activity in three phases. The first phase will require transporting about seven racks of equipment into a manned laboratory and mounting them. If it is assumed that approximately 2 hours of EVA for a 2-man team (4 manhours) is required to move and mount a single rack, then approximately 30 manhours of EVA will be required. The second phase of initial setup will cover the mounting and alignment of the 1.25 meter Telescope and Pedestal assembly. This phase is estimated to require approximately 8 hours EVA for a two-man team (16 manhours) assuming the pedestal was properly designed to be mounted in a module pressure port. The third-phase includes the transport and mounting of the laser transmitter on the pressurized (internal) side of the pedestal. The laser assembly will probably be modularized into at least 2 parts and is estimated to require another 8 hours for the two-man EVA team to transport and mount. After all equipments are in place, it is estimated that the two-man team will require approximately 40 hours of IVA (80 manhours) to complete the integration of the equipment into a working system and to debug and check out lidar operation. For normal operation of the facility, it is anticipated the crew consisting of one professional and one technician will devote approximately 1 hour to equipment start-up, warm-up and test/adjust ment prior to an experiment. The experiment will be conducted for approximately 7 hours on each of the flight test days. Approximately one hour will be required to power down and stow the equipment after each test. Thus, the total time elapsed for a three-orbit flight test will be approximately 9 hours. When the lidar configuraton is to be changed, the operating/servicing crew will be required to make changes to internal equipment (e.g., the laser subsystem) and therefore, may devote several hours per day to the work bench for several days. Other changes (e.g., Nadir viewing angle) will also require EVA for probably three hours to replace instrumentation modules and/or make telescope/ pedestal adjustments.

8.7 GROUND PROCESSING

The flight data obtained from this experiment will be used by the Langley experiment proposers/scientists to develop design criteria for advanced space lidars and to develop/refine analytical models of such lidars as a design/evaluation tool for future earth observation sensors. It is anticipated that other NASA centers and universities will also utilize data from the facility to support development work in components and systems.

8.8 SUPPORT EQUIPMENT

The primary support equipment required will be to test and service electronic components and to align optical systems. Equipment to provide cryogenic cooling of radiation detectors used as signal receivers will also be required, as will cooling provisions to maintain thermal equilibrium in the laser transmitter and electronic data processing systems.

8.9 SPACE STATION RESOURCE REQUIREMENTS

The requirements given in this section are not traceable to the design given in [1]. The referenced design was valuable in estimating telescope and pedestal requirements, however, it is not considered appropriate for the Space Station Lidar Facility because it was optimized in weight and power for a free flyer; and the light weight, single performance mode system is not suitable for a flexible research tool.

8.9.1 Weight

The Lidar System weight without a mounting isolation and pointing stabilization mount is estimated to be approximately 914 kg. as shown in Table 8-1. Of this total, 364 kg is the telescope and mounting pedestal, 232 kg is the laser transmitter, and the remaining 318 kg includes electronics components. In the event a stabilizing mount is required, an additional weight of 214 kg is predicted as shown in Table 8-2. Another 228 kg of supporting equiment is anticipated bringing the total potential payload weight to approximately 1600 kg.

8.9.2 Power

The total power required for the lidar system is estimated to be 2940 watts, of which 1900 watts is in the laser transmitter. An additional 160 watts would be required for the stabilization mount if it is needed and approximately 300 watts for test equipment, bringing the total <u>peak</u> payload power to 3400 watts.

Table 8-1. TDMX 2263: SPACE STATION RESOURCE REQUIREMENTS
CO_2 LIDAR WIND SENSOR

ELEMENT	WEIGHT (kg)	VOLUME (m^3)	POWER (WATTS)	THERMAL REJECT (WATTS)
Telescope/Pedestal	364	14.9*	70	40
Laser Subsystem	232	0.5	1900	1880
Signal Processor	61	0.2	250	250
Data Processor	104	0.3	300	300
Data Storage	36	0.2	100	100
Displays	55	0.2	200	200
Command/Control	28	0.1	20	20
Supporting Equipment	34	0.1	100	80
Total Lidar System	914	16.5	2940 watts	2870 watts

*This includes swept volume as telescope is conically scanned 360^o in Azimuth.

Table 8-2. TDMX 2263: SPACE STATION RESOURCE REQUIREMENTS ISOLATION/STABILIZATION MOUNTS

ELEMENT	WEIGHT (kg)	VOLUME (m^3)	POWER (WATTS)	THERMAL REJECT (WATTS)
Auxiliary Equipment				
• Isolation/Stabilization Mount	200	0.22	150	150
• Star Sensor	14	0.04	10	10
Total Aux. Equipment	214	0.24	160	160
Supporting Test Equipment	228	0.33	300	300
Lidar System	914	16.5	2940	2870
Mounting Racks and Cold Plates	228	0.08	--	--
Payload Totals	1584 kg	17.4m^3	3.4 kW	3.33 kW

8.9.3 Volume

The total volume required for the lidar was estimated to be approximately 16.5 m^3. Most of this volume (14.9 m^3) goes for the telescope and pedestal and includes the cylindrical volume swept out as the telescope is scanned 360^o in Azimuth while pointing to a nadir angle of 55^o. The remaining 1.6 m^3 includes the laser, electronics, and support equipment which are internally mounted within a module. The module containing the lidar should provide an additional working volume approximately 2 meters wide by 3 meters high by 4 meters in length to permit repairs, testing, and occasional configuration changes.

8.9.4 Thermal Requirements

The thermal rejection requirements are estimated to maintain the telescope, optics, and laser subsystems at approximately 330 K, and the heterodyne detector at approximately 77 to 100 K during normal operations. The maximum thermal rejection is estimated at 3400 watts with the majority (around 1880 watts) required to maintain the laser subsystem package at 330 K. This is planned as a liquid coolant loop internal to the lidar system which rejects heat to the Space Station. Most of the remainder (around 1500 watts) is distributed among electronic components in the signal and data processing and display subsystems and power supplies and in system test equipment. The electronics will require a separate cooling loop rejecting heat to the Space Station.

The thermal rejection required under normal operating conditions will be approximately 2.8 kW for periods approaching 9 hours. The daily thermal rejection required under operating conditions will be approximately 38 kW hrs. Based on the Space Station Reference Configuration Description [5], it did not seem reasonable to plan on radiative thermal rejection of this amount of energy.

8.10 INTER-EXPERIMENT COMPATIBILITY

The CO_2 Doppler Lidar Wind Sensor experiment is closely allied to the Sensor System Technology Laboratory (TDMX 2261) and will, if properly designed, contribute to the Sensor System Technology Lab by providing the Optical Receiver/Transmitter which would be considered a part of the laboratory growth version. The Wind Lidar can gain from the laboratory also, by utilizing some of the laboratory's test equipment and possibly some of the laboratory's data processing capability. The pointing control, stability, and determination requirements for both the laboratory and lidar are very similar also.

8.11 REFERENCES

1. Feasibility Study of a Windsat Free Flyer, Final Report - NOAA Contract No. NA82RAC00141 RCA Government Systems Division, Astro Electronics Contract NA82RAC00141 with NOAA-WPL, July 1983.

2. Global Wind Measuring Satellite System-Windsat, Final Report, NOAA Contract No. NA79RSC00127, Lockheed Missiles and Space Co., Report No. LMSC-D767868, April 1981.

3. Hess, R.V., Brockman, P., Schryer, D.R., Miller, I.M., Bair, C.H., Sidney, B.D., Wood, G.M., Upchurch, B.T., Brown, K.G.: Technology Assessment of High Pulse Energy CO_2 Lasers for Remote Sensing from Satellites, NASA TM 86415, April 1985.

4. Feasibility Assessment: Satellite Doppler Lidar Wind Measuring System, NASA George C. Marshall Space Flight Center, Report No. MSFC-MOSD-146, April 1982.

5. Martin, S.C., "Low-Order Digitization of Coherent Lidar Signals" Topical Meeting on Coherent Laser Radar for Atmospheric Sensing, Optical Society of America, WA6, (1980).

9.0 TDMX 2264 - MICROWAVE REMOTE SENSING-PASSIVE

CONTACT: Bruce M. Kendall/Dr. Richard F. Harrington, MS 288/490
NASA LaRC (804) 865-4834/3631

9.1 GOALS AND OBJECTIVE

The objective is the development and evaluation of smart sensor technology for passive microwave remote measurement of various parameters of the Earth's surface. The parameters which appear to be suitable for remote sensing with microwave radiometers include sea state, sea surface temperature and salinity, ice, soil moisture, snow cover, rain, and atmospheric moisture. The remote sensing of these parameters requires the measurement of radiometric brightness temperature at more than one frequency and, in some cases (i.e., soil moisture), requires measurements at relatively low frequencies (i.e., 1.4 GHz). To be of significant value, these measurements must be obtained: 1) with a spatial resolution down to or better than 10 Km, 2) with a radiometric temperature resolution of better than 1 Kelvin, 3) with significant amount (approx. 80%) of coverage over the Earth temperate zone, and 4) with a total response time of data to the potential user of a few days.

Specifically, the objective of the initial experiment will be to develop and evaluate the technology required for an operational system to measure the radiometric brightness temperature at 1.4 GHz. This experiment will evaluate an antenna of required aperture to obtain the desired spatial resolution at the lowest frequency (largest required aperture). The antenna will have multiple beams in order to obtain the desired coverage and temperature resolution. The experiment also includes a compact set of space qualified, feed mounted radiometers, as well as data processing techniques and algorithms to optimize system performance over a range of target characteristics.

9.2 DESCRIPTION

The experiment consists of a space erectable, 118 meter diameter antenna with 3 subapertures and microwave reflecting surface. The antenna has a 52 beam, low loss, microstrip feed assembly and a set of compact, feed mounted, radiometer subassemblies. The antenna beams will be pointed by an antenna control to approximately 1.0 degree with a measurement system accuracy of 0.1 degrees, however, the antenna requires a clear field of view of approximately ± 20 degrees relative to the Earth near nadir for data taking and, periodically, in the anti-Earth direction for calibration purposes. As part of this experiment, the Space Station will carry a programmable, adaptive data processing and display system to allow optimization of spatial and temperature resolution as a function of the expected and observed earth surface characteristics.

The antenna will be deployed/erected and special instrumentation then employed to evaluate antenna shape and reflecting surface characteristics. When the antenna has been adjusted to the required shape and the feed and radiometer assemblies are in place, the antenna will be pointed to space for calibration, and to previously selected Earth target areas which contain ground monitoring and/or radiating equipment. After the microwave and pointing systems have been adjusted and the performance verified, other earth target areas will be observed to evaluate remote sensor effectiveness to measure the desired geophysical parameter(s). An infra-red sensor boresighted with the antenna will collect correlative data on the target areas, which will be merged with additional correlative data from ground and air borne instrumentation systems so that remote sensor performance can be determined. The initial antenna deployment and final adjustment of the microwave system will require EVA. A Space Station mission specialist will be involved in realtime monitoring of all data, and in the adjustment/modification of the signal and data processing algorithms. The data obtained from the system observations will therefore be processed and displayed onboard in addition to stored for transmission to Earth through the Space Station data-link. Reorientation of the antenna will be required for alignment and tracking of the target areas, and periodically for space calibration of the system (anti-Earth alignment).

While evaluation of system operating parameters will be done onboard, determination of remote sensing performance will be performed on Earth, using data from Earth target sites in addition to Space Station flight data.

The experiment consists of a number of subsystems which require research and development prior to flight. One of the more awesome technology drivers is the light-weight 118 meter diameter space-erectable deployable antenna required to achieve the desired spatial footprint. The fundamental design approach, including materials for such an antenna, has been addressed by a number of studies, however, without any clear solution. The final adjustment of the antenna structure and surface to achieve an acceptable "optical figure" for the reflector will require a comprehensive instrumentation/control system. While preceding studies have addressed sensor approaches to determine antenna figure, and other studies have considered distributed control techniques to adjust lightweight structures; a comprehensive systems approach for an antenna of this size has not been defined and analyzed.

Another technology driver for the microwave remote sensor will be the antenna pointing and control system. The large, relatively flexible antenna structure will probably require that the pointing system be integrated with the antenna figure control system, however, the pointing system must provide large angular momentums to control beam direction. The array of compact radiometers and the microwave feed for the antenna will also require significant development prior to experiment implementation. Research has been done and is continuing at a relatively low level, on radiometers and circuits for the required frequency range. Work is also underway for low-loss, micro-strip feeds for antennas of this type.

Experimental determination of the electro-magnetic characteristics of the deployable reflector mesh are required as are analyses to determine the impact of these measurements on the radiometer performance.

9.3 ENVIRONMENT

A circular orbit is desired for this experiment to obtain uniform spatial and brightness temperature resolution over the Earth. With a given bandwidth available for measurement (WARC 79 indicates 27 MHz at 1.4 GHz), and a given spatial resolution requirement (10 km); the orbital altitude determines the brightness temperature resolution and antenna beamwidth (hence antenna aperture size) required. It also affects the swath width (i.e., number of beams) required for a given percent of Earth coverage, and the revisit time. The selected orbital altitude is approximately 678 km, yielding a required aperture diameter of approximately 55 m and a temperature resolution of approximately 0.5 K for a spatial resolution of approximately 10 km. Earth coverage of approximately 86% at this altitude results in a requirement for 52 beams. They are obtained with three, offset-fed, subapertures within a 118 m diameter reflector. The primary effect of increased orbital altitude would be improved temperature resolution with increased antenna diameter while a decreased orbital altitude would decrease the required antenna diameter but would "degrade" the obtainable temperature resolution. The selection of orbit inclination is primarily dependent on the maximum latitude over which coverage is desired. An inclination of from 45 to 60 degrees should be sufficient. For conducting the Earth observation experiment, the required antenna orientation is with the boresight axis aligned to local vertical to an accuracy of approximately 1.0 degree. For the analysis of experiment data, however, knowledge of the angular position of the boresight axis is required to an accuracy of approximately 0.1 degree. For radiometer calibration purposes, it is desirable to periodically (every 30 days) align the boresight axis in an anti-Earth direction. This requirement is to look at a cold sky, hence the tolerance of alignment is not critical with the exception of avoiding hot sources. Calibration could be conducted during Earth shadowing of the sun to avoid its radiation.

Due to the size of the antenna required for this experiment, it is difficult to envision a placement on Space Station that does not seriously interfere with Space Station activities. Since this is an Earth observation experiment, the most logical placement would be at the bottom of Space Station (Earth directed, see Fig. 9-1(a)). This would not significantly interfere with solar panel operation and TDRSS communications, however, it would interfere with Shuttle docking and OTV/OMV activities in addition to eliminating the possibility of any other Earth observation equipment. Figure 9-1(b) shows the only possible site for an antenna of this size to be located with minimal interference to the operation of SS solar and TDRSS communications systems and Shuttle docking; however, it will interfere with the SS proximity operations communications and possibly with OTV/OMV activities in addition to limiting other Earth observations experiments. It would also require reorienting SS attitude for earth observations and sky calibration. Another significant problem presented by this large a structure installed on SS is the effect on Space Station control systems due to the large antenna mass and moment-of-inertia. For these reasons, it is recommended that this experiment be placed on a co-orbiting platform. This is illustrated in Figure 9-2 using a dedicated version of the Core Platform. This will require that the systems for antenna deployment and reflecting surface evaluation and alignment, as well as the experiment operation, be designed for remote operation from Space Station. It would be desirable to conduct the initial antenna deployment and evaluation in proximity operations or control zone for purposes of observation and access. The platform could then be transferred to co-orbiting zone.

9.4 SENSORS

The basic sensor for this experiment is a 1.4 GHz, 52 beam, pushbroom radiometer employing a 118 meter diameter hoop-column antenna for soil moisture measurements. An Infra-red radiometer will also be employed for target thermal temperature inputs to the data. Also included is a data recorder and an adaptive data processor (computer) and display system. A

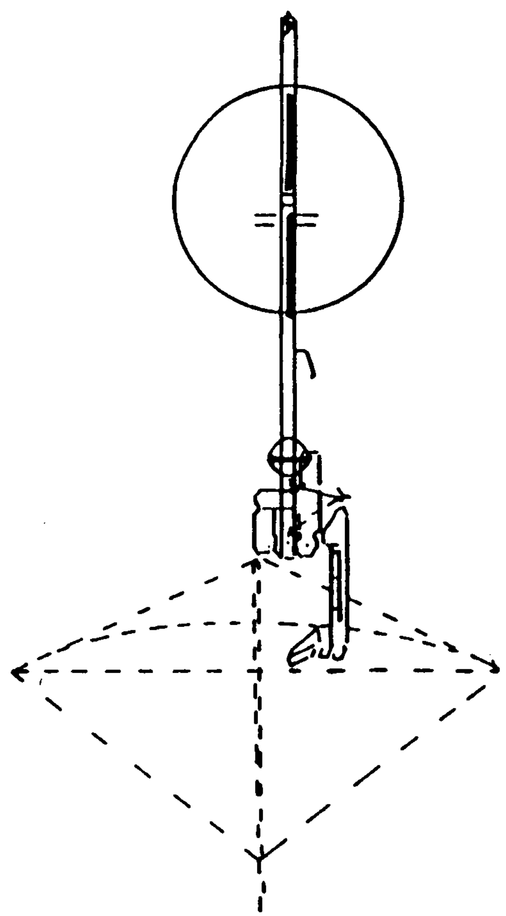


Figure 9-1(a). Desired Antenna Location.

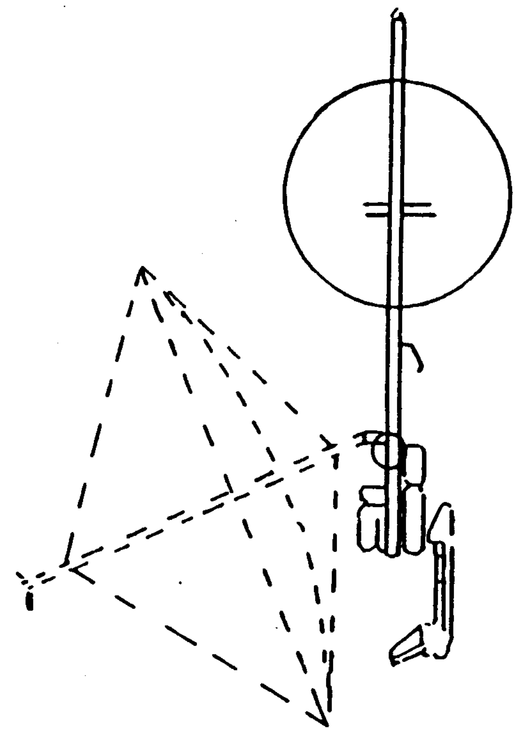


Figure 9-1(b). Probable Antenna Location.

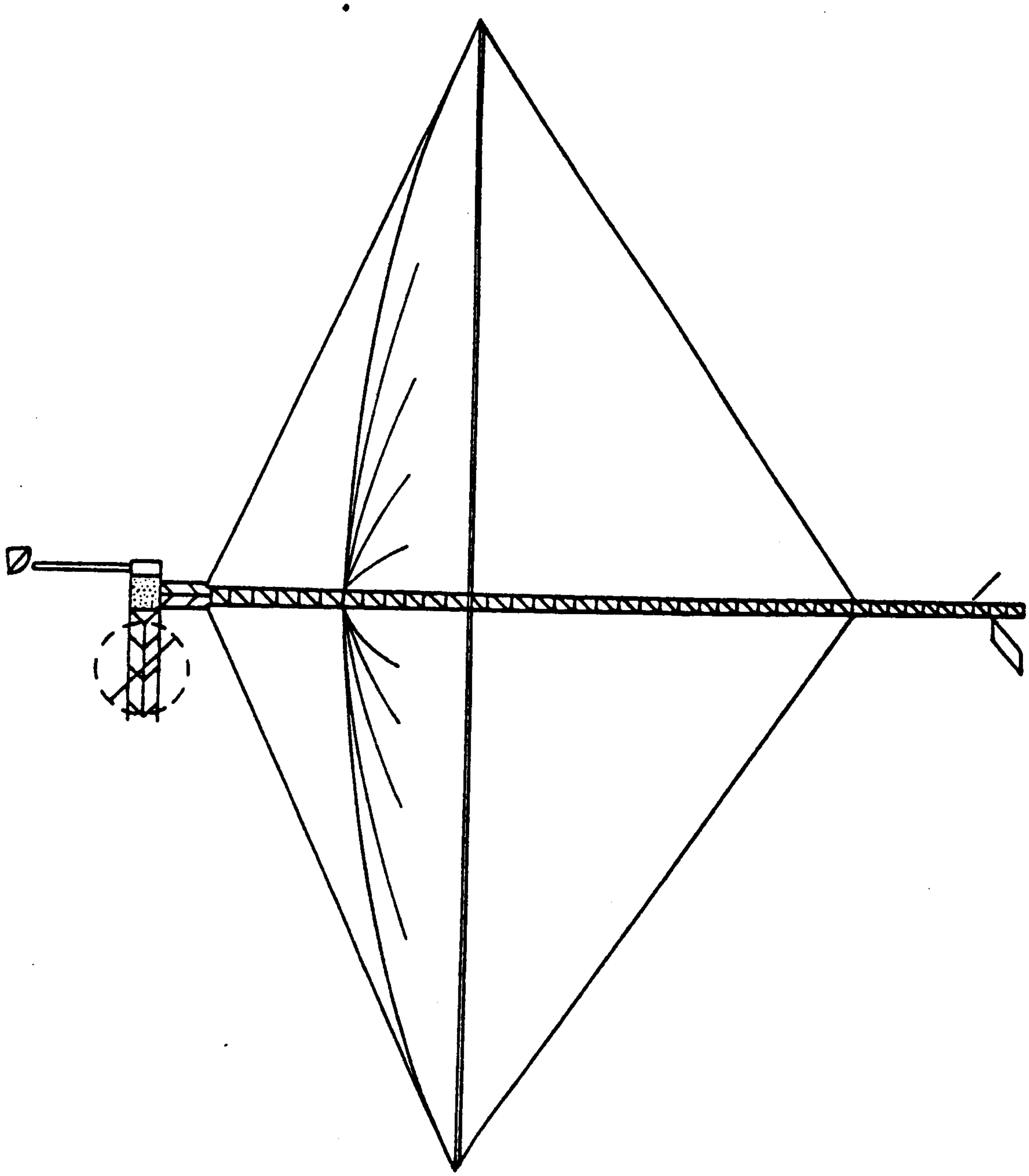


Figure 9-2. Co-Orbiting Platform Location.

major portion of the experiment, however, will involve the deployment, adjustment, and evaluation of the reflecting surface of the antenna. This will require some optical and/or laser equipment for measurement of the antenna surface contour and feed locations and stability and is proposed by NASA Marshall Space Flight Center and JPL.

The 118 m hoop-column antenna consists of a deployable mesh reflecting surface suspended within a hoop and tensioned to the desired surface contour by a supporting cable system attached to the column with the desired feed systems attached to the column at the focal points. It employs three, offset-fed, parabolic subapertures as illustrated in Figure 9-3 with an average subaperture diameter of approximately 55 m and F/D = 1.1 to 1.35. Subaperture #1 is used to generate 18 beams in the center of the swath (+/- 6.75 deg. from nadir) and subapertures #2 and #3 are used to generate 17 beams on both the left and right sides of the swath (+/- 6.75 to 19.5 deg. from nadir). The individual beams have a nominal beamwidth (98% power) of 0.686 degrees. The feed assemblies include the required 52 radiometer subassemblies. The pre-deployment packaged assembly is approximately 14.63 m long and 4.57 m diameter with a weight of 4000 kg, which is compatible with the STS transportation restraints. This sensor will provide for the mapping of the radiometric brightness temperature, at 1.4 GHz, of sections of the earth as it passes overhead. The swath width from a single pass is approximately 455 km with a spatial resolution of approximately 9 km cross-track and 10 km along-track (1.5 seconds integration time). An input dynamic range of at least 300 K, and resolution of 0.05 K or better, is desired. Figure 9-4 is a basic block diagram indicating the major subsystems.

The IR radiometer should scan the same swath width (+/- 20 degrees) with approximately the same spatial resolution as the microwave radiometer. An input dynamic range of approximately 100 K is required with a temperature resolution of approximately 0.1 K.

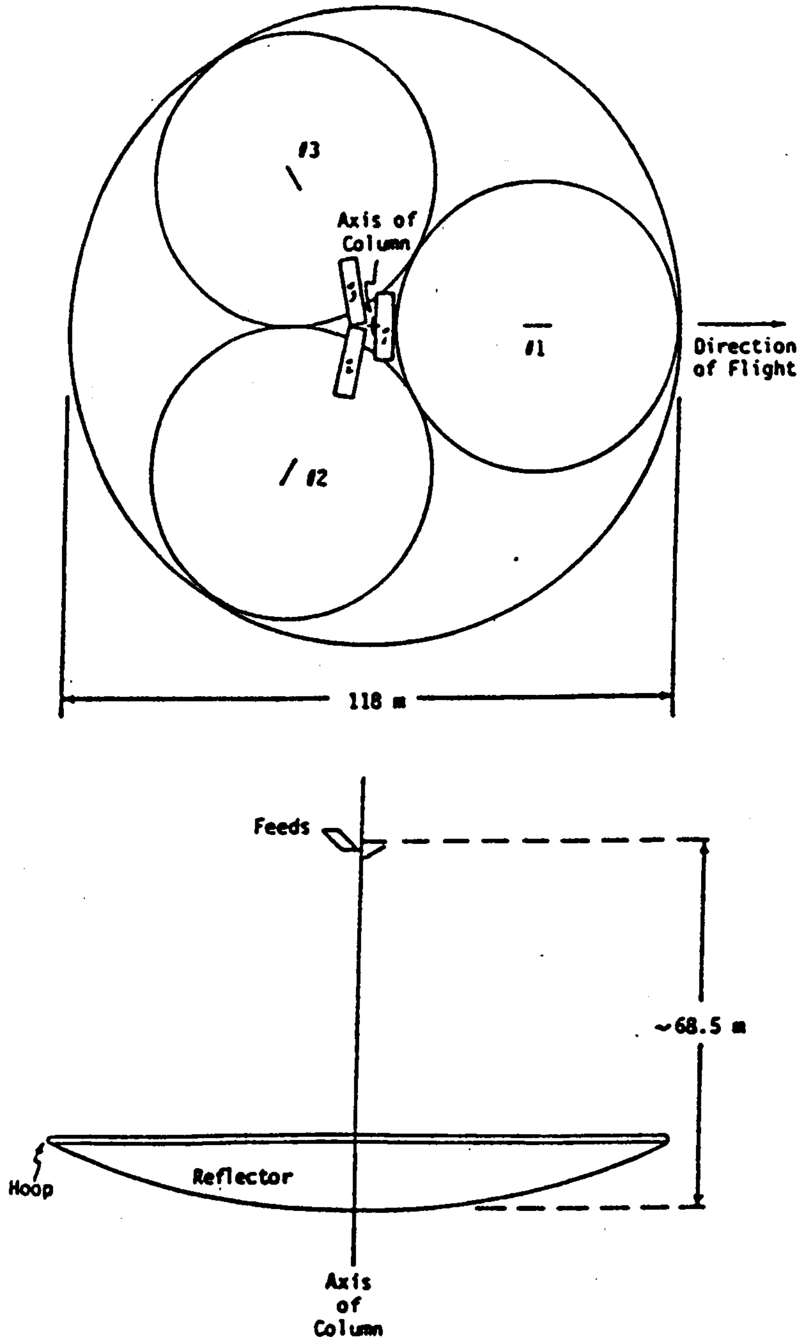


Figure 9-3. Three Sub-Antenna Antenna Configuration.

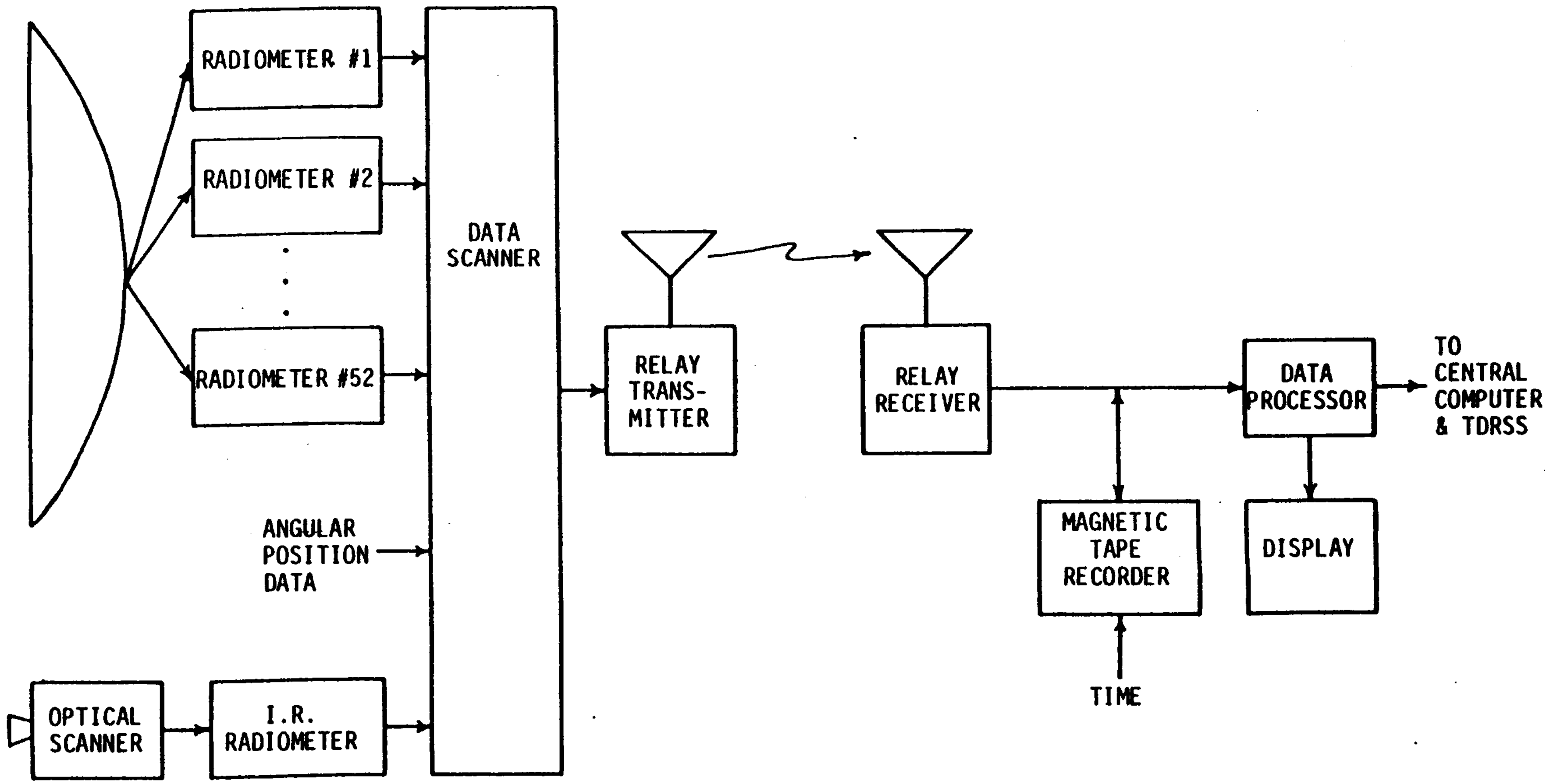


Figure 9-4. System Block Diagram.

