A Feasibility Study for a Remotely Controlled, Low-power-consumption Scanning Electron Microscope Suitable for Space Applications.

John L. Chidester
Utah State University Electron Microscopy Laboratory
C. Lynn Chidester
Globesat, Inc.

Experiments and instruments for operation in and on small satellites are constrained by the volume, weight, power, and ruggedness requirements of the satellite design and structure. It is anticipated that increasing numbers of "laboratory" style experiments will be performed in space. An application of current interest is the in-space observation of material surfaces exposed to the space environment. One such instrument for possible small satellite flight operational use, is a scanning electron microscope (SEM). The design factors and parameters of such an SEM, as well as the trade-offs and instrument limitations, will be discussed. The electronic control and image relay requirements will be presented. This study shows that by choosing the proper design for an SEM, this instrument could be a valuable and useful tool to be flown on a small satellite.

INTRODUCTION

In materials analysis, and in the study of physical and chemical changes in materials, it is often desirable to obtain an image of what is happening. An instrument that has become more and more adaptable to this is the scanning electron microscope. Its desirable qualities are the ability to resolve finer detail and provide more information than can be observed by using the light microscope, particularly of the surface of bulk specimens. The great inherent depth of field and the three dimensional quality of the SEM's images gives a better easier to interpret image than the light-optical microscope system. Although the SEM is usually thought of as a strictly "laboratory type of instrument, requiring many special considerations for successful operation, a special purpose instrument could be designed for the specific use of providing quality images from remote and relatively hostile environments,such as in earth orbiting satellites.

There are many factors involved in the design of and in selecting the design parameters for such an electron microscope. One of these factors is the ability to take advantage of the low pressure of the environment and eliminate the high vacuum system, thus significantly saving weight and simplifying the design. Other design changes, resulting from taking advantage of the special environment in which the SEM would exist, and from limiting the instrument's versatility, could be considered. Most of the controls such as focusing, magnification, selection of imaging area, beam deflection, and adjustment of many of the electron optics parameters can be made by direct electronic means rather than electronic electro-mechanical means. Thus they lend themselves to remote or automatic programed operation. Vibration of the SEM is usually a great problem, but once the instrument is launched into orbit, there should be very little if any detrimental vibration. Selection of the various design parameters such as the type of electron source, the types of detectors to be used, power supplies for acceleration of electrons, lens design for the desired range of magnification, sample handling, and transmission of the image to the viewer, will be discussed in this paper.

DESIGN PARAMETERS

Let us consider the design of an instrument with the following characteristics;

1. A resolution of 500 nm
2. A working distance of 25 mm
3. A depth of field of approximately 10 μm
4. A magnification range of 20X to 2000X in steps
5. A maximum viewing area of 2 mm^2.
6. Image return in a 1024 X 1024 digital format.
7. Power consumption within the capacity of the on-board batteries and the satellite's ability to recharge them, (50 watts maximum, 15 watts continuous.)
8. Weight less than 50 lbs.
9. Sufficient ruggedness to withstand the rigors and vibration of launch.

If we were to replace the high-vacuum system of the SEM and take advantage of the low pressure of the environment, we should consider how the quality of this available vacuum affects our design considerations.

SYSTEM COMPONENTS

Electron Source

Basic to the design is the type of electron source to be selected. The best understood and oldest type of electron source is the direct-heated tungsten filament used in a biased triode gun. The power consumption of this source is modest, but constant during operation. It also must be brought to saturation for stable operating conditions. Due to the constantly changing characteristics of the tungsten filament (the diameter of the wire decreases due to the evaporation of the tungsten from the hot wire), the parameters of saturation may be different for each time the filament is turned on. This can be minimized by slightly over-saturating the filament but at the price of decreased lifetime of the filament. The lifetime of such a filament operated at a pressure of less than 10^{-6} torr under normal saturation conditions is 50 - 150 hours, depending upon the quality of the vacuum. A clean vacuum (lack of oil vapors from diffusion pumps, etc.) would be expected, with the only contaminants arising from outgassing of materials in the equipment and reactive elements present. The tungsten source is simple, rugged, and supplies an adequate electron density in the beam for most applications. For typical electron guns of this type, the diameter of the source at crossover is 25 to 10 μm. For a tungsten filament operating at 2700 K, the beam brightness for 20 kV electrons is about 14 A/cm^2.

Another type of electron source is the Lanthanum Hexaboride (LaB_6) cathode. This gun has its advantage in the lower work function of LaB_6 over tungsten, and at temperature of 1500 K provides twice the current density of tungsten. At 2000 K, it has a brightness of about seven times tungsten. At 1850 K, LaB_6 has a lifetime of several hundred hours and at 20 kV gives a brightness of 65 A/cm^2. This is about five times greater than tungsten operating at its maximum temperature. Saturation of the LaB_6 source is accomplished by heating the tip of a sintered block of LaB_6, ground to a very sharp point, and applying a high potential between it and the anode. The energy required to heat the LaB_6 is rather modest, being in the order of 2 -5 watts, but, due to the fact that the LaB_6 is easily contaminated, it must be cleaned by heating to a high temperature before each use. This heating must be continued for as long as several minutes, to achieve reactivation. Care must be taken to avoid overheating. When properly activated the source will rise to its usual value of saturation without the sharp "knee" of the saturation curve of the tungsten filament. This process could probably be controlled by a small program in a microprocessor.