9.5 DATA ACQUISITION AND CONTROL

Data acquisition and control could be accomplished directly from the ground or through a communications link with SS. The first approach is desired since it eliminates the requirement for SS personnel and systems to handle the daily operations of the experiment. However, in conducting the first flight experiment, the second approach is desired to allow SS personnel to monitor instrument performance and control data processing parameters. This mode of operation is illustrated in Figure 9-5, where a relay link is used to transfer data, in real time, to SS for recording and processing.

The basic radiometric (microwave and IR) data from the sensor consists of 104 data samples every 1.5 seconds. Assuming that the minimum spatial resolution that might be desired from the sensors is approximately 200 km (i.e. over ocean), a single frame of data could consist of 30 seconds of data samples. This is also a suitable period of time for updating the ancillary data (time, boresight angles, instrument parameters, etc). One frame of data, therefore, would be as follows:

ITEM	NUMBER OF BITS
microwave brightness temperature	13520
IR brightness temp.	10400
instrument parameters	2000
boresight angles	24
SUBTOTAL =	25944
sync, parity, etc. (10%)	2594
TOTAL =	28538

This results in approximately 28k bits of data per frame (30 seconds), for a maximum data rate of approximately 1 kbps.

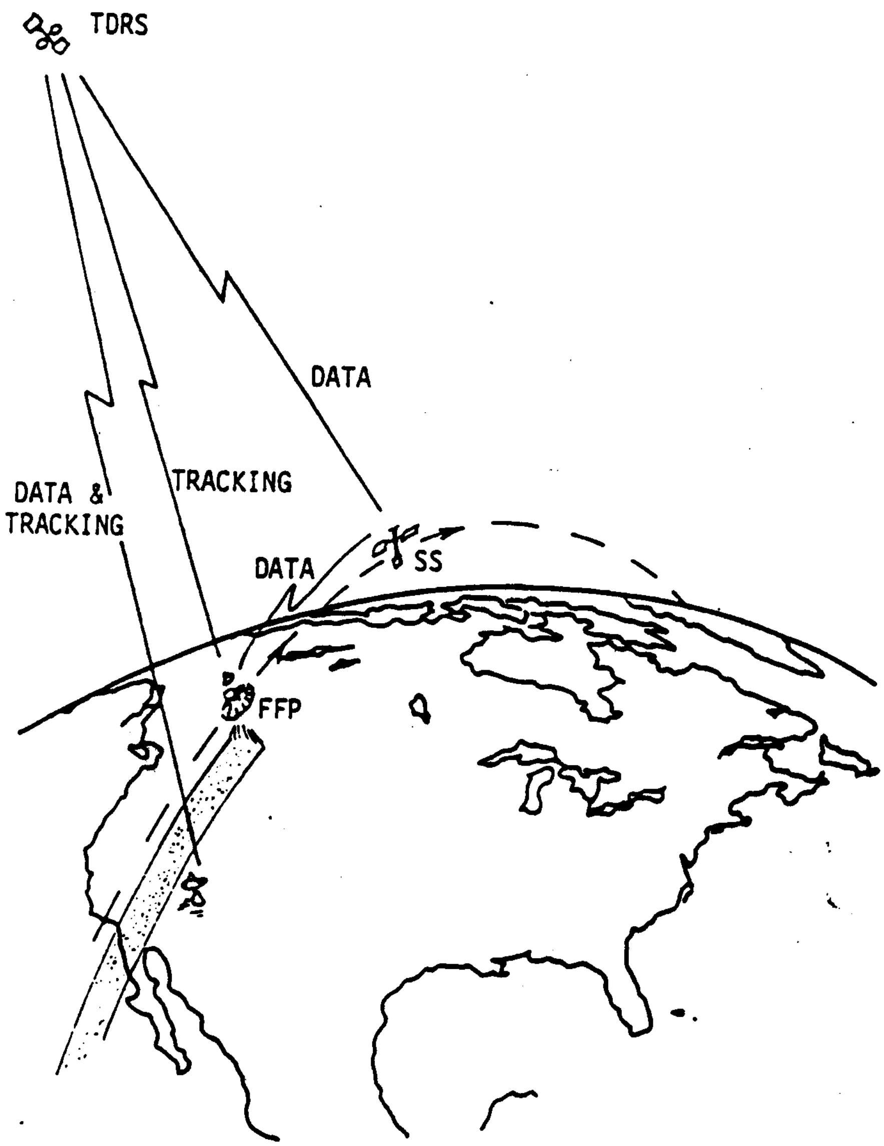


Figure 9-5. Data Acquisition and Control.

For purposes of technology evaluation, the primary data to be collected consists of measurements over areas with significant ground truth information available. This will be, to a great extent, data acquired over North America; hence, a maximum of only about 10% of the total possible data would need to be transmitted to the ground for analysis. The total amount of data available is 86 Mbits per day. The basic instrument data would be relayed directly to SS over a low data rate (1.0 kbps) link and recorded for preliminary analysis, editing, and further processing. An on- board recorder capable of storing at least 258 Mbits (3 days) of data would be required. Editing and data processing on-board SS would reduce the amount of data to be transferred to the ground to a maximum of 9 M bit per day.

Instrument control (i.e., antenna boresight angles, run times) would be conducted from SS through a command/control link to the platform.

9.6 DATA COMMUNICATIONS AND HANDLING

Data from the Platform will be relayed to SS for continuous recording on a dedicated magnetic tape recorder. This data would be reviewed daily for verification of instrument performance, editing and processing and would be transferred to the central SS computer in preparation for transfer to ground. A dedicated data processor and display system would be used for this. The required data capacity in the central computer for data dump would be approximately 30 Mbits.

9.7 TELEMETRY

This experiment will require a dedicated data link (1.0 kbps) from the Platform to SS, and a command/control link from SS to the Platform. It will also require the use of Platform communications through TDRSS with the ground for orbital tracking and platform command control. Data dumps from SS to the ground would be accomplished on the S-band or Ku-band return link and could occur daily or possibly on a 3-day cycle (revisit time).

9.8 CREW ACTIVITIES

The primary crew activities are associated with the transfer of the experiment package from STS to the Core Platform and the subsequent deployment and evaluation of the antenna and transfer to a co-planer orbit. This phase of the overall effort is proposed in other Technology Experiments. Recovery of the antenna is not anticipated. The crew activities for this experiment consist of the monitoring of instrument performance, and the editing and processing of data for transmission to ground. This will be accomplished on a daily basis and will probably required from 2 to 4 man-hours per day. A timeline for daily operations is shown in Table 9-1.

9.9 GROUND PROCESSING

Data from the entire experiment (approximately 3.5 Gbits) would be stored on the ground. Data processing would be required to support the Experiment Group on a daily basis. Data output would be required in both a numerical form and in the form of 3-D mapping. Support would also be required for collating of ground truth data and error analysis. The results of the daily effort in data analysis would feed back to SS for the programming of the experiment.

9.10 SUPPORT EQUIPMENT

The primary requirement for this experiment is the development of a Platform suitable for the deployment and control of very large, lightweight structures such as this antenna. This, of course, includes the development of SS facilities such as Platform docking, payload transfers, and proximity operations (EVA, observations proximity communications, etc). In addition, a dedicated data relay and command/control link between SS and the Platform will be required.

Table 9-1. DAILY OPERATIONS TIMELINES

FUNCTION/TASK	TIME (MANHOURS)	METHOD	FEASIBILITY	COMMENTS
Monitor instrument parameters	0.5	IVA	2	
Review data	2.0	IVA	2	
Program instrument for data acquisition	0.5	IVA	2	

*1-Proven, 2-Easy, 3-Hard, 4-Questionable, 5-Not Possible

9.11 THERMAL, POWER, WEIGHT, AND VOLUME REQUIREMENTS

The pre-deployment package is estimated at 14.63 m long and 4.57 m in diameter with a weight of 4000 kg. The operating power requirement for the radiometer package is estimated at 0.5 kW with a standby power of 0.03 kW. The data recording, processing, and display unit is estimated as 1 X 1 X 0.5 m in size with a weight of 100 kg and an operating power requirement of 0.3 kW.

9.12 INTEREXPERIMENT COMPATIBILITY

This experiment is dependent of the successful completion of a large space antenna deployment and evaluation (i.e., TDMX 2064).

10.0 TDMX 2265 - SATELLITE DOPPLER METEOROLOGICAL RADAR

CONTACT: Dr. Leo D. Staton, MS 490
NASA LaRC (804) 865-3631

10.1 GOALS AND OBJECTIVES

The primary objective is the development of enabling technology for satellite-based, meteorological radars providing three-dimensional mapping of rainfall rates and two-dimensional mapping of ocean-surface wind vectors. This technology must be compatible with the requirement for high spatial resolution (on the order of 1 km) and large area coverage (several hundreds of km).

10.2 DESCRIPTION

This experiment will consist of the development and evaluation of coherent, pulsed radar and data processing techniques for the detection of the desired parameters. Frequency of operation will be optimized for rain (approximately 10 GHz). Multi-beam, planar array antennas will be employed with a maximum size of approximately 13 by 31 meters. The signal will be transmitted through a single, conical fan beam antenna designed to provide approximately 45 degree earth incidence angle, and received through a multi beam antenna. High spatial resolution will be obtained in azimuth and elevation with beam directivity. Doppler processing techniques will also be evaluated for obtaining high spatial resolution in the azimuth and elevation.

Tests will be conducted observing selected earth target areas. This will require the Space Station to be oriented toward Earth. On-board data processing will be employed, with the processing algorithms being varied to evaluate their effectiveness.

Technology driven aspects of the experiment include the design and manufacture of the antenna to derive both the required spatial resolution and low side-lobe levels to suppress ground clutter. Further, the antenna will be mounted on the Space Station with two degrees of freedom with an antenna stabilization system to maintain the major antenna axis to within 0.2 degrees of the local vertical.

The experiment will require a medium-to-large external structure with unobstructed view of the Earth. This could contend with other experiments for location of the Space Station. Also, this experiment was originally envisioned to be flown as a shuttle mission. Special attention need be given when rescaling to the Space Station environment.

10.3 ENVIRONMENT

A circular orbit is desired for this experiment to maintain uniform spatial resolution and coverage over the Earth. An orbital altitude of 500 km was assumed in the conceptual design; however, a lower orbital altitude would be desired since it reduces the size (increased beamwidths) of the required antenna making the experiment more feasible. The lower limit on orbital altitude is determined by the minimum swath width which is acceptable. Assuming a swath width of 600 km is acceptable, results in a minimum altitude of approximately 350 km. An orbit inclination providing significant coverage over the contiguous U.S. (i.e., 50 degrees) is satisfactory.

The required antenna for this experiment is a planar array with a vertical dimension of up to 31 m and a horizontal dimension of up to 13 m. The required orientation is with the vertical axis aligned to local vertical and the angle between the plane of the array and the velocity vector maintained at 71.5 degrees. The tolerance on the alignment of the vertical axis is critical (for maintaining acceptable ground clutter) and should be maintained at better than 0.2 degrees with knowledge of the angle of better than 0.1 degree. A separate antenna control system with external sensors will be required. The tolerance on the alignment of the plane of the array in azimuth is not critical (i.e., 1-2 degrees); however, knowledge of this angle is required to an accuracy of better than 0.1 degree.

The clear field of view required for this antenna is at an elevation angle of -45 degrees over an azimuth angle of approximately +/-25 degrees relative to a line normal to the plane of the array.

Location of the antenna on SS which should not interfere with SS systems operation and should have a clear field of view in the desired direction toward Earth. The antenna will have to be transported to SS in sections and assembled in position.

10.4 SENSORS

The sensor proposed for this experiment is a multibeam (pushbroom), bi-static, 10 Ghz radar to provide three-dimensional maps of the precipitation distribution above land masses and over the ocean. In addition, it will provide two-dimensional maps of the radar reflectivity of the Earth's surface. It was originally proposed as a Shuttle mission and requires significant changes in the conceptual design for application in a Space Station environment.

Figure 10-1 is a block diagram of the proposed radar and Figure 10-2 illustrates the mode of operation. The signal is transmitted through a transmit array with a pattern consisting of a sector of a cone, with the axis of the cone aligned normal to the Earth's surface. The return signal is received on the multibeam receive array with patterns consisting of a narrow (pencil beams) sectors of the same cone. Time gating in the receiver separates the return from various sections (cells) of the atmosphere as the signal progresses towards the ground. Doppler processing may also be used to provide increased spatial resolution. The ground swath required determines the transmit azimuth beamwidth (ϕa), the nadir angle (ϕn), and the number of receive beams required (for a given spatial resolution). Current estimates for the system requirements with respect to spatial resolution and coverage are as follows:

Vertical resolution = $<$1.0 km
Horizontal resolution = approximately 2 km
Swath width = $>$600 km

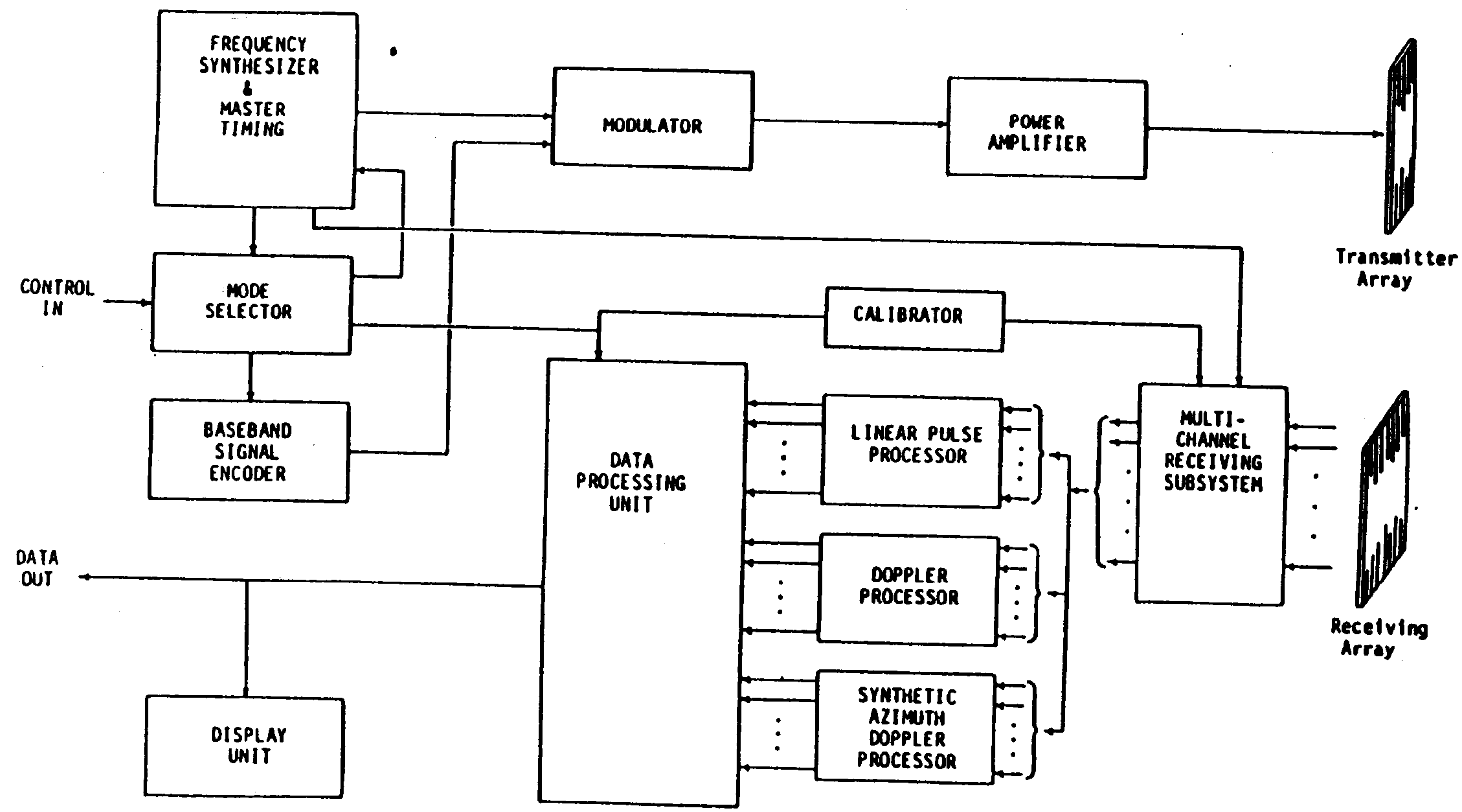


Figure 10-1. Multi-Mode, Bi-Static, Meteorological Radar.

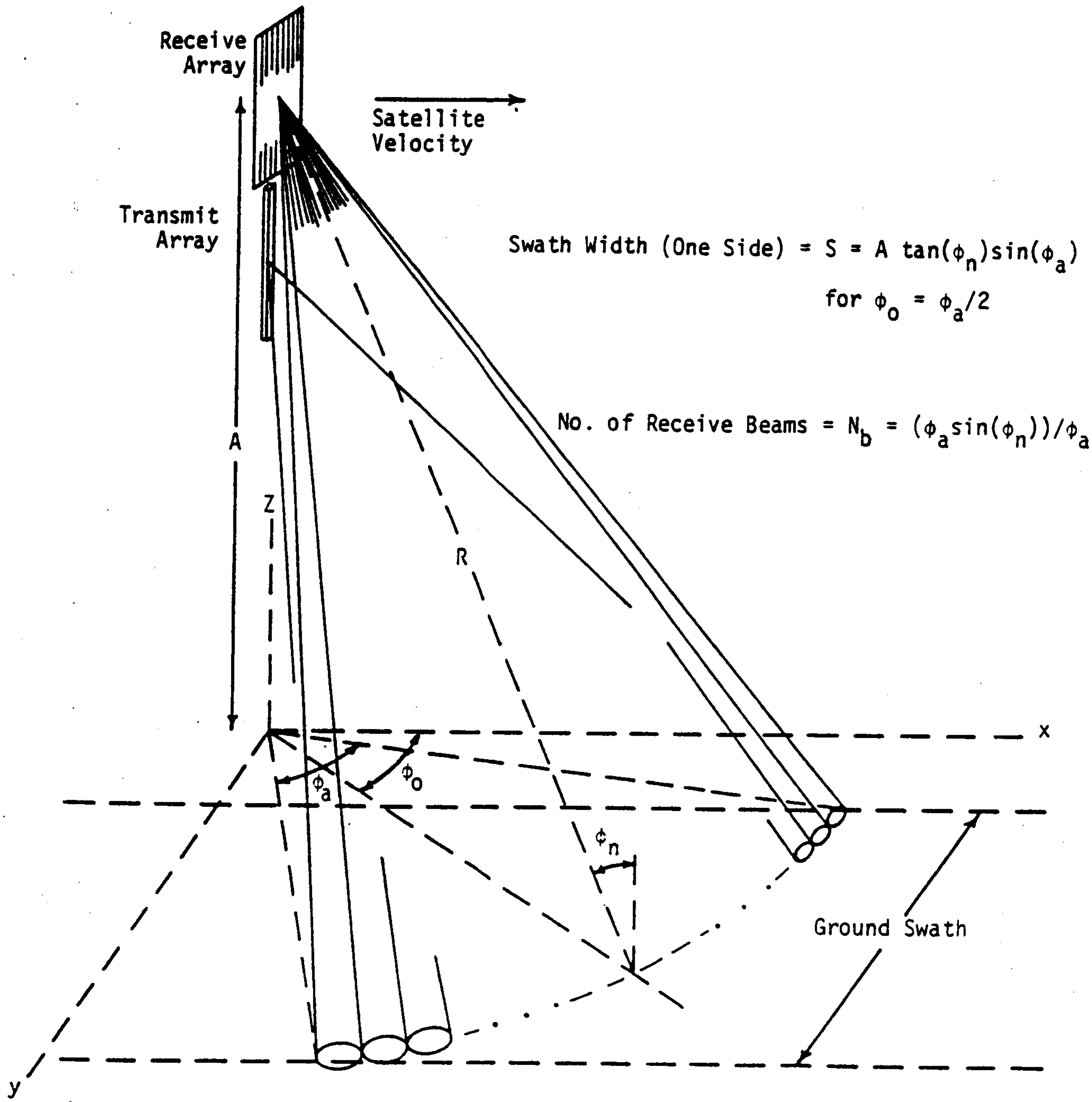


Figure 10-2. Experiment Geometry.

Since an operational system would employ antennas looking both fore and aft, generating a swath on either side of the subsatellite track; a swath width of 300 km with only a forward (or aft) looking antenna may be satisfactory for the initial experiment.

The original proposed (for Shuttle) receive array design is as illustrated in Figure 10-3, and consisted of a horizontal array of 185 vertical slotted waveguide arrays. For the Space Station experiment, this basic antenna design must be modified to meet the previously listed requirements from an altitude of 500 km. For a swath width of 300 km, and azimuth beamwidth of 37 degrees is required assuming $\phi_o = \phi_a/2$ and $\phi n = 45$ degrees (see Figure 10-2). Assuming a required sidelobe level on the order of -20dB results in the horizontal dimension of the transmit array of approximately 0.056 m. This would consist of a horizontal array of two vertical waveguide arrays. The primary problem in the design of both the transmit and receive arrays is that of obtaining the required resolution in the patterns in the elevation plane. This is illustrated in Figure 10-4. It is important that both the transmit beamwidth (ϕe) and the receive beamwidth (θe) be the same and that the sidelobe levels in the nadir direction ($\phi n < 45$) be very low. This is to maintain an acceptable ratio of signal to ground clutter when sampling the atmosphere. Assuming that the integration time (and pulse duration) can be maintained small enough to be considered a negligible contributor to the vertical resolution (Cz), the maximum elevation beamwidth (θe, ϕe) that can be used is 0.11 degrees. Assuming a required sidelobe level on the order of -30dB results in a vertical array dimension of approximately 31 m. The azimuth beam width required for a single beam of the receive array (ϕa) is 0.16 degrees, and 163 beams are developed in the array processor. Assuming a required sidelobe level on the order of -20dB results in a horizontal receive array dimension of approximately 13 m. The resultant transmit and receive antennas are as illustrated in Figure 10-5. If it is assumed that a thin wall (0.5 mm) aluminum waveguide can be used, the weight of the waveguide in the receive array is approximately 1700 kg. Considering the requirement for a support

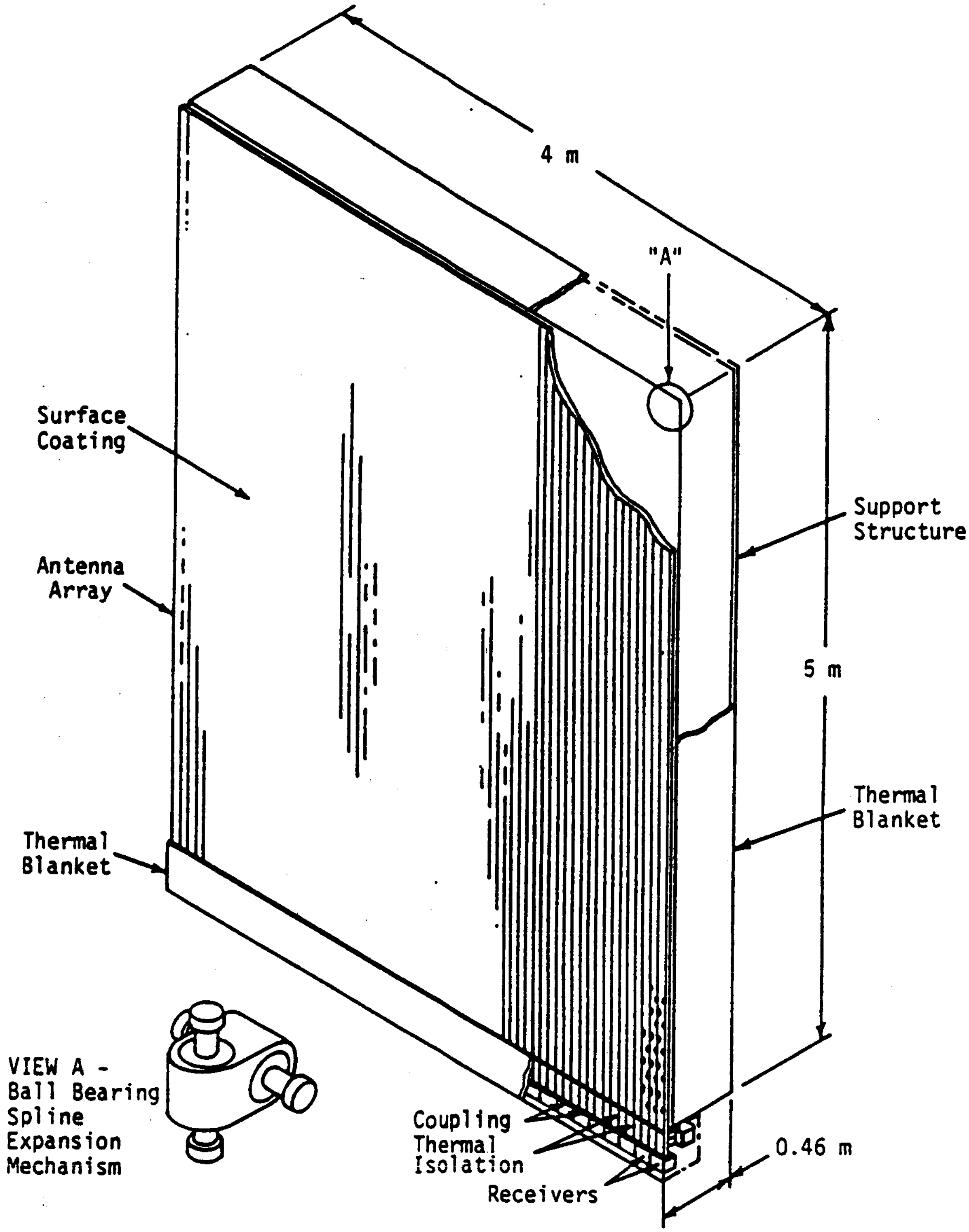


Figure 10-3. Antenna Schematic for Space Shuttle.

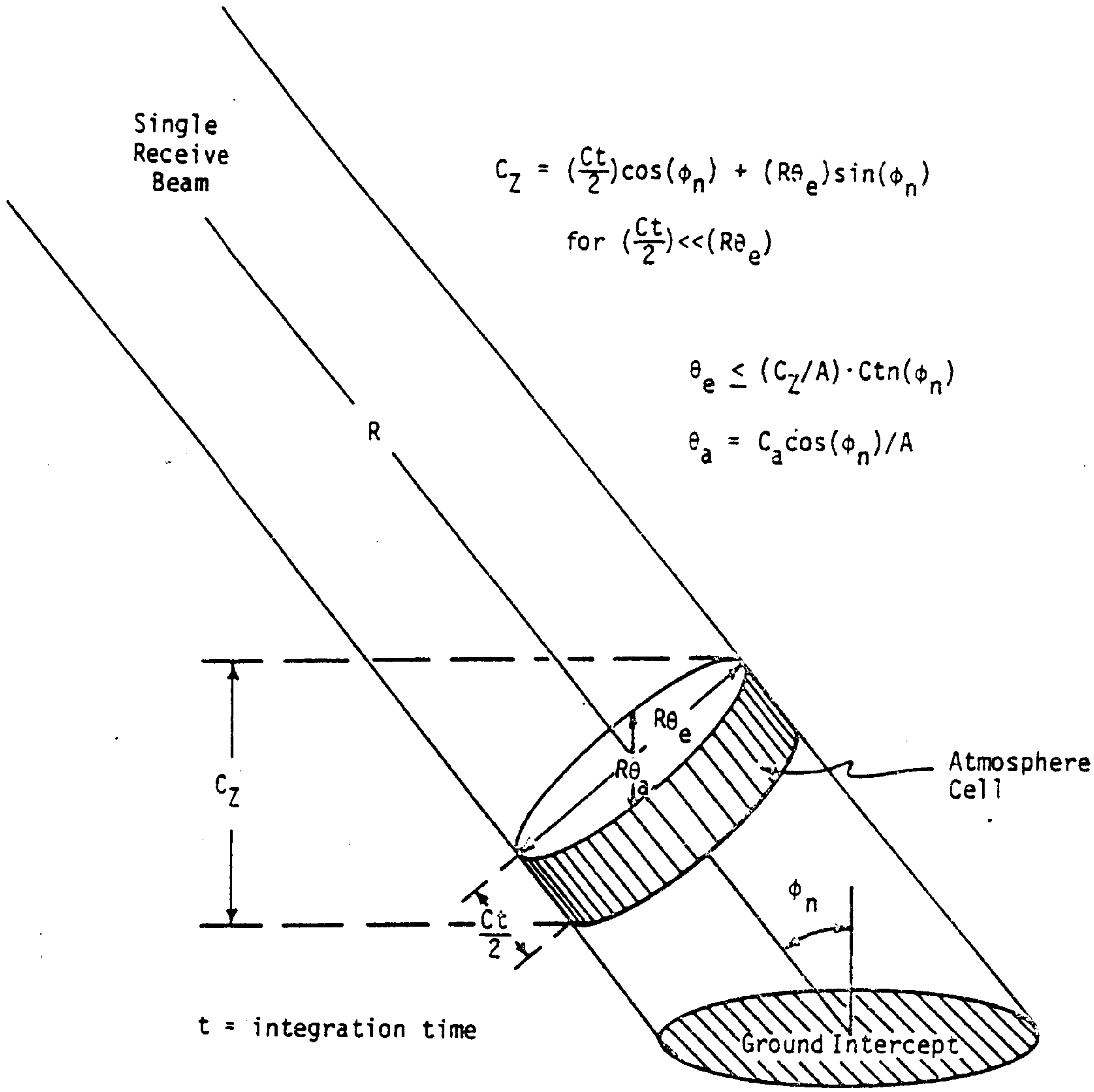


Figure 10-4. Atmosphere Cell Geometry.

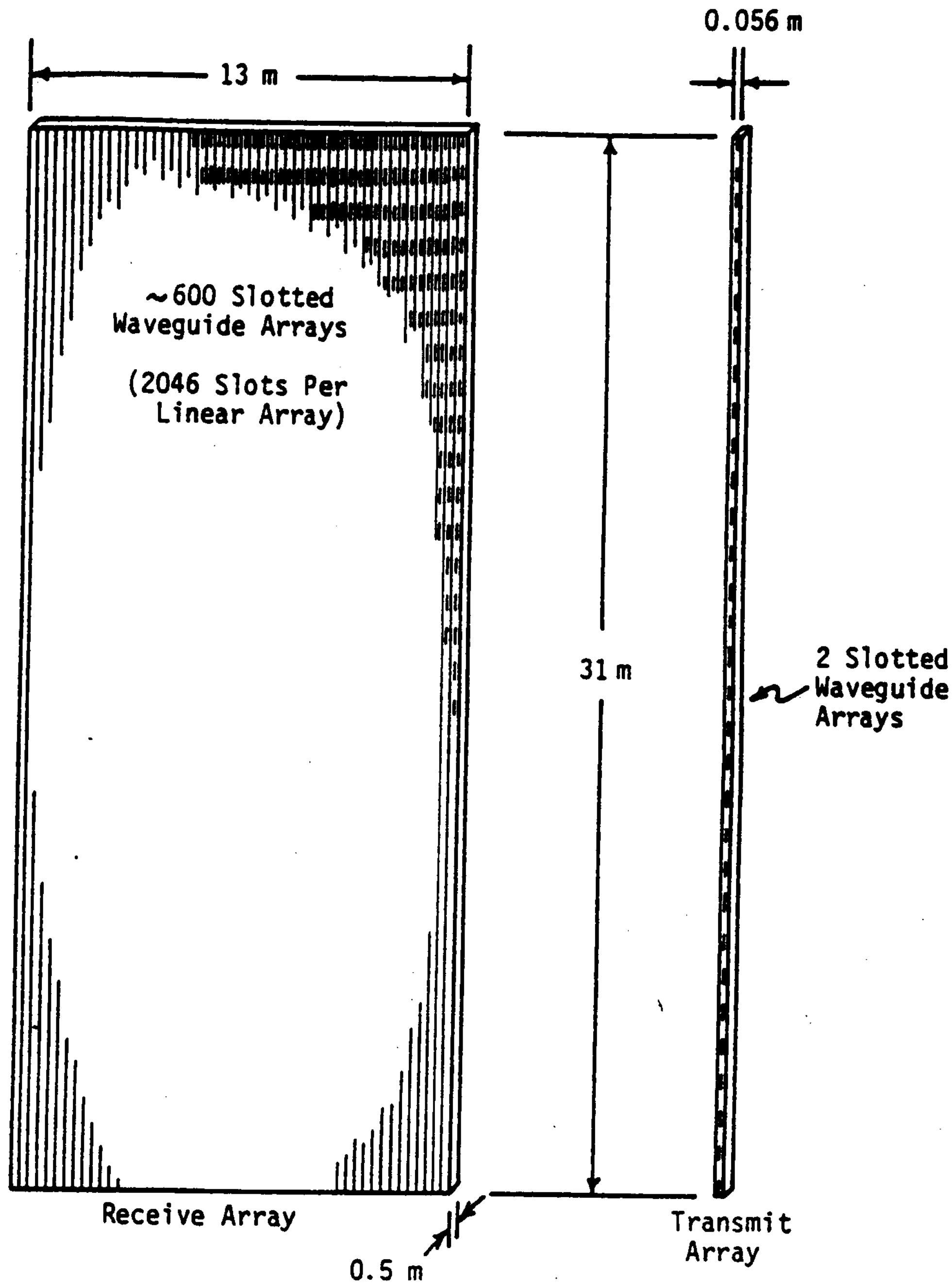


Figure 10-5. Meteorological Radar Antenna for Space Station.

and suspension system, individual receiver elements, thermal protection and feed network, the total weight could be on the order of three times that of the waveguide or 5000 kg.

Considering the fact that these antennas are in excess of 1000 wavelengths long, their development for space application must be considered a significant advancement in the state-of-the-art and will be difficult to accomplish. Consideration should be given to the development of some form of an active (self-aligning) array consisting of a number of radiating elements and phase shifters operating in conjunction with a local source(s) for measurement of relative position of the elements. The difficulty in this is that there are over 1.2 M individual radiating elements in the receive array which would be impossible to duplicate with each element having an active phase shifter. Some combination of traveling-wave, linear, slot arrays, and active phase shifters may, however, be feasible (i.e., 1200 waveguide arrays with active phase control). This, of course, would result in higher sidelobe levels and would require a complete systems analysis to determine feasibility.

The mode of operation of the radar is illustrated in Figure 10-6. A burst of pulses are transmitted at a PRF such that only one pulse or it's return is transiting the atmosphere (0 to 20 km) at any time. The burst is stopped at the time when the return should arrive at the station from the first pulse reflecting from an atmospheric cell 20 km high. The next burst is initiated after the time when the return should arrive from the last pulse reflected from the ground. This results in a PRF of 5.3 kHz and a BRF of 216 Hz with 12 pulses per burst. The signal received on each beam of the receive array is sampled in time as the pulse progresses through the atmosphere to the ground. Allowing $Ct\cos(\phi n)/2$ to be 200 m, results in an integration time of approximately 2.0 microsecond and approximately 100 samples of the atmosphere (overlapping in altitude) per pulse. To guarantee statistically independent samples from each successive pulse, a frequency agile transmitter is used to shift the frequency between each pulse by at least 0.15 Mhz and up to 12 MHz. In 143 ms (approximately 1 km forward motion), 370 independent samples are acquired of each cell. These

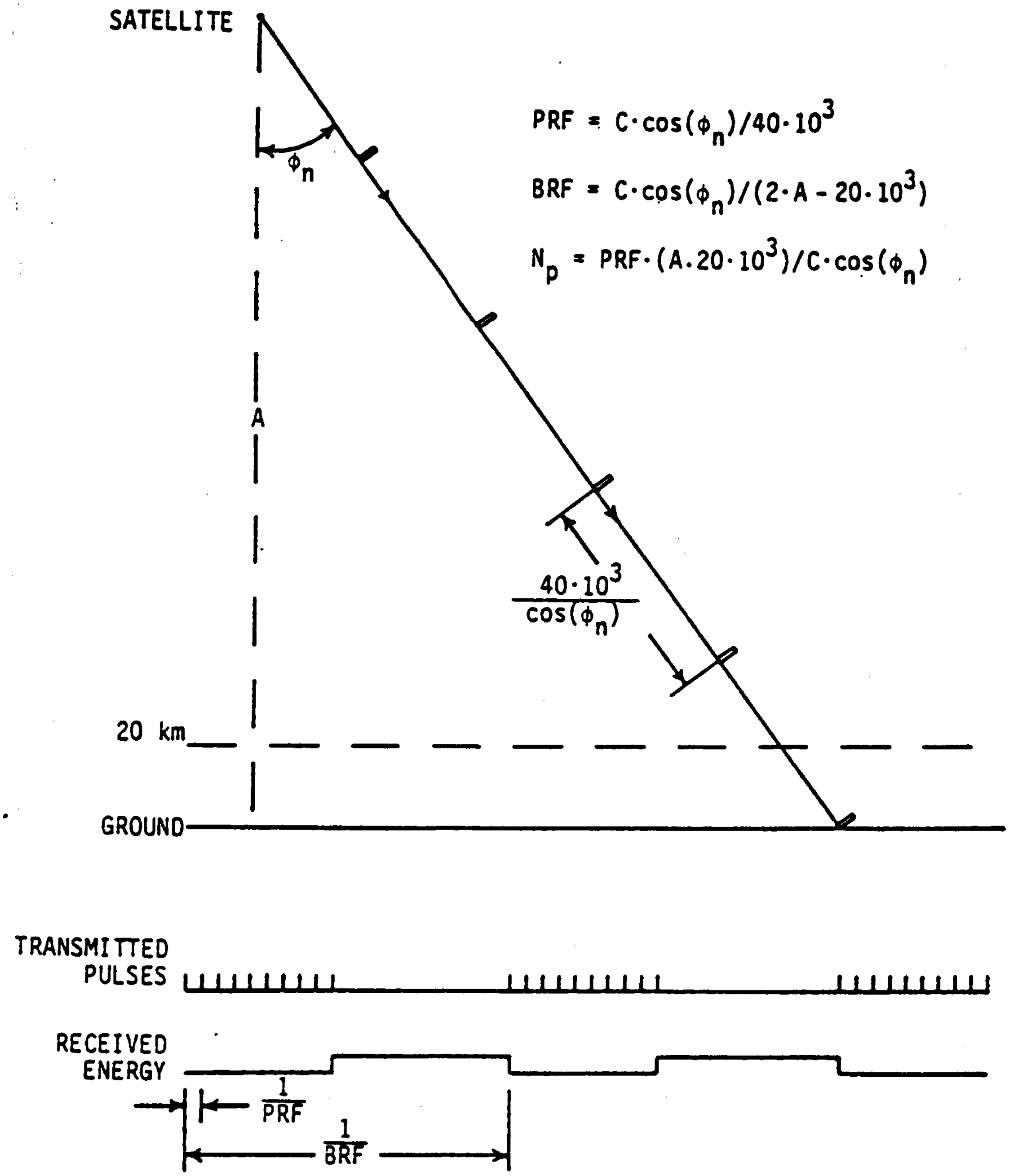


Figure 10-6. Meteorological Radar Pulse Pattern.

samples are averaged in the processor to obtain one cell measurement. To maintain the desired SNR, a peak transmitter power of approximately 25 kW with an average power during the burst of 265 W and a long term average of 132 W is required.

10.5 DATA ACQUISITION AND CONTROL

While operating, the instrument will generate approximately 114 K data samples per second at the output of the data processor. Assuming a dynamic range of 50 dB, and a resolution of 0.1dB; this corresponds to a bit rate of approximately 1.2 Mbits/second. If run continuously, this would be approximately 100 Gbits/day. For purposes of technology evaluation, the primary data to be collected consists of measurements over areas with significant ground truth information available. This will probably limit the amount of useful data to approximately 10-20% of the total available and, assuming a daily data dump to the ground, the on-board storage requirements would be approximately 20 Gbits.

Overall experiment control will be conducted from the ground with an instrument specialist on-board SS to monitor the instrument performance and set-up the instrument run times and various algorithms in the data processor.

10.6 DATA COMMUNICATIONS AND HANDLING

Data acquired by this instrument would be transferred, after data processing, to a central computer and storage facility. This data could be monitored during acquisition, but access to the computer would also be required for review of the data for analysis of instrument performance on-board SS. This analysis would be conducted using the specialized display unit of the instrument. A daily data dump to the ground would limit the required data storage to approximately 20 Gbits.

10.7 TELEMETRY

The primary requirement for telemetry in this experiment is for a daily data dump to the ground. This would require the use of the TDRSS, Ku-band return link, and would require approximately 7 minutes at a rate of 50 Mbits/second.

10.8 CREW ACTIVITIES

The assembly and installation of the antenna system for this experiment involves an estimated 30 manhours of EVA. In addition, the installation of the data processor, control and display units involves an estimated 10 manhours of IVA. After assembly and checkout, the continuing operation of the experiment will require approximately 2 hours per day of an instrument specialist time to monitor the instrument performance and program the instrument for future operations. An estimated 20 manhours of EVA and 5 manhours of IVA would be required for disassembly and repackaging of the instrument for return to earth. A time line for assembly and installation is shown in Table 10-1.

10.9 GROUND PROCESSING

Ground data processing would be required to support the experiment evaluation on a daily basis. Data would be required in both numerical form and in the form of three-dimensional mapping of rain rates at selected altitudes. Support would also be required for collating of ground truth data and errror analysis. The results of the daily effort in data analysis would feed back to SS for programming of the experiment.

10.10 SUPPORT EQUIPMENT

This experiment will require the mobile manipulator and a special mounting structure for the assembly and installation of the antenna system and interconnections. It will use the central computer and data storage facility as well as the TDRSS communications systems.

Table 10-1. ASSEMBLY AND INSTALLATION TIMELINES

FUNCTION/TASK	TIME MANHOURS	METHOD	FEASIBILITY*	COMMENTS
Assemble and install antenna mounting structure	4	EVA/MRMS	2	
Install antenna drive assembly	1	EVA/MRMS	2	
Install antenna subassemblies (panels)	9	EVA/MRMS	4	Antenna not defined, possibly 9 panels. Tolerance unknown.
Transfer data processor, control and display units to SS Module	1	EVA/MRMS	2	
Install cabling from antenna to module	0.5	EVA	2	
Connect console and initial checkout	4	IVA	4	Antenna checkout procedure unknown.
DAILY OPERATIONS TIMELINES				
Monitor instrument parameters	0.5	IVA	2	
Review data	1.2	IVA	2	
Program instrumentation for acquisition	0.3	IVA	2	

Table 10-1. (Continued)

FUNCTION/TASK	TIME MANHOURS	METHOD	FEASIBILITY*	COMMENTS
DISASSEMBLY TIMELINES				
Remove cabling	0.3	EVA	2	
Transfer data processor, control and display units to STS	1	EVA/MRMS	2	
Remove antenna subassemblies and transfer to STS	6	EVA/MRMS	2	

*1-Proven, 2-Easy, 3-Hard, 4-Questionable, 5-Not Possible

10.11 THERMAL, POWER, WEIGHT, AND VOLUME REQUIREMENTS

The estimated packaged dimensions are 12 m length, 4.3 m width, and 4.1 m height. The assembled antenna is estimated as 31 m length, 13 m width, and 0.46 m depth, with a weight of 5000 kg. The data processing, display and control assembly is estimated at 1.0 cubic meter with an operating power requirement of 1.5 kW. The operating temperature range is from -20 to 40 degree C with a heat rejection requirement of approximately 1.2 kw.

11.0 TDMX 2322 -- LASER PROPULSION

CONTACT: Dr. Edmund J. Conway, MS 160
NASA LaRC (804) 865-3781

11.0 GOALS AND OBJECTIVES

The purpose of the Laser Propulsion experiment is to develop the technology of propulsion devices which derive their energy from a focused laser beam. In this experiment, a small thruster will be evaluated in which a cesium-seeded, hydrogen gas stream is heated and expanded by a focused laser beam, producing a jet of high-energy gas. The ultimate application of large, laser-powered thrusters would be for propulsion of orbital transfer vehicles or for overcoming effects of atmospheric draft for large orbiting platforms. Power for these applications would be derived from large centrally located solar collectors and beamed to the vehicle using lasers.

A vehicle employing a laser powered thruster would have two significant advantages over the conventional propellants which derive thrust through expansion resulting from the heat of chemical reaction. These advantages are: (1) the individual vehicles do not have to carry the energy supply or energy conversion devices necessary to support the propulsion, and (2) greater specific impulse[1] may be obtained from propellant material because the laser heating process is not thermally limited to combustion temperatures, hence greater propulsion energy is obtained from a fixed quantity of propellant.

The goal of this mission is to: (1) demonstrate feasibility of solar pumped laser power propulsion units capable of operation in space, and (2) evaluate performance and determine operational parameters for the candidate propulsion system(s).

---

[1] Specific impulse is defined as the length of time a certain quantity of fuel can generate a thrust which is equal to its own weight at the earth's surface.

11.1 METHODOLOGY

11.1.1 Description

Propulsion is generally produced by channeling thermally expanded gases through a nozzle, generating a force from the equal and opposite reaction to the momentum of the escaping gases. Typically the energy for the heating of the gases (or for converting a solid to a gas) is derived from a chemical reaction (combustion). Laser powered thrusters also operate from thermal expansion of gases ducted through a nozzle. The major difference is that the source of energy is a laser beam focused in a chamber containing the gaseous propellant. Gases with low density will provide the greatest propulsion to weight ratio (specific impulse) and therefore are the most desirable for use in the thruster. Hydrogen, the lightest element in the periodic table, provides the greatest specific impulse. However, hydrogen in gaseous form at normal ambient temperatures does not absorb significant energy from incident radiation. Several techniques are available to transfer the energy from the laser beam to the hydrogen:

a. Focus the laser on a solid absorbent material in contact with the hydrogen gas. Energy absorbed by the material is transferred to the gas by conduction.

b. Create a plasma in the hydrogen which efficiently absorbs the radiation from a wide range of the spectrum. The plasma, once established is sustained by energy which it absorbs from the incident radiation. An 8 kW beam is required to sustain the plasma once initiated. Even greater laser power or another mechanism is necessary to establish it.

c. Add another gas to the hydrogen to increase its absorbance to the incident radiation. Although the other gases, because of their higher mass, lower the specific impulse of the hydrogen, they are necessary to increase the optical absorbance of the hydrogen so it may be heated successfully by the incident radiation. A small thruster such as this requires an estimated 1 kW laser beam for operation.

Either b. or c. may be used as the basis of a working engine for generation of thrust. The mechanism described in b., direct absorption of the energy by a plasma, will generally result in higher temperatures (up to 20,000 K at the center of the plasma) and will therefore produce an engine which derives a higher specific impulse from the gas. The mechanism described in c. is less efficient, however since it can operate from a lower powered laser beam, it is the appropriate one for this evaluation.

A small thruster will be fabricated using hydrogen seeded with cesium (and possibly water) as the propellant material. A sketch of the thruster is shown in Figure 11-1. The thruster consists of a cylindrical chamber approximately 3 to 5 cm in diameter with a window/focusing lens on one end to concentrate the laser beam near the center of the chamber. Hydrogen is fed into a coaxial sleeve at the exhaust end of the thruster, and flows over the outside of the thruster to provide cooling. The hydrogen then passes into the chamber through orifices which direct the flow onto the window/focusing lens to keep it as clear as possible. The seedant, cesium or a mixture of cesium and water, is admitted to the chamber upstream of the focal point of the lens. The intense radiation at the focus is absorbed by the seed and raises the temperature of the seedant-hydrogen mixture to about 6000 K causing a high degree of expansion in the gas. The expanded gas is forced through the nozzle to produce a low-level thrust.

The thruster will be mounted along with supporting equipment on a pallet in a configuration resembling that shown in Figure 11-2. The pallet will be installed approximately 35 m from the solar-pumped laser (TDMX 2121) such that the laser beam is incident on the center of the window/focusing lens of the thruster chamber. It will then be evaluated by operating it for short periods of time (e.g., 10 seconds), monitoring the thrust produced and operating parameters such as temperature, flow rates, and outlet gas velocity using an instrumentation system such as that shown schematically in Figure 11-3.

The evaluation of the thruster consists of several tests where each test consists of a series of short duration (10 second) thruster firings. Because of the short duration of each firing, direct control of the

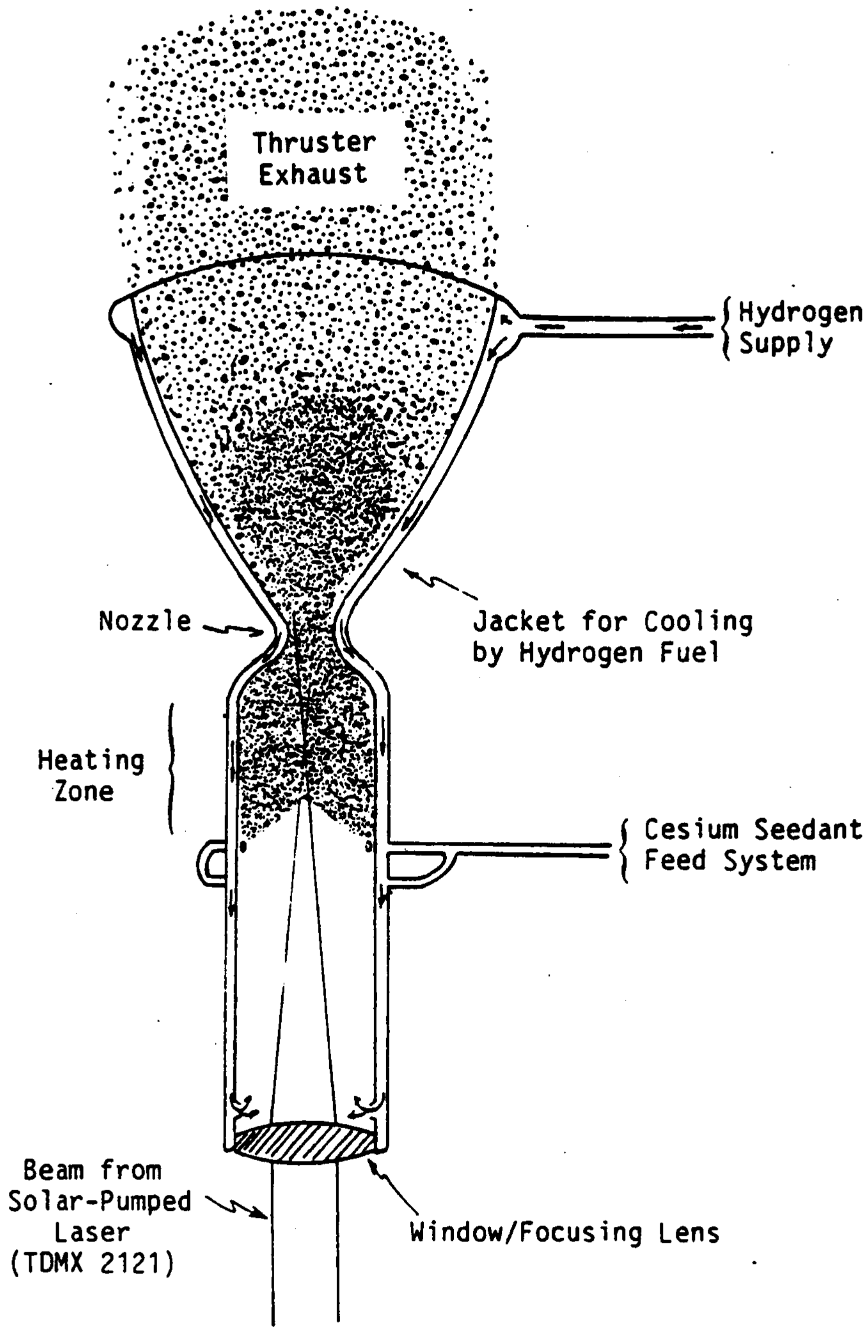


Figure 11-1. Basic Thruster Unit Using Cesium Seedant.

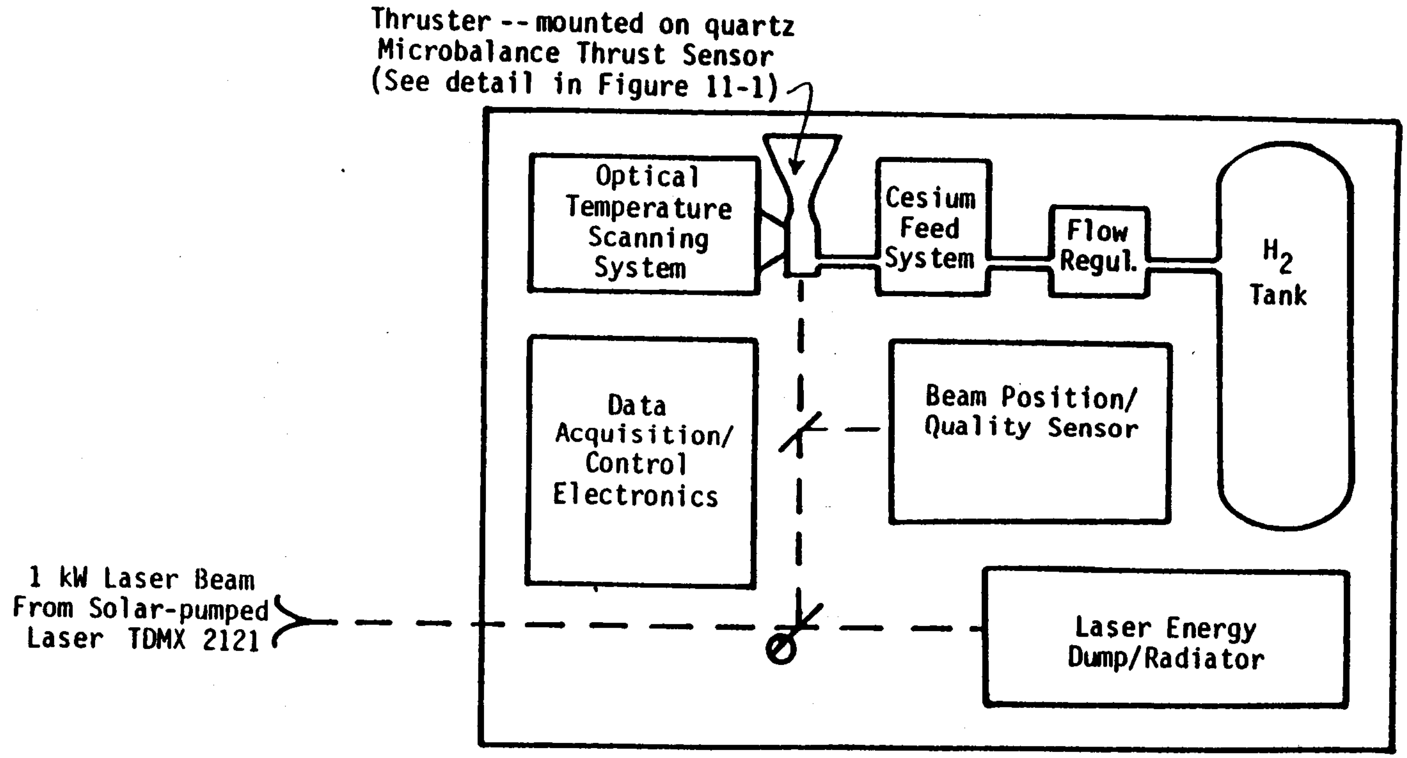


Figure 11-2. Pallet Mounted Thruster Evaluation System.

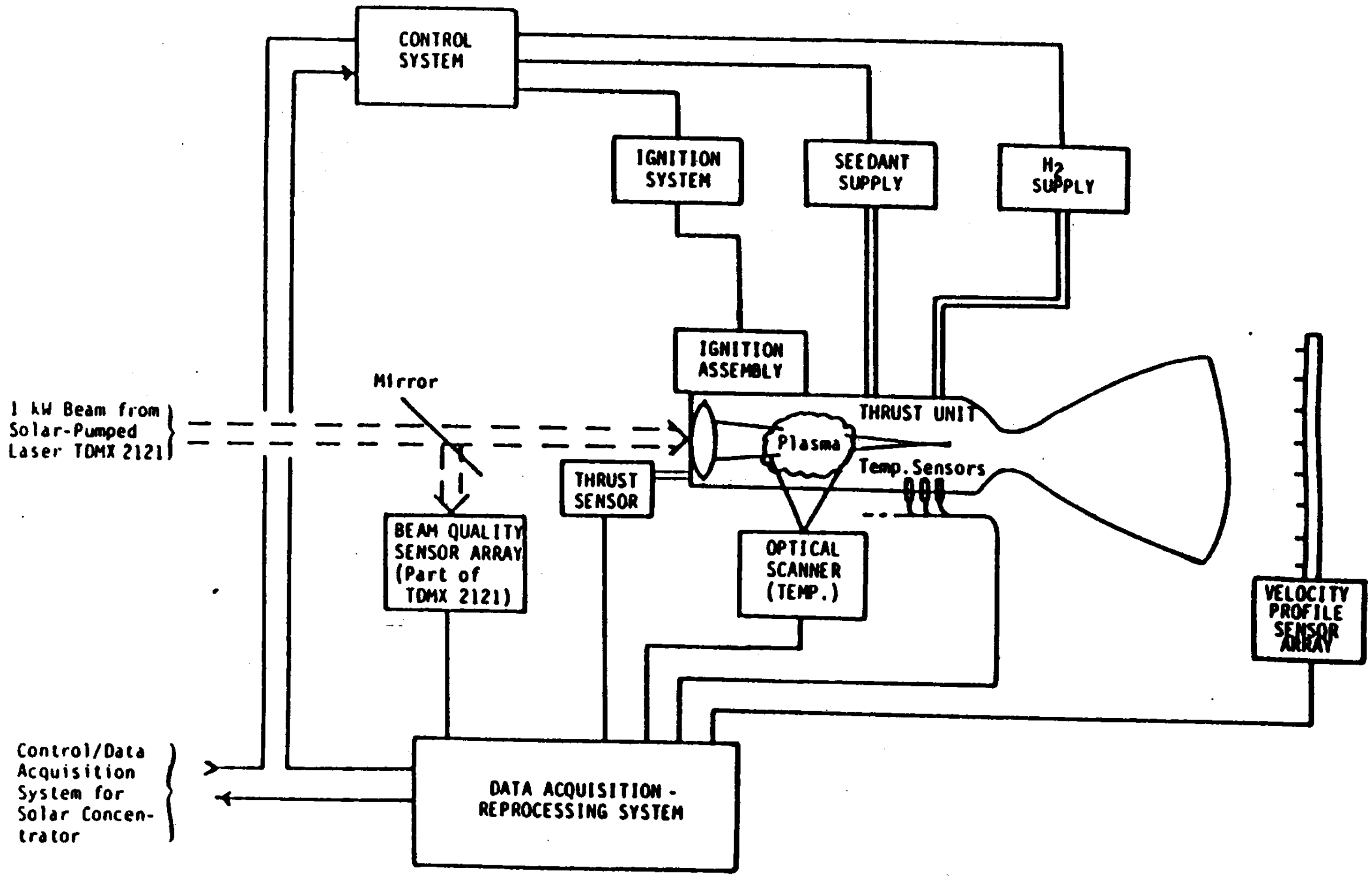


Figure 11-3. Control and Signal Flow Diagram for Laser-Propulsion Experiment.

experiment from the ground during the thruster firing is neither practical or desired. Control of the experiment is most conveniently performed by the computers located within the experimental apparatus with the Space Station crew having the responsibility for initiating a test and having the capability of terminating or suspending the test if necessary. Operating parameters, such as H_2 flow and seedant concentration would be passed to the experiment controller either from the ground via telemetry or from a crew member via keyboard entry. However, strategy for each test series would be developed by the principal investigator on the ground. To provide quick look data for evaluating the success of a test to assist the investigator in planning future tests, a synopsis of the test data for each experiment should be transmitted to the ground during the progress of the test followed by a more extensive data set transmitted later at a much lower priority.

Although NASA, the USAF and the US Army have expended a significant effort toward the development of laser powered thrusters for space applications the technology is in the early stages of development. Insufficient data exists at the present time on cesium-seeded, hydrogen thrusters to produce a detailed design of the thruster or even calculate the desired operating parameters. Additional work, both laboratory scale model testing and modeling will have to be performed before an accurate determination can be made of the experimental apparatus which is suitable for testing this concept on the Space Station. The operational parameters and a detailed test scenario must be known before the logistical support requirements of the test facility can be confidently determined. Consequently, the data presented for this experiment concerning the configuration and operation of the laser powered thruster is based on the experience reported in papers and reports describing modeling efforts and experimental work on thrusters using air as the working gas. New data concerning the behavior of the cesium-seeded, hydrogen thruster may affect the design, significantly changing the requirements.

11.1.2 Environment

The laser propulsion experiment should be located where the chamber may be illuminated by the solar pumped laser beam after it has traveled sufficient distance to adopt far-field characteristics (greater than ten times the laser cavity length, a total distance of 10 m). The thruster must be mounted such that its position relative to the laser beam is stable, although small lateral displacements perpendicular to the beam axis may be tolerated by employing an oversize focusing lens to capture the light when it is off-center. The orientation of the thruster is more critical since any rotation of the thruster relative to the beam will displace the focal region in the thruster thus moving the region of intense heating away from the center axis of the thruster. A significant misalignment would e required before the heating region would be moved sufficiently close to the wall to cause damage. However, the instrumentation for monitoring the thermal profile of the heated volume in the gas may be simplified if cylindrical symmetry can be assumed and movement of the center of the heating region would invalidate the assumption.

Locating the laser thruster on the solar concentrator structure (TDMX 2121), will insure that these stability requirements are met as long as the structure is sufficiently rigid. Structural analysis of the concentrator structure revealed that even under 0.1 g acceleration, the main keel of the structure would only deform a few tenths of a mm, thus maintaining an appropriately stable platform for the mounting of the thruster. The long length of the concentrator keel where the thruster is to be mounted allows it to be located 35 m from the solar laser, safely within the far-field regime.

Since the laser thruster receives its energy from the solar pumped laser, all considerations such as pointing accuracy, orbit, and position on the Space Station are governed by the requirements for the solar concentrator and are specified in Section 4, TDMX 2111. Operation of the concentrator on consecutive orbits will not be required since the thruster will be operated only for short periods consisting of a series of 10 second tests separated by a few minutes followed by a period of inactivity while the results are reviewed.

11.1.3 Data Acquisition, Processing, and Control

Monitored Parameters -- Because of the scale of the thruster experiment, the space environment where it is located, and the test scenario of a series of short term tests, real-time observation and control of the thruster experiment by the crew may not be necessary once reliability is demonstrated. However, the operating parameters of the thruster will be monitored intensively by a recording data acquisition system in order to: (1) detect any unsafe operating conditions, (2) determine the optimum operating parameters, and (3) to improve the modeling of processes within the laser powered thruster.

Table 11-1 contains a list of parameters which may be monitored for the cesium-seeded, hydrogen thruster. Monitored parameters include point measurements of temperatures within the chamber of the thruster, flow rates for materials entering the chamber, and exhaust velocity measurements. Critical measurements for the thruster include the thrust produced and the temperature of the heated cesium-hydrogen mixture in the chamber. Thrust produced would be measured by a sensitive quartz microbalance capable of accurately quantizing the low-level forces (less than 1 N). Platform acceleration measurements would be made simultaneously to allow removal of force measurement artifacts due to Space Station accelerations. Temperature measurements of the gases in the chamber would have to be made by a thermal imaging system located externally to the chamber. Focused imaging and spectrophotometer techniques would be used to determine a two dimensional temperature profile in a plane passing through the central axis of the chamber. Temperatures of the rest of the chamber would be determined assuming axial symmetry. Detection of temperatures in the range produced by the seeded thruster using this technique may prove difficult because, generally, higher temperatures are required for this type of measurement.

Data Acquisition/On-board Processing -- Data acquisition for the thrust experiment would be performed by an intelligent data acquisition system located on the thruster pallet which would then pass the acquired

Table 11-1. Measurements To Be Made On The Cesium Seeded Hydrogen Thruster During Tests.

Measurement	Sensor	No. of measurements in set	Bits per measurement	Sample rate s^{-1}	*Data to be telemetered: measurement	*Data to be telemetered: rate	Data telemetered per 10s test, bits
Thrust	Quartz Microbalance	1	12	10	1(1)	10(2)	1200
Platform Acceleration	3-Axis Accelerometer	3	12	10	3(3)	10(2)	3600
Hydrogen Mass Flow	Heated Tube Flowmeter	2	12	10	2(2)	2(2)	480
Hydrogen Temperature	Thermocouple	2	10	10	2(2)	2(2)	400
Hydrogen Pressure	Diaphram Transducer (strain gage)	2	10	10	2(2)	2(2)	400
Seedant Flow	Mechanical Feed Rate Sensor	1	12	10	1(1)	2(2)	240
Beam Position/Quality/Energy	Photodetector Array	100	10	10	100(10)	0.5(2)	5000
Wall Temperatures	Thermocouples	48	10	10	48(6)	2	9600
Exhaust Temperatures	IR Detector	1	10	10	1(1)	2	200
Heated Zone Temperature Profile	(Imaging Spectrophotometer - To Be Developed)	200	8	10	200(10)	2	32000
Exhaust Velocity	Pressure Profile Sensed By Pilot Tubes	10	10	10	10(1)	2	2000
Hydrogen Tank Pressure	Diaphram Transducer (strain gage)	1	10	0.5	1	**	
Hydrogen Tank Temperature	Thermocouple	1	10	0.5	1	**	
Seedant Supply Level	Displacement Transducer	1	10	0.5	1	**	80
Energy Dump Temperature	Thermocouple	4	10	0.5	4	**	
Energy Into Dump		1	10	0.5	1	**	

* Numbers in parentheses reflect number and rate of measurements for selected tests which must be passed to investigators on ground for timely review.

** Beginning and end of each test.

data through the master data acquisition system associated with the solar concentrator for dissemination to the Space Station crew and onto the ground. Processing would be performed to convert the data to engineering units, monitor the process for unsafe conditions, and compress the data to reduce the quantity of data passed. Also the thermal image data from the temperature sensing system would be processed on-board to reduce the quantity of data necessary for transmission. In these cases, statistical summaries could be transmitted soon after the test for review and assessment of test results by the investigators on the ground. The complete data set could then be transmitted later at a lower priority to allow a more extensive look at the data.

Data Communications -- Data transmitted to the ground would fall into one of two categories, (1) synoptic data required for quick test review and (2) complete data intended for detailed analysis of thruster performance at a later time. Data for both categories would consist of conventional digitized data, formatted and packed for compatibility with the data communication system being used. Data volumes for each 10 second thruster test are 0.95 kbytes for the synoptic data and 6.9 kbytes for the detailed data. Total data for a series of tests made during a single orbit would obviously depend on the number of firings in the series. Assuming a test consisting of 20 firings for a series followed by a period of inactivity at least one day to provide time for data review, the maximum data volume to be transmitted within one day is 157 kbytes. These quantities are based on the monitored parameters and sample rates shown in Table 11-1 and summarized in Table 11-2. The latter table also reflects the assumed up-link data volume of 1 kbyte for test parameters passed to the experiment controller.

Ground Processing -- Major processing of data would primarily be conducted on the Space Station to reduce the quantity of data transmitted to the ground. Ground processing would be limited to presentation of the results in graphic format to aid in interpreting the results and analysis of the results to validate or improve computer models of the thruster

Table 11-2. Data Volume Estimates of Data Telemetered to Earth for Laser Propulsion Experiment

Data to be transmitted for each 10s thruster firing (for "quick-look" review - medium transmission priority)		0.95 kbytes
Data to be transmitted for each 10s thruster firing - complete data set - (sent when time is available)		6.9 kbytes
Data downlinked to earth daily assuming one test per day consisting of 20 bursts		
	Synoptic Data	19 kbytes
	Detailed Data	138 kbytes
	TOTAL	157 kbytes

performance in the space environment. Consequently, there is no requirement for extensive on-line ground processing of experiment data for the laser propulsion experiment.

11.1.4 Required Crew Support

Space Station crew support is required for this TDM in three major areas: (1) installation, (2) monitoring the experiment, and (3) removal of equipment. Specific activities are discussed in the following paragraphs:

Installation -- The installation of this experiment consists of removing the experimental apparatus from the shuttle, transporting it to the solar concentrator, fastening it in place and adjusting the position of the thruster to achieve proper optical alignment. Because of its relatively small size, the experimental apparatus including thruster chamber and associated instrumentation can be assembled on one pallet, eliminating assembly. Instead, EVA time will be spent primarily in the fastening of the pallet to the solar concentrator keel truss, connecting cables between the solar concentrator control electronics and the pallet and aligning the thruster with the laser beam. The estimates of EVA time shown in Table 11-3 assume that the alignment is performed with the aid of a low-level visible laser which is coincident with the output of the solar-pumped laser. (For safety reasons the solar pumped laser must remain inactive when the crew is in the region of the beam path.)

Operation -- Initial tests of the thruster should be initiated by and observed by a member of the Space Station crew from a position within one of the modules. Initiation of the test and viewing of acquired data would be done through a terminal keyboard/display system. Actual visual observation of the thruster in operation should not be necessary. Duration of a sequence of tests would be limited to approximately one-third of an orbit or 30 min (limitation due to the maximum period which the concentrator may track sun continuously).

Equipment Removal -- After completion of all tests, the thruster chamber should be removed for return to the ground for analysis. The remaining instrumentation and logistical supply systems may be scrapped.

Table 11-3. Crew Activities Required for Laser Propulsion Experiment TDMX 2322

	IVA	EVA
<u>Installation</u>		
Remove apparatus from Shuttle		5 min
Transport to Concentrator		5 min
Fasten in place		10 min
Align - physically position thruster (Assumes presence of "safe" alignment laser)		15 min
Connect to concentrator computer/controller		5 min
TOTAL FOR INSTALLATION		40 min
<u>Operation</u>		
Initiate test after parameters are passed from ground to processor	30 min	
Review monitored data on terminal and update test parameters if necessary		
Miscellaneous repair tasks	unknown	
<u>Equipment Removal</u>		
Remove thruster from pallet for shipment to earth for examination		20 min

In the event that multiple thruster systems are developed for testing and evaluation, the estimates shown in Table 11-3 should be multiplied by the number of thrusters. All of the line items in the estimate would apply to each system to be installed and tested.

11.1.5 Logistical Requirements

Estimates for required logistical requirements for this TDM were made by identifying major components required for the TDM and then assessing the weight, size and power requirements of each component. Estimates were made by either obtaining specifications of commercial of equivalent complexity which performed a similar function, or by estimating the size and calculating the mass based on the amount of material necessary for fabrication. The requirements for the experiment were then determined by summing the contributions of the individual components.

The components are listed and discussed individually in the following paragraphs:

Energy Dump for Laser Beam -- An energy dump is required which can collect and dissipate from the 1 kW laser beam. A hollow spherical absorber is proposed with an opening sufficiently large to accommodate a 5 to 10 cm beam size with some additional allowance for beam misalignment. A structure of the size shown will also have sufficient surface are to dissipate the energy without the need for external cooling.

H_2 Cylinder -- Hydrogen requirements for the experiment are 0.1 g/s. For 100 10s tests, 100 g would be required, occupying at STP about 1.1 m^3. Compressing the gas to 1000 psi at standard temperature would reduce the storage volume to 16.2 liters. The size and mass figures given in Table 11-4 are derived from the weight of aluminum cylinders of a similar size used in commercial service.

Thruster -- The thruster package consists of three subassemblies: (1) the thruster and an associated window/focusing lens and nozzle assemblies, (2) a thermal imaging system for monitoring temperature of the heated gas within the thruster chamber, and (3) a seedant injector system. The

Table 11-4. Size, Mass, and Power Estimates for TDMX 2322 Components.

Component	Size	Mass	Electrical Power Required	Heat Dissipated
Energy Dump for 1 kW Beam	0.6 m sphere	9 kg	--	1 kW
H_2 Cylinder	0.2 m dia. x 1 m	15 kg	--	--
Thruster				
Engine/Nozzle	0.1 m dia. x 0.3 m	2 kg	--	400 W*
Seedant Supply System	0.1 m x 0.1 m x 0.2 m	2 kg	20 W	20 W*
Thermal Imaging System	0.1 m x 0.15 m x 0.2 m	2 kg	10 W	10 W*
Beam Quality/Power Sensor	0.1 m x 0.1 m x 0.2 m	2 kg	20 W	20 W*
Electronics for Control and Data Acquisition	0.25 m x 0.2 m x 0.4 m	13 kg	140 W	140 W*
Pallet and Shield	1.0 m x 1.2 m x 0.7 m	10 kg	--	--
TOTAL		55 kg	190 W	

* power dissipation is intermittent. Value shown is peak value.

required size of the thruster is not well defined at this time. For these estimates, the size of the chamber proposed for pure hydrogen plasmas was used as the basis for a preliminary estimate of the required chamber volume. The seedant injection system was assumed to be a heated container to keep the cesium in liquid state. Pressure from the hydrogen supply could then be used to atomize the cesium into the chamber. For a hydrogen-cesium mixture with a 3:2 mass ratio, approximately 67 g of cesium (occupying about 30 ml) would be required for use with the quantity of hydrogen specified in the following paragraph. Thus the cesium injector system would be a very compact assembly. The thermal imaging system would consist of focusing optics, gratings, and photodetector array and data acquisition/processing electronics. Size, weight and power estimates were determined from an estimate of the volume required to contain these items and the power required for a medium-speed A/D converter and associated low power microprocessor.

Beam Quality/Power Sensor -- The beam quality sensor operates off a portion of the beam and can therefore be an array for photodetectors and electronics to determine beam center location, size, and measurements of power distribution across the beam. The integration of this processing into the sensor would reduce the need for multiple connections between the photodetector array and the central electronics for the TDM and would reduce the amount of data to be passed and stored. Power estimates were taken from a moderate performance A/D and low power microprocessor system.

Electronics for Control Data Acquisition -- The electronics for this experiment must perform the following functions.

1. Receive control information from the solar concentrator computer system.
2. Control the hydrogen supply system, the seedant injection system, the mirror which directs the laser beam into the chamber, etc.
3. Accept data from the beam quality/power system and the thermal imaging system.

4. Monitor sensors located on the thruster and throughout the experiment.
5. Process and format data for presentation to the crew member observing the experiment.
6. Format and convey data for transmittal to Earth.

Size and power estimates for the electronics were determined from a commercially available, ruggedized microcomputer system which employs existing technology. Power and weight will probably be reduced as new processing capability is developed between now and the time this experiment is implemented.

All of these components may be mounted on a 1 m by 1.2 m pallet resulting in an experiment envelope of 1 m by 1.2 m by 0.7 m bringing the total mass of the experiment to 55 kg.

The relatively immature state of this experiment does add considerable uncertainty to these estimates. Any change in desired test duration, hydrogen or seedant flow, or input power will have considerable impact on the package size, mass, and the power requirements.

11.1.6 Inter-experiment Compatability

This TDM is the last in a series of four TDMs which progressively develop and demonstrate the technology for beaming power through space from a centrally located solar collector. Therefore, this experiment is dependent on the facilities developed during the first three experiments of the series: TDMX 2111, TDMX 2121, and TDMX 2122.

Characteristics of the thruster experiment which could impact the operation of other experiments or Space Station functions are discussed in the following paragraphs:

Production of Thrust -- The thruster experiment is the evaluation of experimental apparatus whose function is to generate propulsion or thrust. However the thrust will be small, probably less than 1 N. Since this is significantly less than the accelerations generated by crew motions, it is not expected that the effect will be significant.

C-3

<u>Exhaust of Hot Hydrogen-Cesium Gas</u> -- Experiments which would be contaminated by hydrogen or cesium could be adversely affected by this experiment as quantities of approximately 1 to 2 grams may be discharged during each 10 second test. The incompatibilities could be resolved through scheduling so that the affected experiments are not operational during the short operational period of the laser propulsion experiment.

12.0 TDMX 2073 - ADVANCED STRUCTURAL DYNAMICS/CONTROL METHODS

CONTACT: Claude R. Keckler, MS 161
NASA LaRC (804) 865-4591

12.1 GOAL

Future large spacecraft will utilize highly flexible structures as an alternative to rigid structures due largely to cost considerations. Spacecraft designers will, therefore, require a technology data base for the implementation of such structures and their related control systems. NASA programs such as Control of Flexible Structures (COFS) will facilitate the initial development of this technology for implementation on Space Station.

The goal of this technology development mission is to support the on-orbit validation of flexible structure technology utilized in the construction of Space Station. This experiment is specifically designed to evaluate and refine algorithms and techniques for the system identification and control of flexible structures in an orbital environment.

12.2 OBJECTIVES

The objective of this experiment will be to provide an on-orbit test facility, utilizing the Space Station structure, for the study of flexible structure dynamics and control. Initially, acquisition of data representing the transient response characteristics of the IOC Space Station structure and its control systems will be useful in evaluating analytical predictions of on-orbit dynamics with emphasis on such issues as: damping effects in zero gravity, nonlinear joint effects, effects of large amplitude disturbances such as STS docking or reboost, interaction between flexible structures and rigid body control systems such as the attitude control system, multibody dynamics, and thermal distortion effects. Another parallel objective is to utilize the experiment set-up to demonstrate optical remote sensing techniques as an alternative to other types of feedback sensors for the future design of flexible structure control systems.

Experiment activities will include the evaluation and/or the development of system identification techniques for flexible structures in support of the broader objectives of demonstrating techniques for the augmentation of structural dynamic characteristics. Such augmentation techniques might include passive damping, active control with the implementation of distributed hierarchical control methodologies, and ultimately adaptive control utilizing on-line system identification techniques.

12.3 DESCRIPTION

The experiment will incorporate remote optical sensing techniques as a practical means of instrumenting the Space Station structure. A high accuracy figure sensor such as the Spatial High-Accuracy Position-Encoding Sensor (SHAPES), which is currently under development at Jet Propulsion Laboratory, or some other high accuracy device will be utilized for sensing three dimensional displacements for selected points on the structure. An experiment scenario utilizing SHAPES is subsequently described which is representative of similar approaches which would be followed with appropriate modifications for alternative sensors.

Retroreflective targets, placed in specific locations on the structure, will be illuminated by a solid state pulsed laser source and tracked in three dimensions by SHAPES. SHAPES, along with an inertial reference unit, accelerometers, an RF data transceiver, and a power supply will be mobile as a package via the Mobile Remote Manipulator System (MRMS) during experiment operations. This mobility will allow for observations of different sets of optical targets located on the Space Station structure as well as different perspectives of a specific group of targets of interest. This flexibility will enable observations of several parts of the structure or enhance resolution of particular vibrational modes.

SHAPES is a position sensor that determines range based on time-of-flight correlations of laser pulses reflected from cooperative targets and determines bearing from the location of the target images on a CCD detector. The technology is currently under development with the goal of accommodating on the order of fifty retroreflective targets at a sampling rate of 10 Hz. Other sensors having potential application to this effort are also under development and are being reviewed.

An inertial measurement unit will be colocated with SHAPES to provide a real time reference for sensor orientation. Accelerometers will also be colocated with the MRMS instrumentation package so that translational as well as rotational sensor motion can be computationally excluded from the retroreflective target displacement data.

The MRMS instrumentation will communicate with the Space Station Information and Data Management System (IDMS) via a bidirectional RF digital data link as shown in Figure 12-1. Data representing target displacements relative to SHAPES as well as data representing the sensor inertial attitude and accelerations are provided as input to a signal processor which computes real time inertial target displacements.

These data will be stored for subsequent transmission to ground for off-line analysis and are also available as real time input for system identification algorithms and control law computations for follow-on experiment activities. Strain data for selected parts of the structure may also be available as additional input from existing sensors identified for TDMX 2072 - Spacecraft Strain and Acoustics Sensors. Signal processors and data storage hardware are assumed to be part of the Space Station IDMS.

Subsequent portions of the experiment associated with active control will close the loop with actuator signals from a control processor transmitted to the MRMS mounted actuators via the bidirectional RF data link. For the present experiment definition, these actuators will consist of momentum storage type control devices and proof mass actuators.

Initially, retroreflective target displacement-time histories characterizing the dynamic response of selected portions of the Space Station structure to various types of disturbance inputs will be observed for comparison with ground based analytical and experimental predictions. Target displacement data obtained here will also be used for off-line evaluation of system identification algorithms. Subsequently, experiment activities

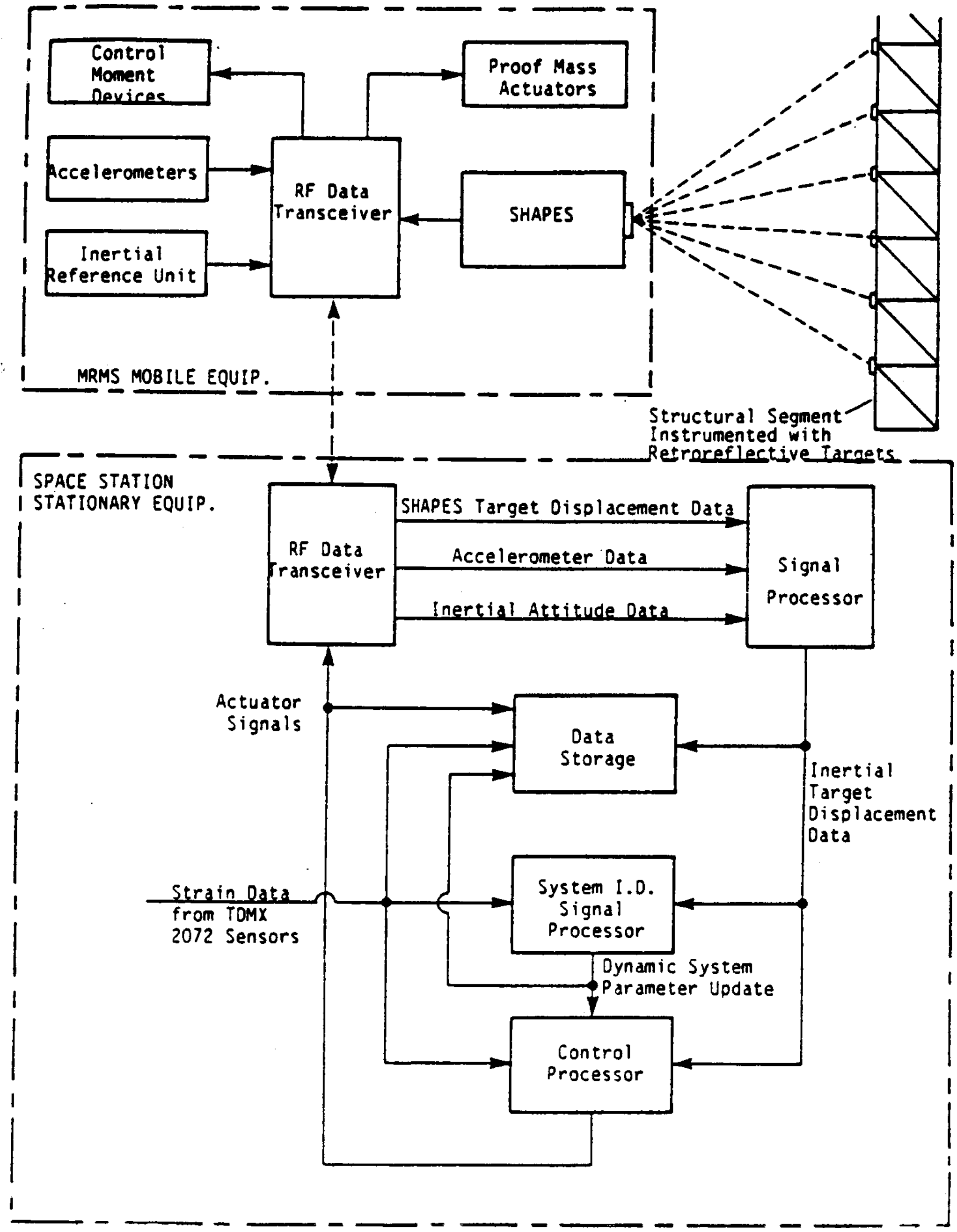


Figure 12-1. Advanced Structural Dynamics/Control Methods Experiment Concept.

will include the on-orbit evaluation of system identification and state estimation techniques providing information which will support the refinement of such techniques or the development of new approaches.

Various methods for modifying structural dynamic characteristics will be examined such as passive damping or active control utilizing SHAPES as a feedback sensor. Verification of such control schemes along with validation of on-line system identification techniques will support follow-on experiment activities such as the development and on-orbit testing of adaptive control of flexible structures.

Finally, the instrumentation will provide a means of observing changes in the dynamic characteristics of the Space Station structure due to evolutionary modifications as well as degradation from exposure to the space environment. Information of this nature will be useful in the design of growth modifications to Space Station as well as the design of other large flexible space structures.

12.4 ENVIRONMENT

The experiment will be executed in the normal Space Station working environment without restrictions on orbit parameters, orientation, etc.

12.5 EQUIPMENT

Experiment equipment can be divided into two categories which are MRMS mobile equipment and Space Station stationary equipment. Table 12-1 summarizes the equipment required for the experiment. Signal processing and data storage equipment is assumed to be available on Space Station for general use as part of the IDMS.

As described in Reference 1, SHAPES is a controls sensor suitable for the determination of the static shape and vibrational motion of large space structures and similar systems and for the determination of position and velocity in rendezvous and docking. It uses a combination of electro-optical techniques to measure the three-dimensional coordinates of targets

Table 12-1. Experiment Hardware

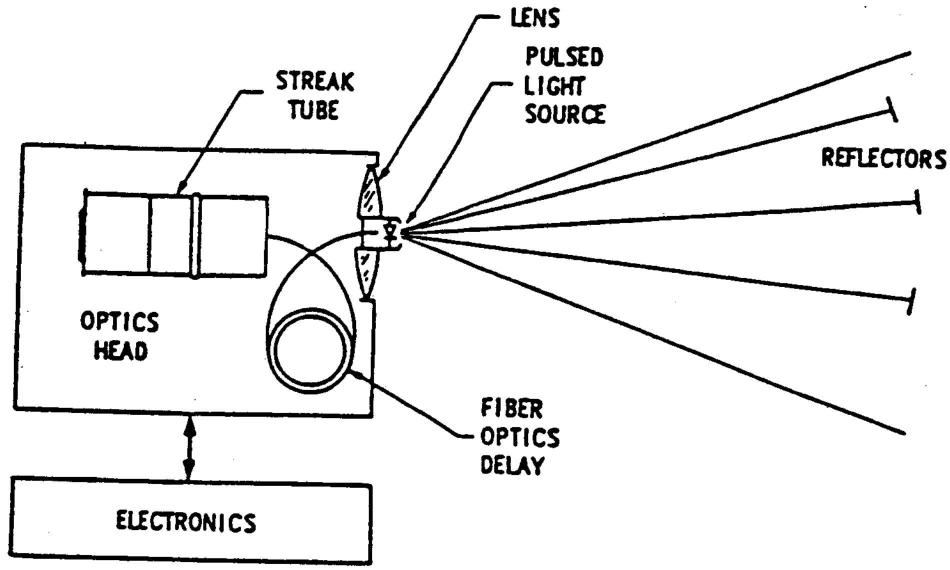
Mobile Equipment	Mass	Volume	Dimensions	Peak Power	Thermal Envir. Operational/ Transport
SHAPES	40 kg	0.028 m^3	TBD	200 W	TBD
DRIRU-II	15.9 kg	0.0205 m^3	33.5 x 26.2 x 29.1 cm	25 W	0^o to 40^oC/ -55^o to 75^oC
Triaxis Accelerometer	1.04 kg	$5.45 \times 10^{-4} m^3$	10.4 x 8.1 x 5.8 cm	1.5 W	-45 to 100^oC/ -55 to 125^oC
3 Control Moment Gyros	156.6 kg	$0.367 m^3$	89.0 x 57.1 x 24.1 cm ea	750 W	TBD
3 Proof Mass Actuators	16.5 kg	TBD	TBD	60 W	TBD
RF Data Transceiver	TBD	TBD	TBD	TBD	TBD
Power Supply	240 kg	0.14 m^3	TBD	N/A	TBD
SUBTOTALS	470 + TBD kg	0.55 + TBD m^3	N/A	1036 + TBD W	N/A
Stationary Equipment					
RF Data Transceiver	TBD	TBD	TBD	TBD	TBD
Retroreflective Targets (388)	TBD	TBD	TBD	N/A	N/A

distributed over the structure at sampling rates that are high compared to the rates at which the coordinates are changing. The technical approach is that of measuring the distance to and the direction of points on the structure from a single sensor head. Many points can be measured simultaneously from a single head without significantly increasing the complexity of the system.

The basic components of SHAPES are illustrated in Figure 12-2. A pulsed light source floods the area containing the points to be measured. The points are designated by reflectors and the light returned to the sensor head from the reflectors is imaged on a streak image tube. The tube can be operated in two modes: (1) a non-swept mode which determines the location of the images on the face of the tube and (2) a swept mode which determines the time of arrival of the returned pulses. The first mode is used to determine the directions of the reflectors from the sensor head while the second is used to determine their distances. The accuracy of the time-of-flight measurements is greatly increased by providing a reference light-pulse signal via a fiber-optics, precision-delay link and measuring the differences in the time of arrival of the pulses from the reflector and the reference pulse.

Several new technologies are utilized by SHAPES. These technologies have all been demonstrated and are in various stages of development maturity. The specific technologies are: the pulsed laser-diode light source, the fiber-optics for the delay, the integrated-optics waveguide switches for switching various delay increments, and the streak tube with a CCD readout.

The expected performance of the SHAPES system can be summarized as follows. The accuracy of the angular measurement with full field optics is one part in 10^4. With the field-compression field optics this becomes approximately one part in 10^5. The range measurement uncertainty is about 0.15 mm and is independent of distance. (The laser pulse rate is 100 MHz, therefore, range dimensions must be known to within 3 meters a priori for absolute range measurements. In this application, however, only dynamic perturbations in target positions need to be measured, therefore, initial



NOTE: Integrated-optics switches are incorporated in the delay and the readout of the streak tube is done with a CCD.

Figure 12-2. Basic Design of the SHAPES System.

knowledge of range dimensions to ±3 meters is of no consequence.) The number of multiple targets can be on the order of 50 and the data rate up to 10 target sets per second [1].

SHAPES is still under development at the present time and has not been space qualified. However, it is estimated that a flight version of SHAPES would have a mass of approximately 40 kg. and occupy a volume of 0.028 m^3. The power consumption of such a sensor is estimated to be 200 watts.

There will at times be significant motion of SHAPES with respect to inertial space which would introduce errors in the sensed target motions with respect to an inertial reference frame. Therefore, corrections will be necessary so that the true dynamics of the structure will be observed. Corrections of this nature will require instrumentation which senses the 6 degree-of-freedom motion of SHAPES.

An inertial reference unit (IRU) will be colocated with SHAPES in order to provide data representing the time variation in SHAPES attitude with respect to inertial space. These data will be utilized to correct errors in the SHAPES output due to rotational motion of the optical sensor accounting for 3 degrees of freedom.

The IRU to be utilized in this experiment is the NASA standard Dry Rotor Inertial Reference Unit (DRIRU-II) which is a high accuracy, three-axis strapdown unit capable of providing analog rate and digital incremental angle information.

The DRIRU-II can operate in either a low rate or high rate mode with respective scale factors for incremental angle outputs of 0.05 arcsec/pulse and 0.8 arcsec/pulse. The analog rate output scale factor is $12V/^{o}/s \pm 0.6V/^{o}/s$. The dynamic range for the low rate mode is ±400 arcsec/sec and $\pm 2^{o}$/sec for the high rate mode with a bandwidth of 7 Hz. The power requirement for the DRIRU-II is 25 watts at 28 VDC ± 7 VDC with a ripple content of less than 1.5 V.p-p from 1 Hz to 10 MHz. The unit occupies a volume of 0.0205 m^3 with a mass of 15.9 kg.

A triaxis accelerometer will also be colocated with SHAPES. The accelerometer data along with inertial attitude data will describe translational motion of the SHAPES with respect to inertial space. This information will be utilized to correct errors in the SHAPES output due to translational motion of the optical sensor accounting for the remaining 3 degrees of freedom.

A representative sensor for this application is the linear triaxis accelerometer manufactured by Bell Aerospace (Model No. 6471-300001) which is a self-contained device including all signal conditioning electronics within a single package. The linear dynamic range is -512 μg to +511 μg for each axis with a nominal output scale factor of 10 millivolts/μg. Output resolution is one μg and the frequency response is flat from 0.01 Hz to 20 Hz. Power requirements are 1.5 watts at $\pm$15 VDC $\pm$3% excitation. The approximate volume of the sensor is $5.45 \times 10^{-4} m^3$ with a mass of 1.04 kg.

Experiment activities related to active control or localized damping enhancement will require some means of imparting forces and/or moments to the structure, namely actuators. While it may ultimately be desirable to distribute multiple actuators in selected locations along a substructure for experimentation with multimodal and/or distributed control strategies, it will be assumed that this experiment will include actuators located at a single point only. This instrumentation will allow simplified experiments in active control of structural dynamics with various parts of the structure inasmuch as the actuators will be mobile via the MRMS. It is also assumed that passive damping experiments can be accomplished with the same apparatus by tailoring actuator control laws to simulate passive devices.

At present, it will be assumed that the momentum storage type control moment actuators will be control moment gyros (CMG's). Three single gimbal CMG's will be required to provide three axis moment control. The Sperry model M225 single gimbal CMG is assumed to be a representative component for this application. The rated torque output of each CMG is 305 N-m with a mass of 52.2 kg and power consumption of 250 W peak.

Force actuators will be assumed as linear proof mass actuators. Three such actuators will be required to provide control forces in three orthogonal directions. Such actuators are not currently available, therefore, the assumed characteristics for these devices are estimates only. Each of the linear actuators will be capable of an output of 45 Newtons, exhibit a mass of 5.5 kg, and require 20 watts of electric power during operation.

Data communications between MRMS mobile equipment and stationary equipment will be accomplished via a bidirectional RF data link. It is estimated that a data rate of 60 kbps will support the experiment. Specific hardware for this purpose has not been identified. However, it is assumed that parameters such as weight and volume for this equipment will be small compared to the totals for the remaining experiment equipment.

Power for the mobile equipment will be supplied by a rechargeable power supply to allow complete flexibility of positioning of the MRMS and experiment instrumentation. For reference purposes nickel/cadmium batteries will be assumed as the primary constituent of the energy storage device. It will also be assumed that the power supply will experience an 80% depth of discharge over 4 hours with a round trip energy efficiency of 70% and exhibit an energy storage density of 22 Wh/kg or 36600 Wh/m^3 [2]. The assumption of an 80% depth of discharge corresponds to a cycle life of approximately 1500 which will be sufficient for this application. Based on these assumptions and the total average power demand for the mobile equipment, the estimated power supply mass is 240 kg with an approximate volume of 0.14 m^3 excluding power conditioning electronics and other support equipment.

It is important to note that the demand on the Space Station power system for the experiment is not equal to the sum of the equipment demands since the rate of charging for the power supply can be much smaller than the rate of discharging during experiment operations. Indeed it could be assumed, for example, that 4 hours of experiment operation utilizing the control actuators will be followed by 20 hours of battery charging which would result in an average load of 300 watts over the charging period.

Stationary equipment dedicated to the experiment includes a counterpart to the mobile 60 kbps RF data transceiver previously described and the retroreflective targets for the SHAPES. It is assumed that signal processing and data storage needs will be accommodated by the Space Station IDMS.

Retroreflective targets will be attached to the structure at selected locations. It is envisioned that the targets will take on the functional form of corner cubes such that they will be "visible" to SHAPES from virtually any practical perspective. Due to the inherent symmetry of the Space Station structure, only half of the truss structure will be instrumented with targets. The number of targets required for the truss structure has been approximated by assuming one target per structural bay which corresponds to 288 targets. In addition, retroreflective targets will also be placed on one inboard solar panel and one outboard solar panel with each panel requiring approximately 50 targets. It is envisioned that the targets will be relatively small and lightweight, and therefore, will not impose a significant impact on weight and volume totals.

12.6 DATA ACQUISITION AND CONTROL

It is estimated that data transfer between mobile instrumentation and stationary equipment can be accommodated by a 60 kbps bidirectional data link as shown in Figure 12-1. Data transfer would include sensor data from the SHAPES, IRU, and accelerometers as well as actuator signals for the CMG's and the proof mass actuators during later phases of the experiment.

Real-time computations will be performed upon raw sensor data by a Space Station provided signal processor such that data representing retroreflective target displacements with respect to inertial space can be stored for subsequent analysis and/or analyzed in real-time by a system identification algorithm implemented on a second signal processor. One of the objectives of the experiment is to evaluate a variety of system identification algorithms with, of course, varying computational requirements. Therefore, it is premature to bound the requirements on the system identification signal processor.

Inertial target displacement data will also be utilized in a third signal processor during parts of the experiment associated with active control. The result of control law computations in the control processor will be data representing actuator signals which will close the control loop. A discussion similar to the previous one applies with regard to the unspecified requirements on the control processor.

Sensors specified in TDMX 2072, Spacecraft Strain and Acoustics Sensors, may be able to provide useful strain data for some instrumented structural segments. Data from such sensors have been shown in Figure 12-1 as input to the system identification and the control processors inasmuch as it may prove useful in evaluating specific system identification and control algorithms requiring such input.

Finally, if technology development proceeds to a sufficient level, the system identification processor operating in real-time will provide periodic dynamic system parameter updates to the control processor such that adaptive control can be realized.

12.7 DATA COMMUNICATIONS AND HANDLING

It is assumed that data transfer between the Advanced Controls Instrumentation and the Space Station data handling equipment will occur via the IDMS [3]. An optical data distribution network (ODDNet) provides data links around which the IDMS is centered. The ODDNet will be a high capacity network easily accessible anywhere in the Space Station by standard optical interface devices (ID). Distributed signal processing resources will be provided by standard subsystem data processors (SDP) built up from modular standard components to accommodate the local signal processing needs of the individual user application. It is these signal processing resources that will be used for on-orbit data manipulation.

The man/machine interface will be provided by a standard multipurpose applications console (MPAC) which can be either fixed or portable. Commands or software changes can be entered directly from an MPAC, from storage, or from the ground via TDRSS.

On-board data storage requirements are estimated to be 40 Mbytes assuming approximately 4 hours of experiment data acquisition prior to TDRSS downlink. It is anticipated that the frequency of data downlink via TDRSS will be approximately once daily. Occasional uplink of software via TDRSS will be necessary on an as-required basis.

12.8 CREW INVOLVEMENT

Tasks requiring crew involvement during this experiment can be accomplished at the task trainable skill level and will consist primarily of IVA. EVA will be required only during installation and removal of external equipment. Specific activities which will occur during the 1 year orbital mission period are subsequently described with a timeline representation in Table 12-2.

Retroreflective optical targets will be installed at selected locations on the Space Station structure by EVA. Although the specific targets have not been defined, or the structure for that matter, the installation will be assumed as a simple task requiring on the order of 16 manhours of EVA. Experiment equipment which will be mobile via the MRMS will be transported as a single package aboard the Space Shuttle and should, therefore, require minimal crew involvement. This involvement is estimated to be on the order of 1 manhour of EVA accompanied by 1 manhour of IVA for transfer to MRMS control. Installation of internal stationary equipment, namely an RF data transceiver, and the configuration of appropriate IDMS hardware for use by the experiment should require on the order of 4 manhours of IVA.

The set-up of stationary equipment will include initial check-out of the stationary data transceiver and verification of communications with the Space Station IDMS and will require approximately 1 manhour of IVA. On the order of 1 additional manhour of IVA will be required to load and check-out command, control, and computational software for experiment execution. Experiment set-up will then proceed with a 2 manhour effort required for powering up the mobile RF data transceiver and verifying essentially error-free communications with the stationary counterpart as well as initial check-out of SHAPES, the inertial reference unit, and the accelerometers.

Rough estimates for the level of crew involvement in initial experiment operations associated with system identification activities are based on the assumed scenario that observations will be taken by the mobile sensor package at approximately 20 locations on the truss structure with

Table 12-2. On Orbit Experiment Task Evaluation

FUNCTION/TASK	(MANHOURS)	METHOD	FEASIBILITY*	COMMENTS
o Install Advanced Controls Experiment Hardware	22	RMS-MRMS-EVA-IVA	2	
o Install Optical Targets	(16)	EVA	2	Targets designed for simple installation at predetermined locations.
o Remove mobile sensors from STS and transfer to MRMS control	(2)	RMS-MRMS-EVA-IVA	2	
o Install stationary data transceiver and configure IDMS hardware	(4)	IVA	2	
o Experiment Set-Up	4	IVA	2	
o Power up and check out stationary equipment	(1)	IVA	2	
o Load control software for IDMS data manipulation	(1)	IVA	2	
o Power up and check out mobile equipment	(2)	MRMS-IVA	2	

* 1-Proven, 2-Easy, 3-Hard, 4-Questionable, 5-Not Possible

Table 12-2. (Concluded)

FUNCTION/TASK	(MANHOURS)	METHOD	FEASIBILITY*	COMMENTS
o Experiment Operations	426		2	
o Data acquisition for off line system identification	(176)	MRMS-IVA	2	
o Active control experiments	(100)	MRMS-IVA	2	
o On-line system identification experiments	(50)	MRMS-IVA	2	
o Adaptive control experiments	(50)	MRMS-IVA	2	
o Data acquisition for modified structure	(50)	MRMS-IVA	2	
o Equipment Removal	4			
o Remove mobile sensor package and stow aboard STS	(2)	RMS-MRMS-EVA-IVA	2	
o Remove stationary equipment	(2)	IVA	2	

* 1-Proven, 2-Easy, 3-Hard, 4-Questionable, 5-Not Possible

multiple view directions at each location. At each location, initial alignment and target aquisition for SHAPES must be completed prior to the identification of initial conditions during a quiescent period. Various disturbance inputs can then be introduced to excite the vibrational modes of the structure for observation by the sensors. It is estimated that 2 manhours will be required to support initial alignment and data acquisition for each view direction and approximately 4 view directions at each location will be assumed leading to an estimate of 160 manhours. Similar procedures will be completed for both an inboard and an outboard solar panel requiring on the order of 16 manhours. This will increase the total crew involvement for initial system identification experiments to 176 manhours.

Active control experiments will follow utilizing the mobile actuators in a closed loop control mode in order to study the effects of localized damping enhancement. It is envisioned that these experiments would deal with a small portion of the truss structure and require on the order of 100 manhours of IVA for completion.

Further time will be allotted as indicated in Table 12-2 for more advanced experiments related to verification of on-line system identification techniques, adaptive control, and the effects of evolutionary modifications on structural dynamics.

Equipment removal will occur in reverse order from installation with the assumption that it will not be necessary to remove optical targets.

12.9 GROUND PROCESSING

Ground based analysis of experimental data will begin as soon as data are available via TDRSS downlink. Raw data will be subjected to analysis by more complicated computational algorithms so that results can be compared to those for simplified flight algorithms. Comparisons of this nature will be the basis for refinements to flight software or the development of new computational techniques.

It is anticipated that ground based efforts will continue for approximately one year beyond the termination of on-orbit acitivites.

12.10 SUPPORT EQUIPMENT

Shipping containers will be required to provide appropriate protection for launch sensitive experiment hardware such as accelerometers. It is assumed that a standard STS pallet will be modified to accommodate the storage of the mobile sensor package aboard the Space Shuttle.

12.11 INTER-EXPERIMENT COMPATIBILITY

The experiment will utilize the MRMS extensively for sensor placement during the experiment operations, therefore, scheduling of other Space Station activities requiring the MRMS must be a consideration.

The experiment will at times require that disturbances be generated to excite the vibrational modes of the structure. The magnitudes of these disturbances have not been defined; however, they may be significant with regard to other experiments having no isolation but requiring a microgravity environment.

The experiment itself requires quiescent conditions at times for the determination of the steady state structural configuration. Therefore, other experiments or Space Station activities must be considered in scheduling of these portions of the experiment.

12.12 TECHNOLOGY DEVELOPMENT ISSUES

It should be emphasized that SHAPES technology is still under development and, while there are no apparent roadblocks to hardware development, the latter has yet to be accomplished for a flight quality system useful for the Advanced Controls experiment. If a SHAPES system is available then it should prove to be an invaluable means for the practical instrumentation of the Space Station Structure; however, if it is not, some of the experiment objectives can be met with alternative sensing techniques such as photogrammetry.

The usefulness of photogrammetry for the sensing of flexible structure motions has been demonstrated on the Solar Array Flight Experiment/Dynamic Augmentation Experiment (SAFE/DAE) carried out on the Space Shuttle [4]. However, the data reduction in this case was completed post flight from video tape rather than in real time. The signal processing requirements for actual real time reduction of such raw video data into structural deflections will be very demanding with the present state of technology. Indeed, there is not a system currently in existence which possesses the required capabilities. Also, the initial target identification for the SAFE/DAE analysis was done by an operator in an interactive mode; however, for real time data handling, an automated methodology for target identification must be developed.

Experiments correlating the dynamic characteristics of the Space Station structure with model predictions or the verification of system identification algorithms, for example, could be accomplished by the off-line analysis of photogrammetric data with the present state of technology. However, experiments requiring real-time data reduction, such as automatic control experiments, would not be feasible without significant technology advances.

REFERENCES

1. McLauchlan, J. M., W. C. Goss, and E. F. Tubbs, "SHAPES: A Spatial, High-Accuracy, Position-Encoding Sensor for Space System Control Applications," Jet Propulsion Laboratory, Annual Rocky Mountain Guidance and Control Conference, AAS Paper No. 82-032, Keystone, Colorado, February 3, 1982.

2. Bechtel National, Inc., "Handbook for Battery Energy Storage in Photovoltaic Power Systems Final Report," SAND80-7022, Prepared for Sandia National Laboratories under contract No. DE-AC03-78ET26902, February, 1980.

3. NASA on B. Johnson Space Center," Space Station Reference Configuration Description," JSC 19989, Systems Engineering and Integration Space Station Program Office, August, 1984.

4. Brumfield, M. L., R. S. Pappa, J. B. Miller, and R. R. Adams, NASA Langley Research Center, "Orbital Dynamics of the OAST-1 Solar Array Using Video Measurements," AIAA/ASME/ASCE/AHS 26th Structures, Structural Dynamics, and Materials Conference, AIAA Paper No. 85-0758-CP, Orlando, Florida, April 15-17, 1985.

13.0 TDMX 2422 - THERMAL SHAPE CONTROL

CONTACT: Dr. Howard M. Adelman, Mail Stop 246
NASA LaRC (804) 865-2887

13.1 GOALS AND OBJECTIVES

Future designs of lightweight flexible surface structures such as large space antennae must maintain a highly accurate shape for acceptable performance. In most cases these structures will require some means of control to maintain high levels of shape integrity.

The goal of this technology development mission (TDM) experiment is to study the feasibility of utilizing the inherent thermal expansion characteristics of construction materials as a mechanism for controlling distortions in the shape of flexible space structures during orbital flights.

Control of distortions by thermal means will be beneficial because thermal loads are self-equilibrating and their use avoids possible drift and orientation changes associated with unbalanced forces. In addition, the stresses in a structure resulting from heat loads will be smaller than those associated with applied forces necessary to accomplish the same task.

The objective is to experimentally determine the performance of a thermally controlled test structure in maintaining a reference shape while being subjected to various shape distorting phenomena such as large temperature gradients due to nonuniform solar heating.

Verification of the concept of thermal shape control requires a long duration mission in a low-g environment. An experiment of this nature also requires a highly variable thermal load environment characteristic of low earth orbit with periodic shading of a test structure by the Space Station.

13.2 DESCRIPTION

The experiment will utilize a structural panel attached by a strut to the Space Station as a site for the experiment. The panel is expected to be approximately 25 m^2 (5 m x 5 m) in area and should be located on the

Space Station in such a way that it will receive periodic shading by the Space Station structure. This panel will contain sensors and thermal actuators which will serve the respective purposes of sensing deviations from the reference shape and providing thermal energy to create temperature distributions necessary for shape control. A sketch of the instrumented panel concept is shown in Figure 13-1.

It is envisioned that the panel will be a two dimensional truss structure comprised of graphite-epoxy members which will respond in a "quasi-static" fashion to thermal loads causing shape distortions. The response will be quasi-static compensation for shape perturbations effected by applied temperature differences due to controlled heat inputs from the thermal actuators.

Ground based modeling analyses prior to experiment execution will have established the control laws necessary for the experiment. Signal processing for real-time control computations during the experiment will be effected by Space Station information and data management resources.

Thermal actuators, temperature sensors and strain sensors will be integrated into the structural elements on the ground, prior to flight. The structure itself will then be deployed as a single package in space.

The location of actuators and sensors within the structure for optimum quasi-static shape control is a key element of the experiment design and will be a focus of preflight analyses. There are, in general, a large number of potential sites. A prime focus of this experiment is to obtain in-space data for a panel with "near-optimal" site selection to reduce the number of actuator and sensor sites. Heuristic algorithms appear to be efficient and accurate approaches for establishing this. Such analyses will be conducted during more detailed experiment definition activities.

13.3 ENVIRONMENT

The local environment for the experimental structure is the driving parameter in the sense that periodic shading from solar heating will cause deformations which the experiment will seek to alleviate by the controlled

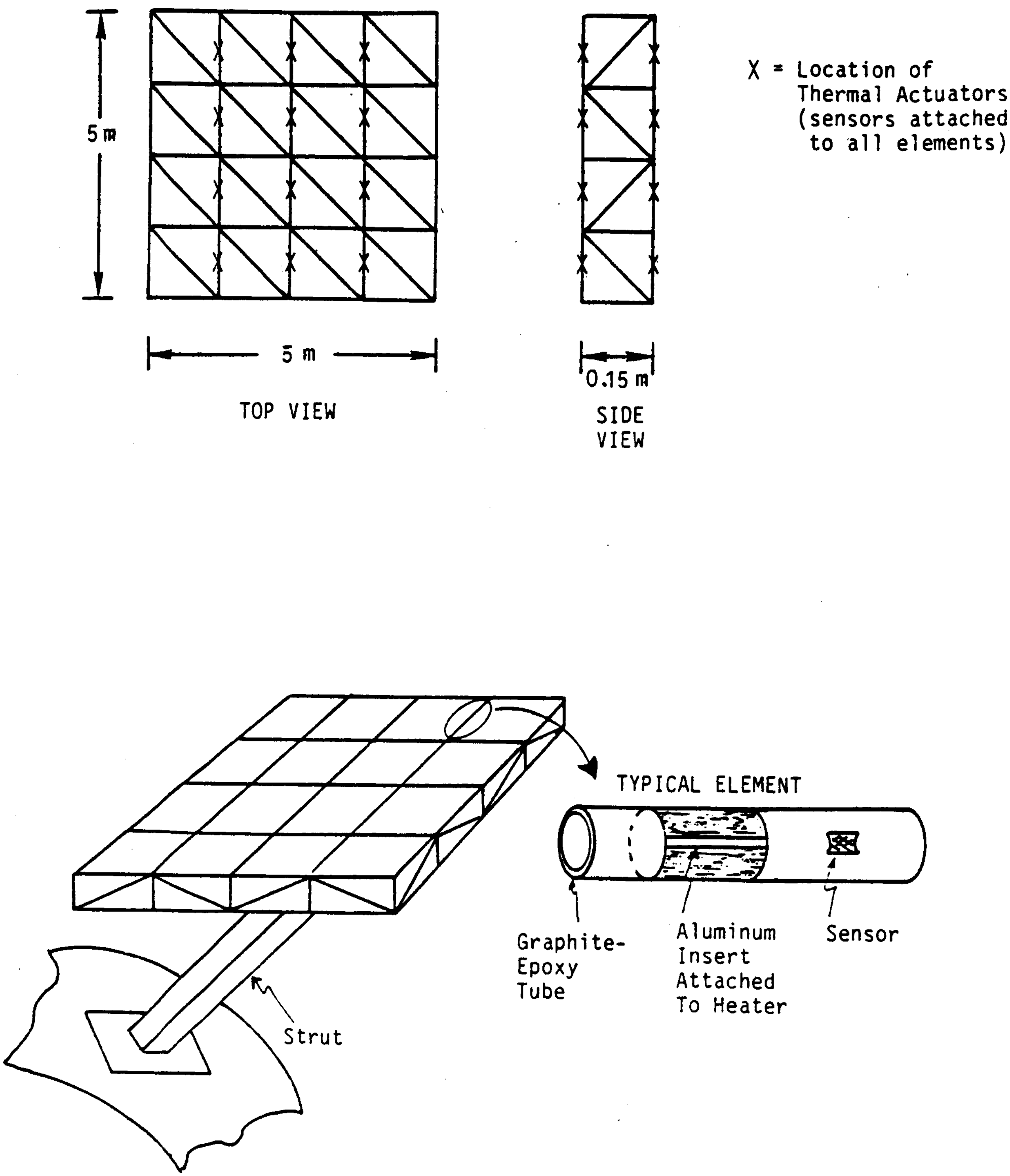


Figure 13-1. Schematic of Thermal Shape Control Experiment Components.

addition of thermal energy to the experimental panel. The low earth orbit anticipated for Space Station is suitable for this experiment since the required highly variable thermal load environment will ensue. Periodic shading of the panel by the Space Station structure as well as shading due to earth orbit will enhance the breadth of experiment results. Structural considerations for the experiment are those associated with the attachment of the panel to the Space Station. Vibrations of the Space Station structure which exhibit frequencies that are near the shape control system bandwidth must not be transmitted to the experimental panel.

13.4 SENSORS AND OTHER EQUIPMENT

Hardware requirements include strain sensors from which deformations can be determined, heat actuators, and ancillary hardware required for interfacing to the power supply and data communications system. A total of 24 thermal actuators (resistance heaters),24 temperature sensors, and 153 strain gauges will be required. These sensors and actuators will be integrated into the graphite-epoxy truss members as shown in Figure 13-1. The strain sensors must have sufficient sensitivity to identify deflections normal to the surface of approximately 1.0 mm. The electrical elements will have an output of approximately 100 W each and will be powered by Space Station d.c. power. Signal conditioning electronics and A/D converters are assumed built into the test structure as well as switching electronics for the thermal actuators. Interfaces to the power supply and data communications system will be attached to the tethering apparatus or will be an integral component of the tethering hardware.

13.5 DATA ACQUISITION AND CONTROL

The experiment is expected to last no longer than one hour during each occurrence and will require bi-directional digital data transfer during experiment operation. The number of sensors required and the sensor sampling rate will output 1 kbps of information per second over the one hour duration of the experiment. Actuator control information could easily be accommodated by a similar data rate. Onboard data storage will be required during periods between data acquisition and TDRSS downlink. This interim storage need should require on the order of 30 Mbits of onboard storage.

13.6 CREW ACTIVITIES

Crew involvement will be limited to initial construction setup and checkout of the experimental facility and subsequent disassembly. It is estimated that approximately 3 manhours during a one day period will be required to complete each of these activities. These tasks can be accomplished at the technician skill level. Table 13-1 provides a timeline which characterizes these crew activities.

13.7 GROUND PROCESSING

Data resulting from the experiment will be used to assess sensor and actuator locations for optimum control, refine and/or develop control algorithms, and verify finite element analyses used to establish local deformation/temperature relationships. Such information could be integral to the flexible structures program currently ongoing at LaRC.

13.8 THERMAL, POWER, WEIGHT, VOLUME

Section 13.4 established an estimate of 100 W for each of approximately 24 actuators for a total peak operating power of 2.4 kW. Peak conditions are expected to occur for no more than 0.2 hr. Nominal average operating power is estimated to be approximately 0.5 kW during the one hour operating period.

Table 13-1. On-Orbit Experiment Task Evaluation

FUNCTION/TASK	TIME MANMINUTES	METHOD	FEASIBILITY*
1. Unpack experimental panel/strut from Shuttle Bay	45	EVA	2
2. Deploy panel/strut and attach to station	120	EVA	3
3. Connect power and signal lines	15	IVA	2
4. Disconnect power and signal lines	15	IVA	2
5. Disconnect panel/strut from station and undeploy	120	EVA	3
6. Repack experimental panel/strut into Shuttle Bay	45	EVA	2

* 1-Proven, 2-Easy, 3-Hard, 4-Questionable, 5-Not Possible

Active thermal control of the experiment panel will establish peak temperatures on the order of 50°C. However, there are no thermal management requirements as far as Space Station systems are concerned due to the nature of the experiment. The experiment is estimated to have a mass of 100 kg with a stowable packaged size of 5 m x 2.5 m x 0.3 m corresponding to a total volume of 3.75 m^3.

14.0 TDMX 2431 - ADVANCED CONTROL DEVICE TECHNOLOGY

CONTACT: Claude R. Keckler, MS 161
NASA LaRC (804) 865-4591

14.1 GOALS AND OBJECTIVES

The goal of this technology development mission (TDM) experiment is to evaluate selected advanced momentum and/or energy storage devices (e.g., second generation magnetically suspended momentum rings) like the Rockwell/Draper and Sperry Annular Momentum Control Device (AMCD) conceptual designs for application to an AMCD Combined Control and Energy Storage System (ACCESS) for the Space Station. Potential advantages of the AMCD over conventional momentum storage devices include optimum momentum to mass ratios, improved life characteristics, and the capability of large radial dimensions (hence, large momentum storage capability). For energy storage, the rim shape allows full utilization of the filament strengths of composite materials by allowing a unidirectional layup.

The proposed uses of the AMCD are to provide attitude control and/or energy storage for the Space Station and related large space structures. Prior to these applications, it is necessary to resolve the problems associated with development of this technology. The objectives of this experiment are to evaluate operational constraints and environmental effects (mainly zero gravity) on elements of the AMCD and to validate selective systems concepts prior to committing to a specific set of operational hardware. The primary elements of this technology to be focused upon include such items as:

a) Magnetic suspension
b) Composite rotors
c) Energy conversion concepts (electrical/rotational)
d) Control laws

14.2 DESCRIPTION

Advanced devices comprising the experimental ACCESS system will be provided as a selectable alternative to existing IOC attitude control and energy storage equipment after initial evaluation phases of the experiment are completed. The ACCESS equipment performance will be evaluated in three experimental phases:

Phase I: Zero and Low Speed experiments to validate magnetic bearing actuator characteristics.

Phase II: Evaluation of the AMCD's efficiency and its various modes of operation.

Phase III: High Energy experiments for control and energy storage system evaluation.

The three experimental phases will be carried out with the ACCESS hardware installed in an operational configuration on board the Space Station.

14.3 ENVIRONMENT

All in-flight experiment activities will occur on board the Space Station with the AMCD components co-located with the Space Station control moment gyros. Structural requirements imposed by the attachment of AMCD's to the Space Station are similar to those for the control moment gyros in that very large attitude control moments will be generated.

The experiment will be conducted in a normal Space Station working environment without constraints on orbit parameters or spacecraft orientation.

14.4 HARDWARE

The existing ACCESS hardware conceptual design is a result of a study completed by Rockwell International and the Charles Stark Draper Laboratory, Inc. [1]. The Rockwell/Draper design will be used as a basis for defining the major components comprising the TDM experiment equipment as subsequently described.

The ACCESS concept is based on the storage of kinetic energy in mechanical rotors which in turn provides an abundance of angular momentum which can be utilized for spacecraft attitude control. The overall system design allows storage and extraction of energy as well as attitude control of the Space Station with negligible interaction between the two modes of operation.

The initial ACCESS configuration will consist of a planar array of five AMCD's along with the supporting power control electronics and signal processing equipment. These major components of the ACCESS system will be preassembled on the ground and transported in a single flight to the Space Station. The AMCD's will be installed in the vicinity of the existing Space Station control moment gyros for the duration of the experiment.

The initial ACCESS system design will be capable of supporting a bus power level of 75 kW during orbital eclipse of 36 minutes with full re-charging occurring during the remaining 57 minutes of the orbital period. The projected round trip energy conversion efficiency is 85% with magnetic bearing losses contributing less than 1% of the total losses.

In the attitude control mode of operation, the IOC system exhibits a momentum transfer capability of 36600 N-m-s and is designed to support the attitude control system bandwidth of 0.05 Hz.

The following is a summary of the major system components with a total estimated mass based on a projected energy density of 22 Wh/kg.

- Annular Momentum Control Devices (5)
- Detached AMCD Signal Processors (5)
- ACCESS Master Signal Processor (1)
- Power Control Electronics (1)

Total System Mass = 4000 kg

The Rockwell/Draper AMCD conceptual design is depicted in Figure 14-1. The AMCD rotor is suspended by a spherical large angle magnetic bearing (LAMB) which serves the dual purposes of a low loss rotor spin bearing, as well as a two degree of freedom gimbal system. The spherical LAMB allows gimballing up to 23^{o} with a control torque capacity of 300 N-m/unit. The energy storage capacity of each AMCD is 13.2 kWh at a 75% depth of discharge. The discharge power level for each AMCD is 21.9 kW over the orbital eclipse period.

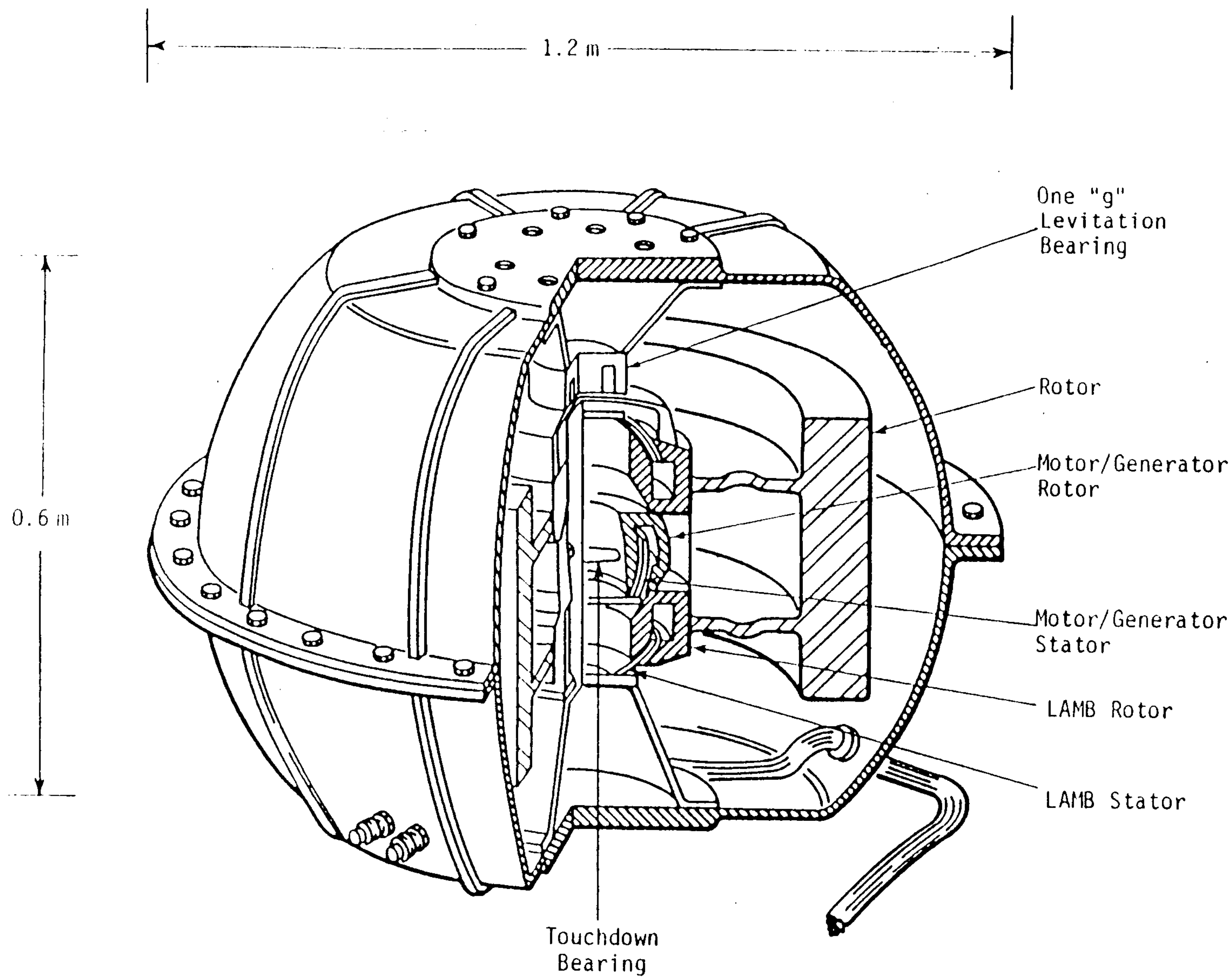


Figure 14-1. AMCD Design Concept.

An inner control loop maintains five degree of freedom control over the magnetic suspension of the rotor while an outer loop controls the gimbal position. Internal sensors, such as proximeters and flux sensors are inherent to each AMCD, and provide feedback information for the control system signal processors as well as performance and health status data. Detached signal processors functioning as individual controllers will be required for each AMCD while an ACCESS master signal processor will control system operation. A levitation bearing system is included in the design so that extensive preflight testing is possible in a one "g" environment.

The power control electronics, utilizing MOSFET semiconductors, will manage the storage and extraction of energy from the ACCESS system with minimal impact on the attitude control function. The ACCESS energy storage system will be operated in parallel with existing Space Station IOC energy storage and power control equipment so that an alternative means of energy storage will be available in the event of an ACCESS system failure.

A resistive load device will be required for the experiment so that the ACCESS depth of discharge can be controlled. It is envisioned that such a device would be located on the exterior of the Space Station structure with radiative heat rejection apparatus capable of dissipating 75 kW. Such a device has not been defined; therefore, in light of the assumption that equipment of this nature would present only a small perturbation on the total impact of this experiment, it will be neglected for the remainder of the experiment definition.

Additional sensors to characterize system performance will not be required for the experiment inasmuch as the necessary power system voltage and current sensors are an integral part of the power control electronics. The attitude control mode of operation will, however, require Space Station attitude and rate information which will be available from the existing guidance and control system in a read only mode of communication.

14.5 DATA ACQUISITION AND CONTROL

Data acquisition during the course of the experiment will be divided into three phases corresponding to the three experiment phases previously described. It is estimated that data acquisition rates will be on the order of 20 kbps.

The first phase, consisting of zero and low speed experiments, will be a system identification effort consisting of obtaining response data in order to define and incorporate the appropriate transfer functions for the spherical LAMB's. Data acquisition for each AMCD will require approximately five minutes for each of the five degrees of freedom. The total data storage requirement will be approximately 10 Mbytes of 16 bit words stored temporarily until TDRSS is available for downlink.

The second phase will evaluate the performance of AMCD's as both energy storage devices as well as attitude control actuators in order to define the necessary control laws for integrated system operation. Data will be acquired which will characterize the energy storage/extraction characteristics of the ACCESS system during full charge/discharge cycles. Data will also be acquired to determine actuator characteristics over the full range of rotor speed. Temporary data storage requirements for this phase of the experiment will be approximately 12 Mbytes of 16 bit words. The third phase will evaluate the combined function operation of the ACCESS system. Initially, system data will be monitored and stored continuously through two orbital periods requiring 6 Mbytes of 16 bit storage. After the initial evaluation, autonomous operation is expected with periodic sampling of data and monitoring of health status requiring only a small amount of data storage with periodic TDRSS data dumps to ground. Occasional modifications of control software, due to ground analysis of data, may be implemented via TDRSS uplink.

14.6 DATA COMMUNICATIONS AND HANDLING

It is assumed that data transfer between the ACCESS hardware and the Space Station data handling equipment will occur via the information and data management subsystem (IDMS) [2]. An optical data distribution network (ODDNet) provides data links around which the IDMS is centered. The ODDNet will be a high capacity network easily accessible anywhere in the Space Station by standard optical interface devices (ID). Distributed signal processing resources will be provided by standard subsystem data processors (SDP) built up from modular standard components to accomodate the local signal processing needs of the individual user application.

The man/machine interface will be provided by a standard multipurpose applications console (MPAC) which can either be fixed or portable. Commands or software changes can be entered directly from an MPAC, loaded from storage, or loaded from the ground via TDRSS. Experiment data, health status information, etc. from the ACCESS hardware will be available to the crew through MPAC communications. Periodic health status checks and evaluation of experiment data will be performed on-board but with minimal crew involvement.

In addition to the temporary data storage requirements described earlier, there will be a modest requirement for permanent storage of command and control software.

A block diagram representing the experiment configuration and data communications concept is shown in Figure 14-2.

14.7 CREW ACTIVITIES

Tasks requiring crew involvement during the execution of this experiment can be accomplished at the technician skill level. Specific activities which will occur during the 365 day orbital mission period are subsequently described with a timeline representation given in Table 14-1.

The ACCESS hardware will be removed from the shuttle cargo bay and installed on the Space Station platform as soon as possible after shuttle

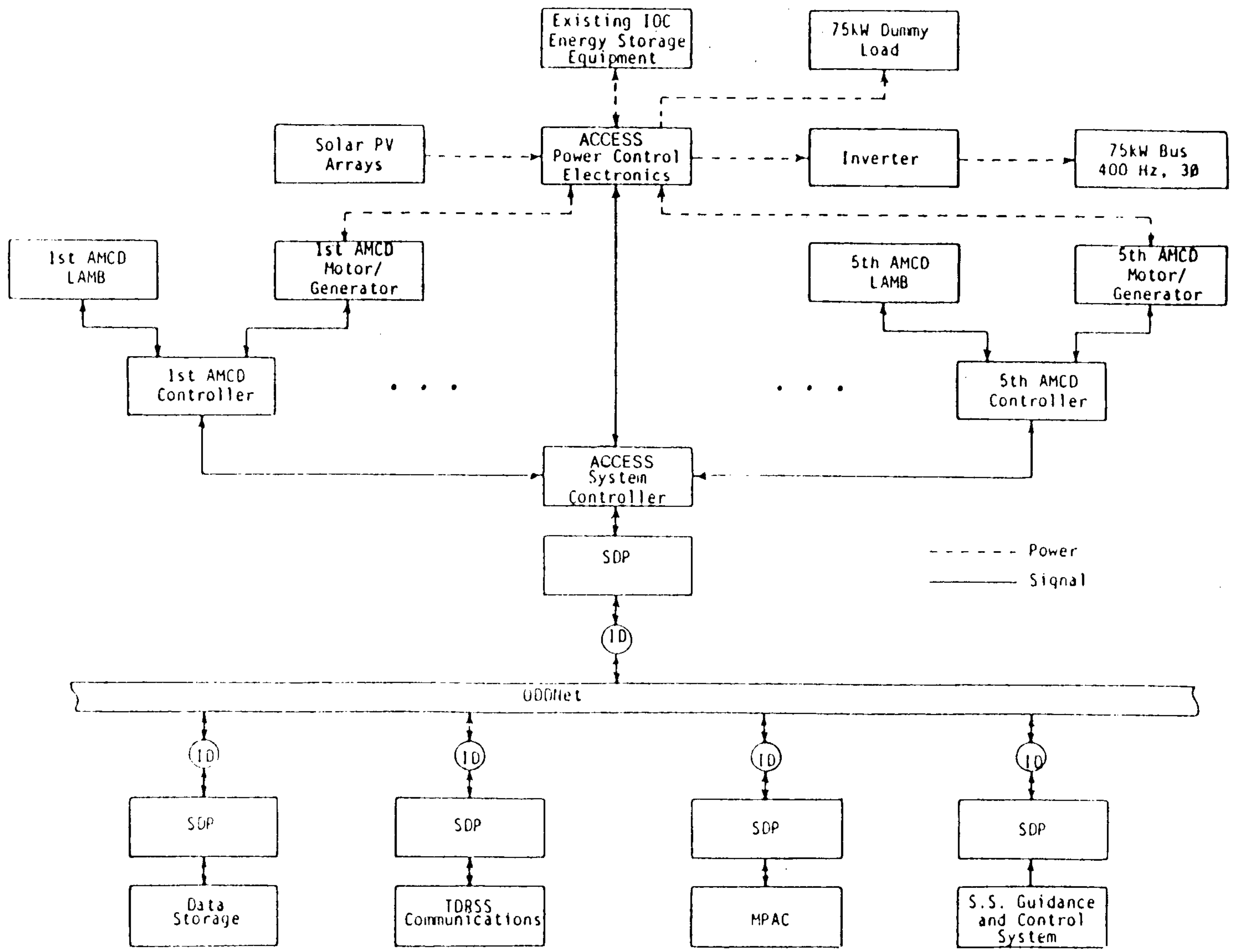


Figure 14-2. ACCESS Experiment Configuration and Data Communications Concept.

ORIGINAL PAGE IS OF POOR QUALITY

Table 14-1. TDMX 2431: On-Orbit Experiment Task Evaluation

FUNCTION/TASK	TIME MANMINUTES	METHOD	FEASIBILITY*	COMMENTS/ASSUMPTIONS
• Install ACCESS Experiment Hardware	1260	EVA-RMS-MRMS-IVA	2	Major experiment hardware components preassembled on the ground
• Remove five AMCD's and supporting electronics equipment from STS cargo bay	(126)	EVA-RMS-IVA	2	
• Transfer equipment to MRMS	(63)	EVA-RMS-MRMS-IVA	2	
• Transport equipment to the installation site near the space station CMG's.	(126)	EVA-MRMS-IVA	2	
• Position and install five AMCD's	(630)	EVA-MRMS-IVA	2	
• Install and complete inter-connections for five detached signal processors and the ACCESS system controller	(126)	EVA-IVA	2	Assume pre-existing Space Station wiring suitable for system inter-connections where appropriate
• Install and complete inter-connections for the ACCESS power control electronics	(189)	EVA-IVA	2	

*1-Proven, 2-Easy, 3-Hard, 4-Questionable, 5-Not Possible.

Table 14-1. TDMX 2431: On-Orbit Experiment Task Evaluation (Continued)

FUNCTION/TASK	TIME MANMINUTES	METHOD	FEASIBILITY*	COMMENTS/ASSUMPTIONS
• Experiment set-up, turn on equipment and verify data communications links and equipment health status	120	IVA	2	
• Phase I, zero and low speed experiments	600	IVA	2	
• Load control software	(60)	IVA	2	
• Initiate and monitor a preprogrammed test sequence	(270)	IVA	2	Data acquisition for determination of spherical LAMB transfer functions.
• Downlink experiment data and repeat the test sequence	(270)	IVA	2	Validation of important data
• Phase II, determination of energy storage and attitude control characteristics	1080	IVA	2	
• Load AMCD control software as modified based on analysis of Phase I data	(60)	IVA	2	

*1-Proven, 2-Easy, 3-Hard, 4-Questionable, 5-Not Possible.

Table 14-1. TDMX 2431: On-Orbit Experiment Task Evaluation (Continued)

FUNCTION/TASK	TIME MANMINUTES	METHOD	FEASIBILITY*	COMMENTS/ASSUMPTIONS
• Initiate and monitor pre-programmed test sequence to evaluate charge/discharge cycle performance	(450)	IVA	2	Test sequence may require interactive crew participation.
• Initiate and monitor pre-programmed test sequence to evaluate system control torque characteristics	(570)	IVA	2	
• Phase III, high energy evaluation of integrated dual function system operation	1780	IVA	2	
• Initiate autonomous operation of the ACCESS system and monitor for two orbital cycles	(22)	IVA	2	Periodic dump of stored experiment data to ground via TDRSS initiated by ground interrogation.
• Continue periodic health status checks at weekly intervals for the duration of the experiment	(1560)	IVA	2	Requires 30 man-min per week
• Incorporate modifications to control software as they evolve from ground analysis of experiment data	As Required	IVA	2	

*1-Proven, 2-Easy, 3-Hard, 4-Questionable, 5-Not Possible.

Table 14-1. TDMX 2431: On-Orbit Experiment Task Evaluation (Concluded)

FUNCTION/TASK	TIME MANMINUTES	METHOD	FEASIBILITY*	COMMENTS/ASSUMPTIONS
• Power down a preparation for removal of experiment equipment	30	IVA	2	
• Remove ACCESS experiment hardware	1230	EVA-RMS-MRMS-IVA	2	
• Disconnect and remove ACCESS power control electronics	(189)	EVA-IVA	2	
• Disconnect and remove the ACCESS system controller and five detached AMVD signal processors	(126)	EVA-IVA	2	
• Remove five AMCD's	(474)	EVA-MRMS-IVA	2	
• Transport experiment equipment to RMS acquisition range	(126)	EVA-MRMS-IVA	2	
• Transfer equipment to RMS control	(63)	EVA-RMS-MRMS-IVA	2	
• Position and secure experiment equipment within the STS cargo bay	(252)	EVA-RMS-IVA	2	

*1-Proven, 2-Easy, 3-Hard, 4-Questionable, 5-Not Possible.

docking. Major components of the ACCESS system, i.e., AMCD's and electronics packages, will be preassembled on the ground to simplify on-orbit installation efforts. The actual crew involvement to complete the instal lation of experiment hardware including five AMCD's, control electronics, and electrical interconnections is estimated to consist of 12 manhours of EVA and 9 manhours of IVA.

The initial experiment set-up will require 2 manhours of IVA to power up the ACCESS equipment, verify data communications links, and to evaluate the initial system health status. The first phase of experiment execution consisting of zero and low speed data acquisition will require 10 manhours of IVA. The second phase, consisting of the determination of energy storage and attitude control characteristics of the ACCESS system, will require 18 manhours of IVA.

The third phase, consisting of high energy integrated system experiments, will continue over the remainder of the 365 day orbital mission with the ACCESS system at times providing the actual energy storage and attitude control requirements for normal space station operation. After initial monitoring of system operation, the crew involvement in Phase III will be limited to periodic health status checks occurring approximately once weekly with occasional unscheduled implementations of software modifications which may evolve due to ground analysis of experiment data. Sampled performance data from Phase III will be stored on board the Space Station until periodic data dumps via TDRSS return service are initiated from the ground. Phase III will require a total of 30 manhours of IVA excluding unscheduled activity.

Removal and storage of experiment hardware aboard the shuttle at the end of the 365 day mission will occur in reverse order of installation and will require 12 manhours of EVA along with 9 manhours of IVA.

14.8 GROUND PROCESSING

A two man-year long term ground processing effort will begin with extensive analysis of ACCESS system data obtained from preflight testing in a one g environment. Efforts will continue with the early analysis of on-orbit experiment data leading to modifications in control software for optimum performance in the Space Station environment. A data receiving station will be required to accept and store the telemetered experiment raw data on magnetic tape which will be formatted and time tagged. Finally, a complete analysis of experiment data along with a feasibility determination and a technology definition will be completed and documented in a final report as soon as possible after experiment termination.

A destructive analysis of the ACCESS hardware will be completed after the orbital mission for the purpose of gaining insight into the survivability of such equipment in a space environment while having been subjected to long term cyclic operation. The hardware analysis will be documented in a final report completed six months after post flight hardware availability and will require a two man-year level of effort.

14.9 SUPPORT EQUIPMENT

Shipping containers will be required to provide appropriate protection for launch-sensitive experiment components. An STS pallet will be necessary for transport of the ACCESS hardware aboard the space shuttle. Ground test equipment will be required in order to verify the pre-launch and post flight status of the system components.

14.10 INTER-EXPERIMENT COMPATIBILITY

The ACCESS experiment will function independently of other Space Station experiments although consideration should be given to total Space Station power requirements during testing of the energy storage mode of operation.

REFERENCES

1. Oglevie, R. E. and D. B. Eisenhaure: "Integrated Power and Attitude Control Systems for Space Station." AIAA-85-0358, AIAA 23rd Aerospace Sciences Meeting, January 14-17, 1985, Reno, Nevada.

2. NASA Lyndon B. Johnson Space Center: "Space Station Reference Configuration Description." JSC 19989, Systems Engineering and Integration, Space Station Program Office, August, 1984.

15.0 TDMX 2432 - ADVANCED EXPERIMENT POINTING AND ISOLATION DEVICES

CONTACT: Claude R. Keckler, MS 161
NASA LaRC (804) 865-4591

15.1 GOAL

The goal of this technology development mission (TDM) experiment is to establish the technology necessary for the successful completion of missions requiring highly accurate and stable experiment pointing and micro-gravity environments on board large, flexible space vehicles such as the Space Station.

This TDM experiment will capitalize on the technology developed under the joint NASA/Sperry Flight Systems' Annular Suspension and Pointing System (ASPS) program. This program has produced a device which has undergone thorough analyses and laboratory evaluation. Results of these tests and of associated simulations afford optimistic but confident predictions of the device's on-orbit performance.

15.2 OBJECTIVES

The objectives of this experiment are to provide: 1) subarcsecond pointing and long-term stabilization for experiments dedicated to stellar, solar, and terrestial observations, and 2) a micro-gravity environment for the support of acceleration-sensitive industrial processes and developments. Another, broader objective of this TDM experiment is to evaluate the performance of elements and/or subsystems of an ASPS-like device during long term exposure in space.

15.3 DESCRIPTION

Auxiliary pointing system concepts, based on the ASPS program technology, will be defined and evaluated on the Space Station. The associated experiments will validate the capability of these concepts for providing the high accuracy pointing and stability required by various scientific and commercial payloads in the presence of environmental disturbances arising from spacecraft operations, crew motions, and other on board equipment. The success of these concepts will allow the conduct of very precise experiments without a need for interrupting other Space Station system or payload operations.

In addition, the magnetic suspension system technology that has evolved from the ASPS program will be instrumental in the definition of advanced micro-gravity concepts for the isolation of acceleration-sensitive experiments, incompatible modules, or segmented mirror isolation and control.

The techniques and concepts established in the development of these systems will be directly applicable to systems capable of performing critical alignment functions such as those required for large multi-mirror optics or laser ranging devices.

The configuration of the hardware that has resulted from the ASPS program has been described in a number of reports, briefings, and scientific papers. In particular, the 1982 paper by Keckler and the 1983 paper by Keckler and Hamilton present very concise descriptions of the ASPS device [1, 2].

According to reference 2, "the ASPS is a modular device consisting of three state-of-the-art gimbals for coarse pointing and a magnetic suspension system for load isolation and precision pointing. Existing brushless d.c. motor designs are utilized in the gimbals to supply the three-axes coarse pointing and maneuvering capability of the ASPS. To facilitate the transfer of power and data to and from the payload across the rotating interfaces, the gimbals are equipped with flex capsules, which consist of flat, flex wire cables looped between two concentric cylinders. Resolvers are used for motor commutation and gimbal position information. Each gimbal motor also incorporates a backup winding which permits restowing of the pointing system and its payload for safe return in cases of primary component failures."

"The magnetic suspension or vernier system is mounted on top of the gimbals to provide vibration isolation as well as the high accuracy pointing and stabilization (approx. 1/100 of an arcsecond) of the payload. The experiment package is attached to the payload plate which is supported by six magnetic bearing actuators (MBA's). This actuator complement consists of three (3) axial and three (3) tangential MBA's. The axial set of MBA's provides control over the translations along the payload line-of-sight (LOS) and the rotations about the two axes perpendicular to the LOS (payload transverse axes). The tangential actuators effect control over the payload/payload plate translations along the transverse axes, as well as rotation about the experiment's LOS. The servo loops for these MBA's are closed with noncontacting, vibrating, quartz crystal force sensors, and temperature insensitive position sensors measuring the location of the payload plate within the magnetic bearing gaps."

"Experiment control is effected based on attitude information obtained directly from payload optic and/or from sensors located adjacent to the payload on the suspended mounting plate."

"Power for the experiment is transferred to the levitated portion of the vernier via a rotary transformer, while commands and data are transmitted through bidirectional, high rate (approx. 30 Megabits per second) optical data links."

An experiment configuration proposed by Keckler provides a point of departure for the understanding of the physical and support requirements of the TDM experiments [3]. Reference 3 describes this experiment entitled "Vibration Isolation Technology Experiment" which has been proposed for the STS.

15.4 ENVIRONMENT

15.4.1 Orbit Parameters

Specific orbit parameter requirements are not essential to the ASPS experiment since the objective is to determine the effectiveness of the system in providing subarcsecond pointing accuracy and vibration isolation for an experiment payload. If a functional experiment payload, versus a dummy payload, is defined in the future, then orbit parameters to satisfy payload requirements may become necessary.

15.4.2 Location

The experiment will be located on the Space Station with the ASPS and a structural interface attached directly to the Space Station structure. The specific location along the structure is not critical, but should allow an adequate field of view for solar or stellar pointing portions of the experiment and should minimize the physical obstructions which would interfere with the pointing motions of the ASPS.

During the experiment the ASPS will function in a normal Space Station operating environment in order to determine the ability of the system to maintain subarcsecond pointing accuracy and vibration isolation while normal activities are occurring elsewhere on the station. There will also be an evaluation phase in which disturbances are purposely introduced in order to characterize the performance of the ASPS.

15.4.3 Orientation

The current ASPS design provides gimbal excursions of $\pm 100^{\circ}$ about the elevation axis, $\pm 60^{\circ}$ about the lateral axis, and $\pm 190^{\circ}$ about the roll axis. These motions are only limited by adjustable mechanical stops, and thus allow for large payload viewing angles. Total viewing angles will be established by the location of the ASPS on the Space Station, proximity and nature of adjacent equipment and other field of view obstructions. Since these items are as yet undefined, total allowable viewing envelopes cannot be established but are, for the definition of this experiment, assumed to be equal to those resulting from the available ASPS gimbal freedom defined above.

15.4.4 Structural Considerations

The structural attachment to the Space Station must accommodate the mass of the ASPS, the maximum size payload, and the mounting assembly under worst case loading conditions.

At specified times during the course of the experiment, the mounting assembly will be subjected to the effects of a broad band "shaker" type disturbance generator. The magnitude of such disturbances has not been specified, however, the estimated bandwidth will not exceed 200 Hz.

15.5 EQUIPMENT

15.5.1 Hardware

The ASPS, including the magnetically suspended vernier pointing system, three coarse pointing gimbals, and control electronics, is conceptually depicted in Figure 15-1. The payload plate is approximately 1.0 meter in diameter and the local vertical dimension of the system is approximately 1.2 meters. The approximate mass of the device as shown is 240 kg.

Viewing angle range available from the three gimbal system is set by adjustable mechanical stops to the following values:

elevation - $\pm$ 100°
lateral - $\pm$ 60°
roll - $\pm$ 190°

The Vernier pointing system has an operating range, as determined by the magnetic suspension gaps of $\pm 0.75^{\circ}$ about axes normal to the payload line-of-sight (LOS), and of $\pm 0.5^{\circ}$ about the payload LOS.

Coarse pointing gimbal position information is available from resolvers while fine pointing position information is available from sensors on board the payload plate. The control electronics for the vernier pointing system as well as the coarse pointing gimbals are self-contained and are an integral part of the ASPS.

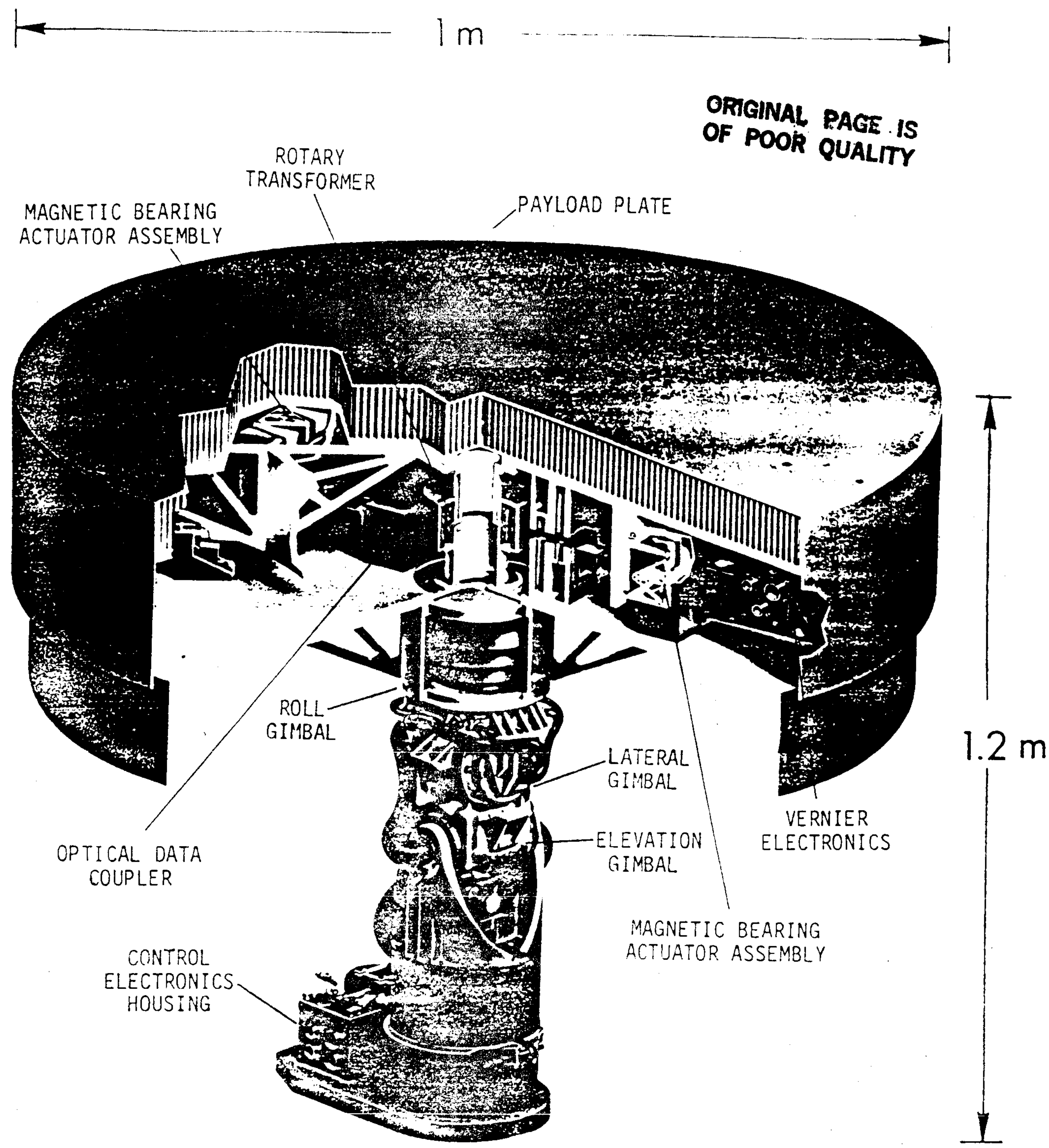


Figure 15-1. Annular Suspension and Pointing System Concept.

The experiment payload for the ASPS has not been defined at the present time. The payload can be either a functional experiment operating simultaneously with the ASPS experiment or it can be a dummy payload mass. Requirements on the payload are very relaxed since performance projections suggest that the ASPS will accomodate a range of payload mass between 300 kg and 7500 kg with a center of mass offset of up to 3.5 meters from the payload plate. A payload mass of 1000 kg will be assumed at the moment for the purpose of completing the experiment definition.

A mounting assembly will be required to provide the structural interface between the ASPS and the Space Station. Although a candidate design for such a structure has not been identified, it is envisioned that it will utilize tie down points inherent to one of the bay sections comprising the main structure of the Space Station. The mounting assembly must not exhibit a fundamental natural frequency of less than 25 Hz so that it will function as a rigid attachment. A mass of 150 kg will be assumed for the mounting assembly for purposes of completing the experiment definition.

A three-axis disturbance generator will be attached to the mounting assembly in order to simulate spacecraft vibrations for the purpose of experimentally determining the degree of vibration isolation and the pointing stability realized by the ASPS payload. The required magnitude of the disturbances has not been determined; however, it is estimated that the required bandwidth will not exceed 200 Hz. The actual hardware comprising such a disturbance generator has not been identified and, for the intent of this document, it will be assumed that the power requirements and mass characteristics of such equipment will present a small impact on the overall experiment totals and will therefore be neglected.

Payload sensors will also be required as part of the experiment package to provide both feedback information for the ASPS controller as well as data to characterize the performance of the system. Candidate sensors for this application are subsequently described with some general characteristics summarized in Table 15-1.

Table 15-1. TDMX 2432: Experiment Hardware

EQUIPMENT	MASS	VOLUME	DIMENSIONS	PEAK POWER	AVE. POWER	THERMAL ENVIRONMENT (OPERATIONAL/ TRANSPORT)
ASPS	240 kg	Approx. 0.27m^3	1m dia. X 1.2 m (See Figure 15-1)	997 W	Approx. 250 W depending on conditions	-10^{0} to 60^{0}C/ -65^{0} to 125^{0}C
ASPS Information Panel	4.5 kg	0.013 m^3	17.8 X 48.3 X 15.2 cm	7 W	7 W	
ASPS Mounting Assembly	150 kg	TBD	TBD	N/A	N/A	
Experiment Payload	1000 kg	TBD	TBD	TBD	TBD	TBD
Triaxis Accelerometers (2)	2.08 kg	10.9×10^{-4} m^3	(2) X 10.4 X 8.1 X 5.8 cm	3.0 W	3.0 W	-45^{0} to 100^{0}C/ -55^{0} to 125^{0}C
Star Tracker	5.45 kg	7.46×10^{-3} m^3	16.5 X 17.8 X 25.4 cm	10 W	10 W	
Sun Sensor Unit	1.8 kg	3.6×10^{-4} m^3	TBD	TBD	TBD	22^{0} to 28^{0}C/ -15^{0} to 50^{0}C
Sun Sensor Electronics Package	0.8 kg	1.5×10^{-3} m^3	TBD	2.2 W	2.2 W	5^{0} to 25^{0}C/ -15^{0} to 50^{0}C
DRIRU-II	15.9 kg	0.0205 m^3	33.5 X 26.2 X 29.1 cm	25 W	25 W	0^{0} to 40^{0}C/ -55^{0} to 75^{0}C
TOTALS	1421 kg	0.314+TBD m^3		1044+TBD W	297+TBD W	

Accelerometers will be required at the base of the system and on the payload plate to measure accelerations for three orthogonal axes in order to evaluate the microgravity environment provided by the ASPS during this mode of operation. A typical candidate sensor, such as the linear triaxis accelerometer manufactured by Bell Aerospace (Model No. 6471-300001), should be a self-contained device including all signal conditioning electronics within a single package. Using the Bell Aerospace unit as a model in this definition, the linear dynamic range is -512 μg to +511 μg for each axis with a nominal output scale factor of 10 millivolts/μg. Output resolution is 1 μg and the frequency response is flat from 0.01 Hz to 20 Hz. Power requirements are 1.5 watts at $\pm$ 15 VDC $\pm$ 3% excitation. The approximate volume of the sensor is 5.45×10^{-4} m^3 with a mass of 1.04 kg.

A high accuracy star sensor will be required to provide attitude information during the fine pointing mode of operation. Such a sensor has been specified by NASA for another application and has the following characteristics. It is capable of searching through its field of view (FOV) to detect 3 to 10 stars at the selected threshold or brighter. The outputs are identification magnitude and position coordinates within an accuracy of 1 arcsecond. The maximum tracking rate is 400 arcseconds/second with output data updated at 1 second intervals. Power requirements are 10 watts at 28 VDC $\pm$ 7 VDC. The approximate volume of the star tracker is 7.46×10^{-3} m^3, excluding the lens baffle, and the mass is 5.45 kg.

A solar sensor will be required to provide pointing information during solar pointing portions of the experiment. The fine pointing sun sensor used previously on the Solar Maximum mission is accurate to within 5 arcseconds. The solar detection FOV is 4^{o} along two orthogonal axes while the linear tracking FOV is 1^{o}. The outputs are analog with a scale factor of 0.5 mv/arcsecond over the linear tracking FOV or digital with a resolution of less than 0.15 arcsec over a 20 arcmin. FOV. The sensor bandwidth is 20 Hz and the power requirement is 2.2 watts at 28 VDC $\pm$ 7 VDC. The sensor unit exhibits a volume of 3.6×10^{-4} m^3 with a mass of 1.8 kg including a mounting bracket. A separate electronics package occupies 1.5×10^{-3} m^3 with a mass of 0.8 kg.

An inertial reference unit (IRU) will be required to provide attitude and rate information during the experiment. The NASA standard Dry Rotor Inertial Reference Unit (DRIRU-II) is a high accuracy, three-axis strapdown unit which provides analog rate and digital incremental angle information. The DRIRU-II can operate in either a low rate or high rate mode with respective scale factors for incremental angle outputs of 0.05 arcsec/pulse and 0.8 arcsec/pulse. The analog rate output scale factor is $12V/^{o}/s \pm 0.6$ $V/^{o}/s$. The dynamic range for the low rate mode is ± 400 arcsec/sec and $\pm 2^{o}$/sec for the high rate mode with a bandwidth of 7 Hz. The power requirement for the DRIRU-II is 25 watts at 28 VDC $\pm$ 7 VDC with a ripple content of less than 1.5 Vp-p from 1 Hz to 10 MHz. The unit occupies a volume of 0.0205 m^3 with a mass of 15.9 kg.

15.6 DATA ACQUISITION AND CONTROL

Data and command transfer across the ASPS magnetic suspension gap is via a 30 Mbps bidirectional optical data link for the ASPS experiment sensors as well as functional experiment payloads. Power is available on the magnetically suspended payload plate via a noncontacting rotary transformer. The transformer can deliver 2500 W for sensors and experiment payloads.

The actual data rate utilized for the experiment is estimated to be 40 kbps excluding any functional payload data transfer requirements.

15.7 DATA COMMUNICATIONS AND HANDLING

It is assumed that data transfer between the ASPS sensors and the Space Station data handling equipment will occur via the information and data management subsystem (IDMS) [4]. An optical data distribution network (ODDNet) provides data links around which the IDMS is centered. The ODDNet will be a high capacity network easily accessible anywhere in the Space Station by standard optical interface devices (ID). Distributed signal processing resources will be provided by standard subsystem data processors

(SDP) built up from modular standard components to accomodate the local signal processing needs of the individual user application.

The man/machine interface will be provided by a standard multipurpose applications console (MPAC) which can be either fixed or portable. Commands or software changes can be entered directly from an MPAC, from storage, or from the ground via TDRSS. Experiment data, health status information, etc. from the ASPS will be available to the crew through MPAC communications. Additional ASPS status information will be available to the crew via an ASPS information panel. Periodic health status checks and evaluation of experiment data will be performed on board but with minimal crew involvement.

On board data storage requirements include approximately 50 kbytes for command storage and 20 Mbytes for experiment data excluding functional payload data.

Periodic down-link of experiment data from storage will be required approximately once daily via TDRSS return service. Occasional up-link of command data or ASPS software on the order of 20 kbytes will occur via TDRSS forward service on an as-required basis.

A block diagram representing experiment data communications is shown in Figure 15-2.

15.8 CREW ACTIVITIES

Tasks requiring crew involvement during the execution of this experiment are minimal and can be accomplished at the task-trainable skill level. Specific activities which will occur during the 90 day orbital mission period are subsequently described with a timeline representation in Table 15-2.

The ASPS experiment hardware will be removed from the Shuttle orbiter cargo bay and installed on the Space Station as soon as possible after orbiter docking. The ASPS, mounting assembly, and experiment payload will be preassembled and can therefore be manipulated as a single module with

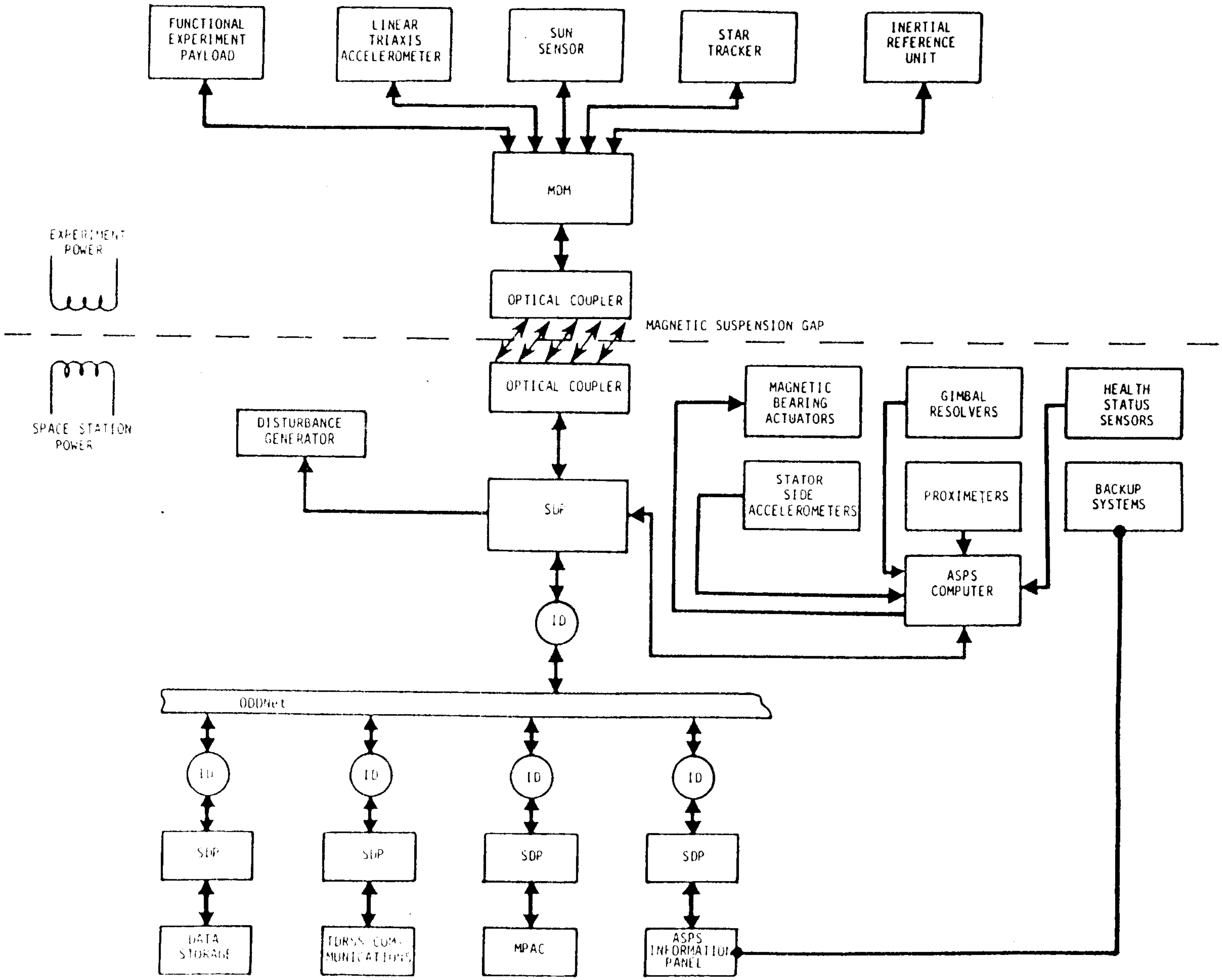


Figure 15-2. Data Communications Concept.

Table 15-2. TDMX 2432: On-Orbit Experiment Task Evaluation

FUNCTION/TASK	TIME MANMINUTES	METHOD	FEASIBILITY*	COMMENTS/ASSUMPTIONS
• Install ASPS Experiment Hardware	480	EVA-RMS-MRMS-IVA	2	Modular Pre-assembled ASPS, Mounting Assembly and Experiment Payload
• Remove ASPS from STS Cargo Bay	(48)	RMS-EVA-IVA	2	
• Transfer ASPS to MRMS	(48)	RMS-MRMS-EVA-IVA	2	If installation site is out of RMS range
• Transport ASPS to Installation Site	(96)	MRMS-EVA-IVA	2	
• Position and Secure ASPS to the Space Station Structure	(192)	MRMS-EVA-IVA	2	
• Complete External Power and Signal Interconnections	(48)	EVA	2	Assuming that Space Station power and ODDNet are accessible at the installation site
• Install and Interconnect ASPS Information Panel	(48)	IVA	2	Assume pre-existing Space Station wiring suitable for system interconnections

*1-Proven, 2-Easy, 3-Hard, 4-Questionable, 5-Not Possible

Table 15-2. TDMX 2432: On-Orbit Experiment Task Evaluation (Continued)

FUNCTION/TASK	TIME MANMINUTES	METHOD	FEASIBILITY*	COMMENTS/ASSUMPTIONS
• Experiment Set-Up	30	IVA	2	
• Turn on Equipment Power	(15)	IVA	2	
• Load Control Software and Commands	(15)	IVA	2	
• Initial Experiment Monitoring	30	IVA	2	
• Periodic Monitoring of Health Status and Experiment Data	360	IVA	2	Monitoring to occur at approximately 1 week intervals over the 90 day mission with 30 man-minutes required at each interval
• Operation Mode Switches	60	IVA	2	
• Switch from Stellar Tracking Mode to Solar Tracking Mode	(30)	IVA	2	
• Switch from Solar Tracking Mode to Vibration Isolation Mode	(30)	IVA	2	

*1-Proven, 2-Easy, 3-hard, 4-Questionable, 5-Not Possible

Table 15-2. TDMX 2432: On-Orbit Experiment Task Evaluation (Concluded)

FUNCTION/TASK	TIME MANMINUTES	METHOD	FEASIBILITY*	COMMENTS/ASSUMPTIONS
• Power Down and Preparation for Removal of Equipment	30	IVA	2	
• Remove ASPS Experiment Hardware	480	EVA-RMS-MRMS-IVA	2	
• Disconnect and Remove the ASPS Information Panel	(48)	IVA	2	
• Disconnect External Power and Signal Interconnections	(48)	EVA	2	
• Remove ASPS from the Space Station Structure	(48)	EVA-MRMS-IVA	2	
• Transport ASPS to Close Proximity of the Shuttle RMS	(96)	EVA-MRMS-IVA	2	
• Transfer ASPS from MRMS to RMS	(48)	EVA-MRMS-IVA-RMS	2	
• Position and Secure the ASPS Within the STS Cargo Bay	(192)	EVA-RMS-IVA	2	

*1-Proven, 2-Easy, 3-Hard, 4-Questionable, 5-Not Possible

the orbiter remote manipulator system (RMS) and/or the Space Station Mobile RMS (MRMS). The actual crew involvement to complete the installation of the experiment hardware including external power and signal interconnections is estimated to consist of 4 manhours of extravehicular activity (EVA) occurring simultaneously with 4 manhours of intravehicular activity (IVA) which will be required for control of the manipulator systems as well as installation of the ASPS information panel.

The initial experiment set-up will require 0.5 manhours of IVA to power up the ASPS equipment and experiment sensors as well to load commands from storage. An additional 0.5 manhours of IVA will then be required to monitor the initial performance of the system.

The remainder of the experiment will proceed largely in a quasi-autonomous fashion requiring limited crew involvment. Periodic monitoring of the system health status and experiment data will be required at 1 week intervals requiring 0.5 manhours of IVA at each interval. Changes in the mode of operation of the ASPS will occur twice during the mission such that the stellar pointing, solar pointing, and vibration isolation functions are all evaluated. Each mode switch will require 0.5 manhours of IVA for completion. It is envisioned that the daily downlink of stored experiment data via TDRSS can be completed by ground initiated interrogation, therefore excluding crew involvement for this activity. Similarly, the disturbance generator will be controlled by preprogrammed commands or commands received via TDRSS uplink.

Removal and storage of ASPS equipment aboard the Shuttle orbiter will occur in reverse order from installation. Equipment power-down and preparation will require 0.5 manhours of IVA prior to 4 manhours of EVA and 4 manhours of IVA required to remove the ASPS from the Space Station, transport it to the orbiter and secure it in the cargo bay.

15.9 GROUND PROCESSING

A data receiving station will be required to accept and store the telemetered experiment raw data on magnetic tapes which will be formatted and time tagged. Analysis of the data will begin during the experiment period as soon as data are available with a preliminary evaluation completed in 3 months after the end of the flight. Final report documentation will be completed within 12 months of the end of the flight with the entire analysis program requiring a 2 manyear level of effort.

The manufacturer of the ASPS will complete a postflight teardown and inspection of the hardware in order to determine the effects of the low Earth orbit environment on system components. Final report documentation of the postflight inspection will be completed within 6 months after the mission and will require a 2 manyear level of effort.

15.10 SUPPORT EQUIPMENT

Shipping containers will be required to provide appropriate protection for launch-sensitive experiment components such as the low-g triaxis accelerometer. A launch support structure will be necessary for transport of the ASPS aboard the Space Shuttle. Ground test equipment will be required in order to verify the prelaunch and postflight status of the ASPS system.

15.11 INTER-EXPERIMENT COMPATIBILITY

The ASPS experiment will be compatible with other proposed experiments on Space Station. Inasmuch as the ASPS gimbal system allows significant movement of the experiment payload, due consideration should be given to collision avoidance logic with other experiments or objects moving in close proximity. The ASPS will be insensitive to other experiment activities as long as EMI levels do not exceed Space Station specified limits.

REFERENCES

1. Keckler, C.R.: "ASPS Performance with Large Payloads Onboard the Shuttle Orbiter." Journal of Guidance and Control, Volume 5, Number 1, January-February 1982, pp. 32-36.

2. Keckler, C.R. and B.J. Hamilton: "A System for Load Isolation and Pointing." SPIE Advanced Technology Optical Telescopes II, Proceedings, pp. 132-138, SPIE Volume 444, September, 1983.

3. Keckler, C.R.: "Vibration Isolation Technology Experiment." pp. 221-231, NASA CP-2294, 1982.

4. NASA Lyndon B. Johnson Space Center: "Space Station Reference Configuration Description." JSC 19989 Systems Engineering and Integration, Space Station Program Office, August, 1984.

16.0 TDMX 2442 -- TRANSIENT UPSET PHENOMENA IN VLSI DEVICES

CONTACT: Felix L. Pitts, MS 130
NASA LaRC (804) 865-3681

16.1 OBJECTIVE

The objective of this TDM experiment is to develop a data base which will contribute to the understanding of alpha particle and cosmic ray induced single event upsets of Very Large Scale Integrated (VLSI) circuits. VLSI complexity is considered to mean greater than one million active elements on a chip. In digital system design for space applications, the greater the complexity (and therefore density) of the circuits used, the more vulnerable they are to upsets caused by high energy particles such as alpha particles, cosmic rays, and heavy ions. It is currently believed by many researchers in solid state electronics that neither shielding nor hardening of the integrated circuits will be able to alleviate these problems completely, for VLSI and higher circuit densities. There is thus a need to understand the nature of system upsets that can be induced by high energy particles in the complex devices which could be used to enhance space application capabilities.

The data base provided as a result of this experiment would contain micro-event data on system level upsets obtained from actual environment exposure, to high energy particles in space, of test bed designs using ultra large scale integrated circuit devices performing generic versions of critical application tasks. The devices used in the experiment would be specially fabricated for this purpose using state-of-the-art processes, and would include circuits to facilitate monitoring of upset activity. Similar studies to this can be performed in ground-based laboratories using simulated fault injection; however the pattern of faults would not be representative of those obtained in an actual space application. This experiment would be exposed to the same environment as the applications which would utilize its results, and therefore relevance of the data base is assured.

16.2 DESCRIPTION

Refer to Figures 16-1 and 16-2 for a block diagram and sketch of the experiment configuration. The experiment would consist primarily of two main components: the Unit Under Test (UUT) and the Experiment Control and Monitoring computer (ECM). The UUT would be a digital system utilizing specially fabricated VLSI devices and programmed to perform a generic version of some critical space application function. The ECM would be essentially a general-purpose computer programmed to perform the same application function reliably for comparison with the UUT outputs, to determine when an upset has occurred, and to record the values of the monitored signal lines from the UUT at these times. The UUT would be minimally shielded in order to deliberately expose it to a worst-case radiation environment for the given Space Station orbit. The ECM equipment would require maximum shielding, however, or compensate for the lack of shielding by incorporating a high degree of fault-tolerance. It is important to the success of the experiment and reliability of the data base obtained from it that the probability of undetected faults in the ECM equipment be extremely low. One possiblity for implementation of the ECM computer is to use the VHSIC fault-tolerant processor proposed for TDMX 2443. That ultra-reliable computer will have the capacity to handle the processing tasks required of the ECM computer.

The VLSI integrated circuits used in the unit under test will be specially fabricated to include logic that will permit accurate detection and characterization of system upsets. The output signals generated by this on-chip logic will be monitored by the ECM computer. The special monitoring circuitry will make lower-level functions for the system observable which would otherwise not be possible. These signals will include internal data, control, and address busses as well as register clocks and state vectors which are not normally available for monitoring.

Some analysis of the events monitored will be performed on board in the ECM computer, thus reducing the amount of data which will require storage and/or transmission to the ground. A tradeoff thus exists between the amount of on-board processing power and the data communications requirements.

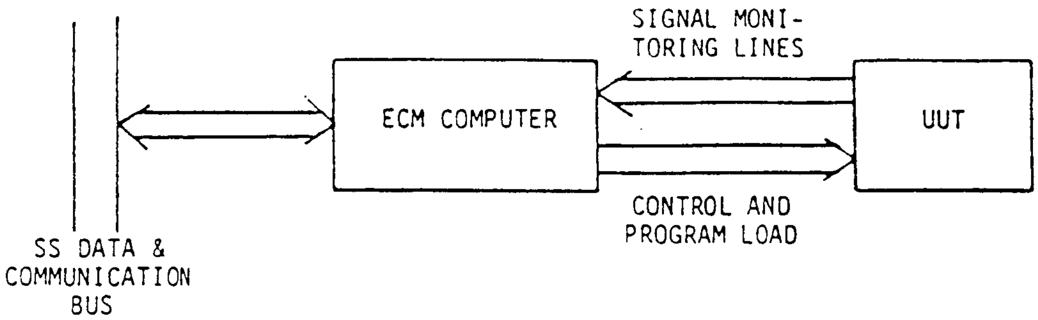


Figure 16-1. Transient Upset Phenomena in VLSI Devices - Block Diagram.

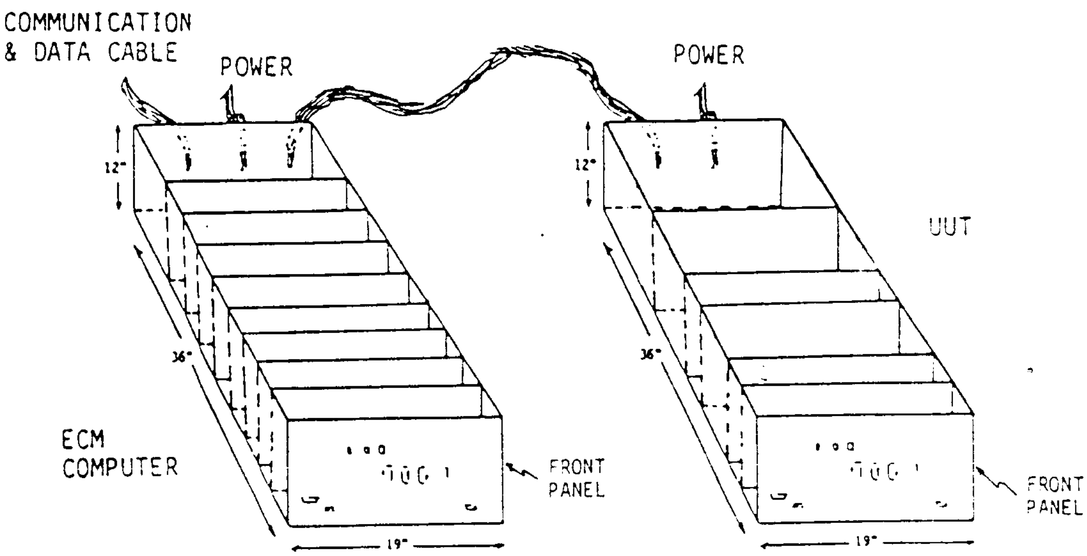


Figure 16-2. Transient Upset Phenomena in VLSI Devices - Configuration.

16.3 ENVIRONMENT

The environmental requirements to be considered include structural considerations, orbital parameters, platform utilized, and required orientation.

No special orbit is required for this experiment, as any orbit in which the Space Station is likely to be placed would expose the unit under test to a sufficient level of natural radiation for purposes of the test. The desired location of the equipment is internal to a pressurized Space Station module for easy crew access. The unit under test must not be blocked by anything which might excessively shield it from the radiation environment. Other than this limitation, there is no particular field-of-view or orientation requirement.

16.4 SENSORS

The unit under test in this experiment is acting as a sensor in that it reacts to the radiation in its environment. Unlike a more conventional sensor, however, the purpose of exposing it to this environment is not to make measurements of that radiation but to record the UUT's reaction to it. The UUT is simply a general-purpose digital processor system implemented on a VLSI chip which has been specially fabricated to allow enhanced fault detection and characterization. The parameters to be measured will be the outputs of the special monitoring circuitry on the chips, in addition to any data outputs from the program being run on the processor. The special monitoring circuitry will make lower-level functions of the system observable which would otherwise not be possible. These signals will include internal data, control, and address busses as well as register clocks and state vectors which are not normally available for monitoring. In addition, the ECM computer would require station data from the on-board guidance, navigation and control system. This data would be sampled whenever a system upset occurred in order to record the time, position, and orientation of the event.

The entire UUT could be implemented on a single circuit board incorporating power supply regulation and circuitry to interface it to the ECM computer. More data can be obtained by placing several similar or identical VLSI devices in the same experiment package, and multiplexing the outputs to the ECM computer. In this case the package must be designed and located so as to facilitate easy replacement of individual chips which have exhibited evidence of hard failure. The duty cycle of the experiment is continuous 24 hours per day operation.

16.5 DATA ACQUISITION AND CONTROL

Data acquisition for this experiment as well as control functions will be accomplished by the ECM computer system, which will be continuously monitoring the unit under test for system upsets. The ECM system will store in its memory a record of such upsets or a statistical summary of them. The more analysis of the upset data that is performed by the ECM, the greater the data compression that will be achieved. This will reduce the amount of on-board storage required for the results. The data will be periodically transferred to a backup mass storage device and/or transmitted to the ground. A reasonable estimate of the amount of random access memory required for the ECM computer is 1 megabyte.

16.6 DATA COMMUNICATIONS AND HANDLING

Data communications within the experiment package will be handled by a special data bus between the UUT and the ECM, and will not require any Space Station resources. The requirement for availability of station data from the on-board guidance navigation and control system will require an interface to the Space Station data bus. These data will be sampled at a very low rate, as they are only required when recording system upset data.

16.7 TELEMETRY

A rough estimate of the amount of data to be transmitted is 1 megabit per month, with retransmission required for reliability. The data need not be monitored in real time, as adequate buffering of the data is available in the ECM. Therefore, no significant burden will be placed on the TDRSS communications link to accommodate this experiment's telemetry.

16.8 CREW ACTIVITIES

Very minimal crew interaction with this experiment is required. No EVA activities are needed, and the IVA is limited to setup and stow operations plus an occasional module replacement at the circuit board level requiring minimal skill level. Table 16-1 summarizes these activities in terms of duration and frequency. All tasks in this table are of the task-trainable level of difficulty--that is, they do not require any special skill level. The assumption is made that unloading of experiment packages from the docked orbiter which are destined for the interior of a pressurized lab module will not require EVA, and similarly for loading of the package from the lab module. If this assumption proves false, then the setup and stow operations will require EVA with a corresponding increase in duration.

16.9 GROUND PROCESSING

As indicated above, some data analysis steps will be performed by the on-board ECM computer system, primarily for the purpose of achieving data compression. More detailed evaluation and interpretation of the data will be performed on the ground. Data processing performed on the ground will consist primarily of reformatting the data, printing human-readable summary reports, and archiving the data on tape. The results of these analyses may lead to required changes of equipment (particularly the unit under test) as well as changes in the programming of both the unit under test and the ECM computer. The procedures for interpreting single event upset data are not yet fully developed, and work is ongoing presently to refine such procedures.

16.10 SUPPORT EQUIPMENT

No special support equipment would be required at the station for this experiment. The only support equipment required on the ground would

Table 16-1. Crew Activities Timeline for TDMX 2442.

FUNCTION/TASK	TIME (MIN)	METHOD	FEASIBILITY*
Unload and Setup Experiment:	45	IVA	2
Replacement of UUT Modules--			
After Hard Failures (once every 2 months average):	120	IVA	2
Replacement of UUT Modules--			
For New Circuit Introduction (once every 6 months):	15	IVA	2
Install New Software (once every 6 months):	15	IVA	2
Recovery of Backup Storage Media (once per year):	10	IVA	2
Stów and Load:	45	IVA	2

*1 = Proven, 2 = Easy, 3 = Hard, 4 = Questionable, 5 = Not Possible be general-purpose computing capacity for analysis and formatting of data received from the experiment.

16.11 THERMAL, POWER, WEIGHT, AND VOLUME REQUIREMENTS

It is estimated that the unit under test, its exposure chamber, and the experiment control and monitoring computer would occupy approximately 0.3 cubic meters. Power requirements are estimated to be 5 watts for the UUT and 90-100 watts for the ECM computer. Voltage regulation will be provided by the experiment electronics.

The thermal characteristics of this experiment require equipment temperature to be maintained in the range 0-40 degrees C. Waste heat rejection capacity of approximately 100 watts is required while the unit is operational.

16.12 INTER-EXPERIMENT COMPATIBILITY

This experiment is closely related to TDMX 2443, VHSIC Fault Tolerant Processor, in that they both will contribute to advancements in the state-of-the-art in fault tolerant digital system technology for space applications. However, neither experiment depends upon the other, nor will they be required to interface to each other. Fundamental differences exist in both the approach and objectives of the two experiments. TDMX 2442 will acquire lower-level data on the characteristics of system upsets which cause faults to appear, while TDMX 2443 will be monitoring the behavior of an existing fault-tolerant architecture in the space environment. Thus, TDMX 2443 can be considered more of a concept demonstration for the use of VHSIC technology in a fault-tolerant architecture in the space environment, while TDMX 2442 will contribute more basic research results to the development of new fault-resistant circuit technology.

The potential does exist, however, for the processing power of TDMX 2443 to be used as the ECM (experiment control and monitoring) computer system required by TDMX 2442.

17.0 TDMX 2443 -- VHSIC FAULT TOLERANT PROCESSOR

CONTACT: Dr. Harry F. Benz, MS 473
NASA LaRC (804) 865-3777

17.1 OBJECTIVE

The mission objective for this experiment is to acquire realistic data on single event upset recovery in a self-testable general purpose computer configuration which uses 1.25 micrometer Very High Speed Integrated Circuit (VHSIC) technology. This data will then be used to establish a system error rate and statistical data base. A support data system will be used to monitor the VHSIC processor to characterize its faults and recovery modes. The data collected by this experiment will result in a thorough understanding of the reliability and fault tolerance of such a system in a realistic operating environment for space applications.

Several advanced system-level technologies are available to implement recoverability from device failures, including built-in test and system reconfiguration algorithms. Utilization of these technologies can enhance the fault tolerance of advanced data systems, especially important for space applications in light of the increased radiation sensitivity of higher-density circuit technologies. The Department of Defense VHSIC program is the first system-level technology which addresses these as realistic goals.

17.2 DESCRIPTION

Refer to Figures 17-1 and 17-2 for a block diagram and sketch of this experiment. The experiment package will consist of a fault-tolerant general purpose computer designed around the VHSIC Phase I chip set, programmed for tasks which simulate mission-critical functions as well as fault monitoring activities. The system will record fault detection and isolation sequences, as well as the occurrence of hard faults and the resulting reconfiguration activities.

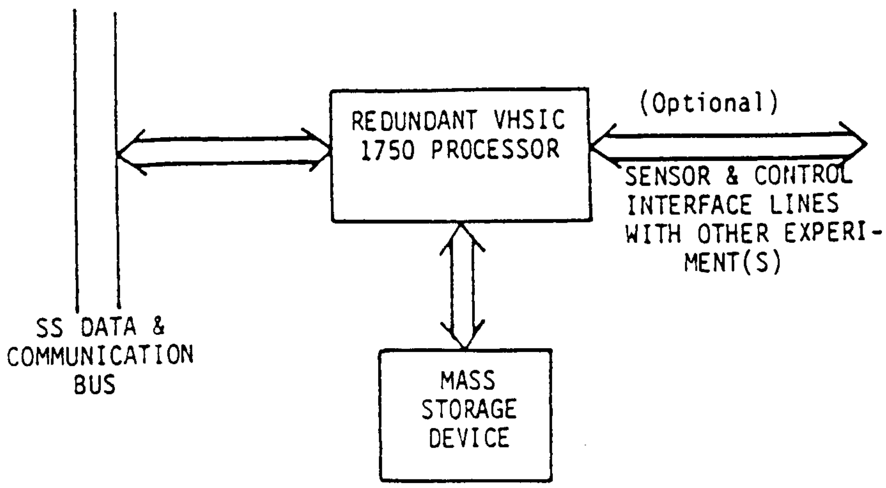


Figure 17-1. VHSIC Fault-Tolerant Processor - Block Diagram.

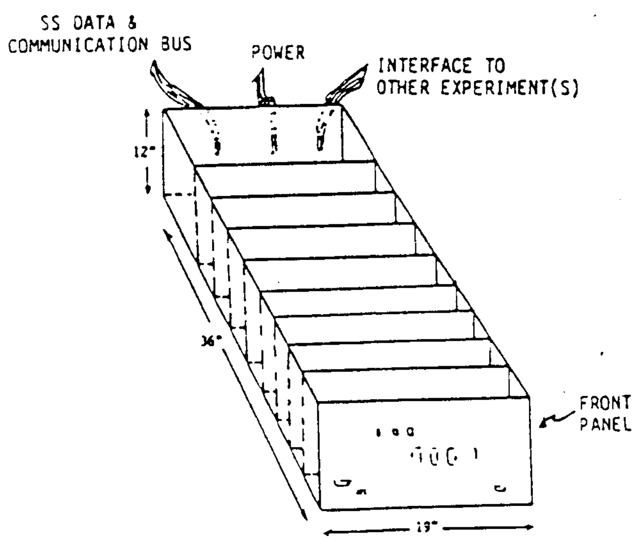


Figure 17-2. VHSIC Fault-Tolerant Processor - Configuration.

The VHSIC processor will be implemented with an architecture that is resistant to single event upsets, and will utilize the highest chip density component-level technology available. It is anticipated that at the start of the experiment this density will be represented by the VHSIC Phase I 1.25 micrometer technology, while during the life span of the experiment the VHSIC Phase II sub-micrometer technology will become available. Planned technology upgrades will be made in the experiment package through the use of functional modularity at the processor level, as well as software modularity through the use of the Ada programming language.

17.3 ENVIRONMENT

The environmental requirements to be considered include structural considerations, orbital parameters, platform utilized, and required orientation.

No special orbit is required for this experiment, as any orbit in which the Space Station is likely to be placed would expose the system to sufficient levels of natural radiation for purposes of the test. The desired location of the equipment is internal to a pressurized Space Station module for easy crew access. The system must not be blocked by anything which might excessively shield it from the radiation environment. Other than this limitation, there is no particular field-of-view or orientation requirement.

17.4 SENSORS

The VHSIC processor used in this experiment is acting as a sensor in that it reacts to the radiation in its environment. Unlike a more conventional sensor, however, the purpose of exposing it to this environment is not to make measurements of that radiation but to record the system's reaction to it. The system will be extremely fault-tolerant, to the point that it is unlikely that any anticipated failure would undermine its ability to monitor and reconfigure itself. After enough hard failures have occurred to make this more likely, the failed modules will be replaced.

The package must be designed and located so as to facilitate easy replacement of individual chips or circuit board which have exhibited evidence of hard failure.

17.5 DATA ACQUISITION AND CONTROL

Primary data acquisition will be performed by the VHSIC fault-tolerant computer itself, and stored in on-board random access memory until it can be transmitted to the ground. A mass storage nonvolatile memory device which will also be used to supply programs to the system can be utilized for backup storage of data. An optical disc drive is being considered for this purpose. If a flight-qualified optical disc drive is not available for this experiment then bubble memory technology will be used for nonvolatile memory needs. The data system monitoring software will log all abnormal events, such as single event upsets, as well as gradual radiation degradation caused by integrated dose. In addition, the system will log the sequences for fault identification and isolation, chip self-test results, and system restoration modes. Where hard faults are found it will log the fault site on the chip for future analysis and will reconfigure the system to accommodate faults. The system will require knowledge of the station's position and attitude from the on-board guidance, navigation and control system. This data will be used to log the abnormal event information.

The design of the computer system will be based on the MIL-STD 1750 architecture. A high degree of fault tolerance will be achieved through the use of massive redundancy, with possibly as many as 6 redundant processors. The main memory requirements for the system are estimated to be 2 Megabytes of shared memory plus 64 Kbytes per processor.

17.6 DATA COMMUNICATIONS AND HANDLING

All data communications within the experiment will be handled by internal data busses and subsystems, and will not require any external Space Station resources. The requirement for availability of station data from the on-board guidance navigation and control system will require an interface to the Space Station data bus. These data will be sampled at a very low rate, as they are only required when recording system faults.

17.7 TELEMETRY

A rough estimate of the amount of data to be transmitted is 1 megabit per month, with retransmission required for reliability. The data need not be monitored in real time, as adequate buffering of the data is available in the system. Therefore, no significant burden will be placed on the TDRSS communications link to accommodate this experiment's telemetry.

17.8 CREW ACTIVITIES

Very minimal crew interaction with this experiment is required. No EVA activities are needed, and the IVA is limited to setup and stow operations plus an occasional module replacement at the circuit board level requiring minimal skill level. Table 17-1 summarizes these activities in terms of duration and frequency. All tasks in this table are of the task-trainable level of difficulty--that is, they do not require any special skill level. The assumption is made that unloading of experiment packages from the docked orbiter which are destined for the interior of a pressurized lab module will not require EVA, and similarly for loading of the package from the lab module. If this assumption proves false, then the setup and stow operations will require EVA with a corresponding increase in duration.

17.9 GROUND PROCESSING

Detailed evaluation and interpretation of the data on upset events, fault identification and isolation, and reconfiguration activities of the VHSIC processor system will be performed on the ground after collection of the data, which would be periodically transmitted from the station. Health and status information on the system would be transmitted at a low data rate to enable recognition of the need for non-routine maintenance operations. Any additional ground processing required would be limited to reformatting of the data and printing summary reports, as well as

Table 17-1. Crew Activities Timeline for TDMX 2443.

FUNCTION/TASK	TIME (MIN)	METHOD	FEASIBILITY*
Unload and Setup Experiment:	45	IVA	2
Replacement of Circuit Modules-- After Hard Failures (once every 2 months average):	120	IVA	2
Replacement of Circuit Modules-- For New Circuit Introduction:	15	IVA	2
Install New Software (once every 6 months):	15	IVA	2
Recovery of Backup Storage Media (once per year):	10	IVA	2
Stow and Load:	45	IVA	2

*1 = Proven, 2 = Easy, 3 = Hard, 4 = Questionable, 5 = Not Possible transferring it to a different medium (tape) for archival and distribution purposes.

17.10 SUPPORT EQUIPMENT

No special support equipment would be required at the station for this experiment. The only support equipment required on the ground would be general-purpose computing capacity for analysis and formatting of data received from the experiment.

17.11 THERMAL, POWER, WEIGHT, AND VOLUME REQUIREMENTS

It is anticipated that the complete system will require one cubic meter of volume, and have a mass of approximately 100 kg. Power consumption is estimated to be approximately 1 kW continuous. The experiment package will include the required voltage regulation. Heat rejection at 40 degrees C will be required.

17.12 INTER-EXPERIMENT COMPATIBILITY

This experiment is closely related to TDMX 2442, Transient Upset Phenomena in VLSI Devices, in that they both will contribute to advancements in the state of the art in fault tolerant digital system technology for space applications. However, neither experiment depends upon the other, nor will they be required to interface to each other. Fundamental differences exist in both the approach and objectives of the two experiments. TDMX 2442 will acquire lower-level data on the characteristics of system upsets which cause faults to appear, while TDMX 2443 will be monitoring the behavior of an existing fault-tolerant architecture in the space environment. Thus, TDMX 2443 can be considered more of a concept demonstration for the use of VHSIC technology in a fault-tolerant architecture in the space environment, while TDMX 2442 will contribute more basic research results to the development of new fault-resistant circuit technology.

The potential does exist, however, for other experiments to utilize the processing power of this experiment. In particular, this processor could be used as the ECM (experiment control and monitoring) computer system required by TDMX 2442.

18.0 TDMX 2521 - ACOUSTIC CONTROL TECHNOLOGY

CONTACT: David A. McCurdy/David G. Stevens, MS 463
NASA/LaRC (804) 865-3561

18.1 GOALS AND OBJECTIVES

The goal of TDMX 2521 is to develop a data base of vibroacoustic measurements and human responses that will establish levels, frequency, and duration of exposure within the Space Station. The experiment will provide information that will be used to characterize vibroacoustic exposure on the station, to determine the effects of this exposure upon man and system performance and to compare with model predictions. The experiment would, therefore, serve to validate methods for predicting the level, frequency, and time history of exposure at specific locations, allow for the development of vibroacoustic exposure criteria and the identification of strategies for controlling exposure to meet these criteria.

This experiment will develop the methods and technology required to design and operate a Space Station to ensure acceptable levels of vibroacoustic exposure and establish coincidence with the requirements for various system and crew functions. Human habitability/productivity functions to be addressed include hearing, speech, communication, performance, comfort, and sleep. Levels of vibroacoustic exposure will also be studied to determine their effects upon system performance functions such as fine pointing accuracy and structural distortion due to vibratory excitation.

18.2 METHODOLOGY

The vibroacoustic environment of the Space Station will directly affect the comfort, performance, and utilization of the on-board personnel as well as the performance of on-board equipment. Although criteria can be developed for crew habitability, ground based tests that simulate these conditions cannot provide the sustained low-gravity, long duration confinement or sustained exposure necessary to validate the noise and vibration criteria.

Extensive programs are ongoing in the area of aircraft noise control involving prediction, propagation and transmission of noise, and development of human criteria. Programs have been developed, facilities are being designed, and flight experiments utilizing current space flight systems are planned to develop methodologies for vibroacoustic prediction, criteria, and control. These methodologies are to be used in the design and operation of a Space Station to ensure high levels of habitability and productivity.

18.2.1 Description

Objective and subjective tests will be conducted to validate habitability criteria developed for the noise and vibration environment of the space station. Tests will assess the effects of the space station noise and vibration environment on crew hearing, speech, task performance, annoyance, and sleep. Other tests will measure and monitor the noise and vibration environment of the Space Station for comparison with predicted environments. The experiment will require manned support and the following equipment:

- Sound level meter and octave band filter.
- Vibration measurement equipment.
- Noise dosemeters.
- Tape recorder/microphone.
- Noise monitor.
- Portable audiometer.

18.2.2 Procedure/Data Acquisition

The experiment is divided into nine tasks which monitor the vibroacoustic environment and its effects on the crew. These tasks, as shown in Figure 18-1, will be completed by a combination of crew support and the appropriate hardware required for source generation, data acquisition, and data storage.

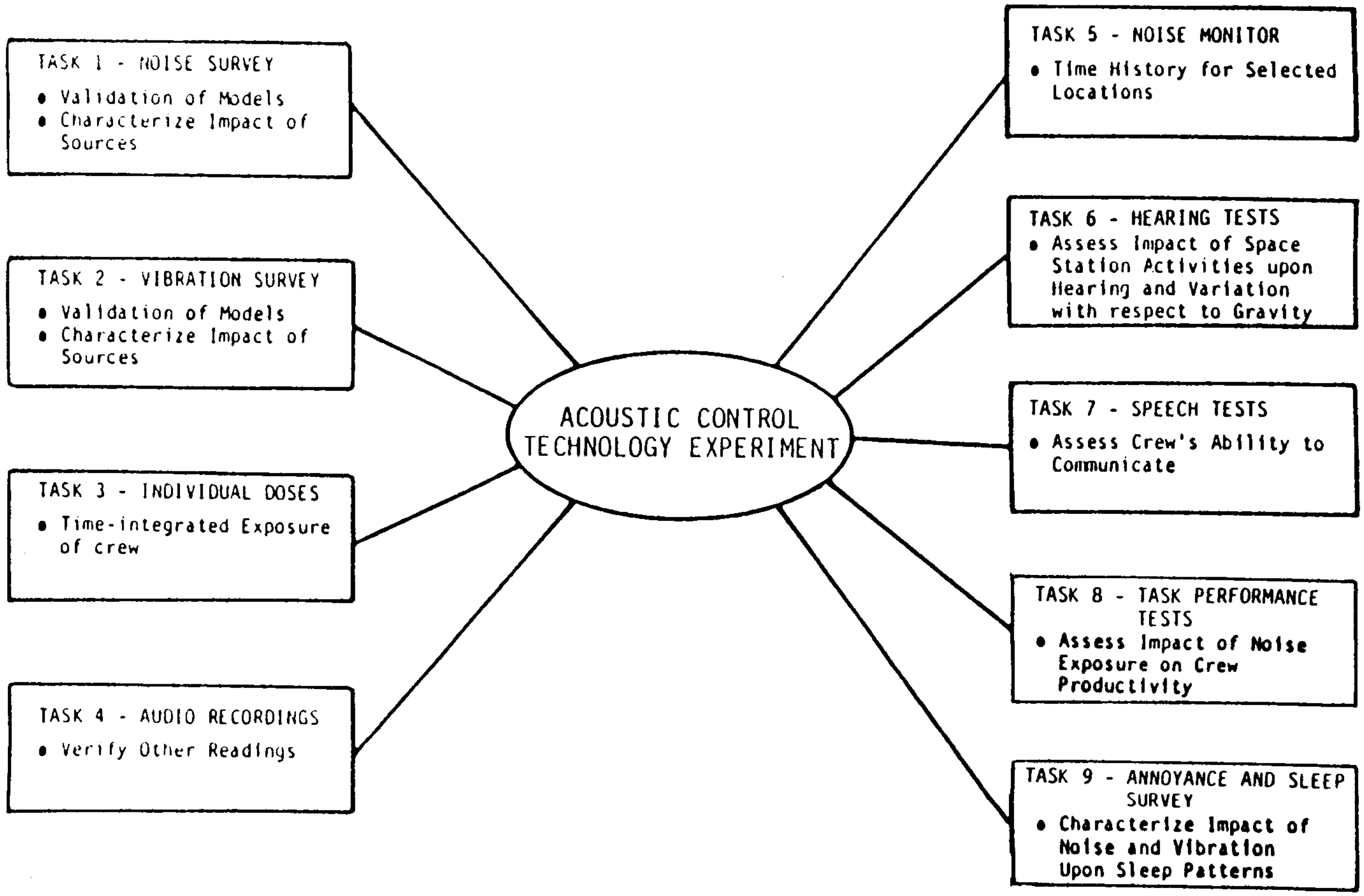


Figure 18-1. Acoustic Control Technology Experiment Description.

The general procedure and data acquisition requirements for each of the nine tasks are discussed below:

- o Task 1 - Noise Survey

  This task will involve taking noise measurements, both overall and within individual frequency bands, at predetermined locations for comparison with predictions. Also, other locations identified by crew members as noisy could be designated as sites for additional noise measurements as a first step towards controlling the noise.

- o Task 2 - Vibration Survey

  This task is similar to Task 1, except that the focus in this case is upon vibration measurements.

- o Task 3 - Individual Doses

  This task will measure the individual's noise exposure over a 24-hour period as the individual moves about the Station.

- o Task 4 - Audio Recordings

  Listening quality recordings will be made at various locations for analysis on earth. These recordings would be used to verify readings obtained in other tasks and to perform more detailed analyses of the characteristics of the noise.

- o Task 5 - Noise Monitoring

  Measurements would be taken in selected locations over an extended period of time to determine a time history of exposure for each location.

- o Task 6 - Hearing Tests

  Four audiograms would be performed at the following times: (1) on Earth before departure, (2) on station immediately after boarding, (3) on station just before departure, and (4) on Earth immediately after return. This information would be used to assess the impact of the Space Station environment (e.g., noise, vibration, zero gravity, reduced pressure) upon hearing.

o Task 7 - Speech Tests

Tests would be conducted to assess the ability of crew members to communicate effectively during selected activities and in specified areas. In one form of these tests, selected phrases would be played from a tape recorder or spoken by one crew member to another crew member whose understanding of the phrase is assessed.

o Task 8 - Task Performance Tests

Tests will be conducted to determine how noise and vibration exposure on the station affects crew productivity.

o Task 9 - Annoyance and Sleep Survey

A series of questionnaires would be answered by crew members in an attempt to characterize the impact of noise and vibration upon crew sleeping patterns and the level of nuisance associated with this exposure.

18.2.3 Evaluation and Interpretation of Results

Experimental results of level, frequency, and time history of exposure will be used as a benchmark for comparison with those predicted by model. This will be useful in enhancing these predictive procedures to improve the quality of future missions. Exposure data will also be used to characterize the exposure of crew members so that existing exposure criteria may be judged, and if appropriate, to modify these criteria to incorporate experimental findings. Finally, the results will be used to establish approaches for control of vibroacoustic exposure.

18.3 ENVIRONMENT

The particular space environment parameters do not drive the successful completion of the experiment. The experiment will be conducted totally within the space station habitat and will not be affected by orbital parameters or space station orientation. It will have no interaction with co-orbiting platforms, nor will the experiment have any structural implications upon the space station. However, successful completion of the experiment and implementation of resulting recommendations will act to improve the environment of the Space Station habitat. Implementation of the recommendations could also have a positive influence upon pointing accuracy.

To ensure reliable data, audio tapes used for data acquisition should not be exposed to magnetic fields.

18.4 HARDWARE

The following information describes the requirements of the hardware needed for completion of the experiment. Many of these requirements can be satisfied by commercially available equipment; however, much of this equipment has not been flight qualified.

18.4.1 Sound Level Meter

This meter will be used for measuring noise levels at various locations throughout the station and are commercially available. The device, in its present configuration, would be handheld and would require manual recording of meter output. Methods for automatic data recording will be investigated.

Parameters measured: Overall sound pressure level (SPL), A-weighted SPL, and third octave band SPL

Outputs: Visual meter

Data Rates: As specified

Duty Cycle: One minute per reading

Dynamic Range: 20 dB to 140 dB

Sensors: Part of portable device

18.4.2 Vibration Transducer

The vibration transducer will be used in conjunction with the above sound level meter for measuring vibration levels at various locations on-board the station. Data output will be of the same as described in the sound level meter section.

Parameter measured: Vibration level

Outputs: Visual meter

Data Rates: As specified

Duty Cycle: One minute per reading

Frequency Range: Less than 500 Hz

Sensors: Piezoelectric or Piezoresistive accelerometers.

Commercially available accelerometers are hermetically sealed, unaffected by vacuum, and well suited for space applications. Appropriate choice of sensor must be made to exclude effects of magnetic and RF fields. Thermal characteristics of available devices are compatible with requirements.

18.4.3 Noise Dosemeter

This meter will be used to obtain an individual's level of time integrated noise exposure. The meter is commercially available through a number of suppliers. These devices can be completely self-contained and are of carry-around size (122 mm x 75 mm x 28.5 mm). The device provides continuous and periodic readout of integrated noise dose received by the individual. Available hardware responds to noise peaks as short as 100 s and possesses an accelerated mode for quick surveys. Approximate weight for the device ranges from 0.4 to 1.0 kg.

Outputs: Visual meter

Duty Cycle: 24-hour cycle on specified days

Dynamic Range: 60 dB (or as low as available) to 140 dB

Sensor Type: Part of portable dosemeter

18.4.4 Noise Monitor

This device would automatically sample and record noise levels over an extended period of time at particular locations. Monitors designed to provide these noise level time histories are commercially available.

Parameters measured: SPL at discrete time intervals

Outputs: Sound levels

Data Rates: To be specified (probably 1/sec)

Duty Cycle: 24-hour cycle, 7 days of operation

Frequency Range: 4 Hz to 40 kHz

Sensor Type: Capacitor microphone

Ancillary Data: Records of activity on board the Space Station that would influence noise levels during the monitoring period.

18.4.5 Audio Recorder

This device will record noise on the space station at times and locations of interest for more detailed analysis on the ground. Stereo cassette analog recorders can provide high quality signal storage with small associated weight. One channel can be used for noise data storage and the other channel used as voice channel for time and condition markers. Commercially available equipment includes Nakamichi or JVC with dBX or Dolby-C noise reduction. The transport and/or electronics for unnecessary functions can be modified to minimize weight.

Output: None

Data Rates: None

Duty Cycle: To be specified

Frequency Range: Limited by microphone, 20 Hz to 20 kHz

Sensor Type: High quality microphone for signal

Low mass microphone for voice channel

Ancillary Data: ambient conditions and time recorded on voice channel

18.4.6 Portable Audiometer

This device will be used to measure the hearing sensitivity of crew members. Typically, this device would play tones through headsets. Hearing tests will be administered to each crew member at the beginning and end of the orbital mission. A rather crude audiometer has flown on a previous shuttle flight.

18.4.7 Power Amplifier and Speaker

This device would function as a noise source during crew communications tests and might also serve to initiate vibration and acoustic propagation tests. Signal source could be the previously mentioned cassette recorder which would offer distinct advantages in weight and simplicity of operation over reel-to-reel hardware.

18.5 DATA COMMUNICATIONS AND HANDLING

Data resulting from the conduct of the experiments will be in the form of data sheets, audio tape, and printer paper. Return of data for ground analysis at end of the 90 day mission would be sufficient, although transmission to ground of selected data (on a non-priority basis) prior to the end of the mission would be useful in planning for subsequent mission(s). Methods for automating data recording and handling will be investigated.

18.6 GROUND PROCESSING

Data resulting from the experiment will be analyzed to validate methods for predicting level, frequency, and time history of noise and vibration at each location, to develop vibroacoustic exposure criteria, and to develop strategies for controlling exposure to meet these criteria.

18.7 CREW ACTIVITIES

Skills required are operation of various acoustical equipment and administration of various subjective response tests and surveys. The experiment consists of nine major tasks with task frequency ranging from one to nine times per 90 day mission. Time durations for each task occur rence range from 0.5 hours to four days. Table 18-1 summarizes the crew activities for each task while Figure 18-2 provides a preliminary time sequence of events.

Table 18-1. Acoustics Control Technology Development Experiment Requirements

SPACE STATION CAPABILITY EVALUATION
ACOUSTICS CONTROL TECHNOLOGY DEVELOPMENT EXPERIMENT
TASKS

TASK	SKILL	OCCURRENCES PER MISSION	DAILY TASK DURATION, HR.	PEOPLE REQUIRED	TIME REQUIRED PER PERSON, HR.	TOTAL MANHOURS	HOURS OF EQUIPMENT OPERATION
1. NOISE SURVEY	EQ. OP.	2	8	2	8	32	64
2. VIBRATION SURVEY	EQ. OP.	2	8	2	8	32	64
3. INDIVIDUAL DOSES	EQ. OP.	8	24	1	1/2	4	192
4. AUDIO RECORDINGS	EQ. OP.	1	8	1	8	8	8
5. NOISE MONITOR	EQ. OP.	7	24	1	1	7	168
6. HEARING TESTS	EQ. OP.	2	4	4	2	16	16
7. SPEECH TESTS	TEST ADM.	1	8	4	4	16	8
8. TASK PERFORMANCE TESTS	TEST ADM.	2	8	4	4	32	0
9. ANNOYANCE AND SLEEP SURVEY	NONE	7	1/2	4	1/2	14	0

	WEEKS AFTER EXPERIMENT INITIATION												
WEEK	1	2	3	4	5	6	7	8	9	10	11	12	13
TASK 1			Δ								Δ		
TASK 2			Δ								Δ		
TASK 3			Δ							Δ (4 DAYS)			
TASK 4											Δ		
TASK 5			Δ (3 DAYS)							Δ (4 DAYS)			
TASK 6	Δ												Δ
TASK 7							Δ						
TASK 8		Δ										Δ	
TASK 9	Δ		Δ		Δ		Δ		Δ		Δ		Δ

FIGURE 18-2. ACOUSTICS CONTROL TECHNOLOGY DEVELOPMENT EXPERIMENT TASK TIMELINE.

Standard Bibliographic Page

1. Report No. NASA CR-178153	2. Government Accession No.	3. Recipient's Catalog No.
4. Title and Subtitle Concept Definition for Space Station Technology Development Experiments - Experiment Definition		5. Report Date April 1986
		6. Performing Organization Code
7. Author(s)		8. Performing Organization Report No. RTI/3042/07-01F
9. Performing Organization Name and Address Research Triangle Institute P.O. Box 12194 Research Triangle Park, NC 27709		10. Work Unit No.
		11. Contract or Grant No. NAS1-17639
12. Sponsoring Agency Name and Address National Aeronautics and Space Administration Washington, DC 20546		13. Type of Report and Period Covered Contractor Report
		14. Sponsoring Agency Code 480-52-05-01
15. Supplementary Notes Langley Technical Monitor: Lenwood G. Clark Final Report - Task 2		

16. Abstract

This report documents the second task of a study with the overall objective of providing a conceptual definition of the Technology Development Mission Experiments proposed by LaRC on Space Station. During this task, the information (goals, objectives, and experiment functional description) assembled on a previous task was translated into the actual experiment definition. Although still of a preliminary nature, aspects such as: environment, sensors, data acquistion, communications, handling, control telemetry requirements, crew activities, etc., were addressed. Sketches, diagrams, block diagrams, and timeline analyses of crew activities are included where appropriate.

17. Key Words (Suggested by Authors(s)) Space Station Technology Development Experiments		18. Distribution Statement Unclassified - Unlimited Subject Category 15	
19. Security Classif.(of this report) Unclassified	20. Security Classif.(of this page) Unclassified	21. No. of Pages 288	22. Price A13