LaB_6 is highly chemically reactive when hot and forms compounds with all elements (except carbon and rhenium). These contaminants "poison" the cathode, with the result that it is no longer an electron emitter. Thus LaB_6 requires a very high, very clean vacuum. At any pressure above 100 μPa, an oxide forms, impairing the performance of the emitter.

The electron source that provides the greatest brightness of all is the Field Emission Gun. The cathode of this gun is a very sharp rod, with a point radius of less than 50 nm. When this tip is at a strong negative potential with respect to the anode, the electric field at the tip is so strong (10^7 V/cm) that the potential barrier becomes very narrow and reduced in height. This permits electrons to "tunnel" directly through the barrier and leave the cathode without the necessity of cathode heating. The cathode current density can be between 1000 and 10^6 A/cm^2. This gives an effective brightness of many hundred times that of a heated tungsten source. The cathode material is usually a single crystal of tungsten, with a chosen orientation, of very clean material. A single atom of contamination can increase the work function and lower the emission. The tip can be cleaned and reactivated by heating to a high temperature in a very clean vacuum. Usually the tip is kept warm (800 -1000°C) to immediately reevaporate contaminants. The required operating pressure is 10 nPa or lower. This pressure may not be obtainable in lower earth orbits.

The effective size of the Field Emission source at crossover is only about 10 nm, compared with 10 μm for LaB_6 and 50 μm for thermionic tungsten. This small size does not require any further demagnification to provide a sufficiently small electron probe for high resolution SEM. With the Field Emission gun, only a single objective lens is used to provide focusing on the specimen and a "hinge point" for the scanning coils. This is usually a magnetic lens. The high voltage requirements for this type of source are a voltage of about 3 kV between the tip and the first anode (to control the beam current) and a voltage of up to 30 kV between the tip and the second anode to accelerate the electrons.

Thus the type of electron source can be selected from the operating environment of the instrument: thermionic tungsten for low orbital use, LaB_6 for higher orbits, and perhaps Field Emission sources for extreme altitudes.

Electron Optics - Column Design

For practical purposes, the size of the exploring probe determines the resolution of the SEM and therefore its purpose and capabilities. In order to obtain the greatest signal to noise ratio, it is necessary to have sufficient energy in the exploring probe beam to provide the signal for the detector. When these parameters are determined, they more or less dictate the type of electron optical column used.

In the design of an SEM for lower satellite orbit, the quality of the available vacuum dictates the choice of a thermionic tungsten source. With a source diameter of about 50 μm, the electron optic column would have to demagnify this source at least 1000 X. This could be done quite readily by a compound optical arrangement of two tandem stages of demagnification. Two condenser lenses with focal lengths of approximately 5 mm could accomplish this in a reasonable column length. These lenses could be of fixed focal length and therefore be energized by permanent magnets. In a suitable arrangement of the pole pieces, a single permanent magnet in the form of a cylinder of sintered light-weight magnetic material and of suitable length could excite both lenses and provide less leakage flux. The remaining magnetic flux could be "shunted out" by multiple layers of high coercivity materials until the level of the remaining leakage flux was negligible. Another method of canceling out the leakage flux, is to wrap coils of wire around the outside and pass a current through it to produce an magnetic field that would oppose the leakage flux field. This would consume power, and although a rather small amount, it would be a continuous power consumption. In some situations,it would be worth the expenditure of the power. A permanent magnet may also be used to oppose the leakage flux.

The exploring probe (or spot) would be focused on the specimen by a final or objective lens that would be either a fixed-focus permanent magnet lens, or a combination of a permanent magnet and an electromagnet coil to shunt it and provide adjustable focus. The energy consumed by the electromagnet shunt coil would be considerably less than in the case of using an electromagnet for the total excitation of the objective lens. Another arrangement to focus the spot on the specimen could be to use all permanent magnet excited lenses, and then to vary the accelerating or beam voltage. This also is an "all electronic" type of control that lends itself to remote operation.

An electrostatic lens system could also be used in the column to demagnify the image of the electron source. Electrostatic lenses focus the electrons by means of shaped electrostatic fields, the shape of which is established by the shape of the electrodes of the lens and the potential gradient through which the electrons pass. The electrodes must be axially symmetrical, but may be of light-weight metal. They are usually designed so that a set of three electrodes comprises each lens element, with the correct potential being supplied to each element. A relatively high potential must be applied to the electrodes to form the electrostatic fields. This is of the same order of magnitude as the accelerating voltage, and may be obtained from the accelerating voltage if it has a sufficiently high current capacity. The lenses themselves do not require a high current, but the voltage dividers used to adjust the electrodes to the required potentials draw some current. Separate high voltage supplies for each lens perhaps could be a more power-economical design, but this method may require more weight. One advantage of operating the lenses from the same power supply as the accelerating voltage is that variations in the supply voltage affect both the lenses and the velocity of the accelerated electrons in the beam, and the effects cancel each other out to a high degree. This of course would not be true with separate power supplies. Although the "electret" is the electrostatic equivalent of the permanent magnet, using electrets in combination to produce electrostatic lenses has, to our knowledge, not been done and may require considerable development. Electrostatic lenses are, however, already in a form relatively easily controlled by remote means, in that the adjustment of the focus is done by varying the potential applied to the sets of lenses in common.

Some of the disadvantages of the electrostatic lenses is their inherently higher coefficient of spherical aberration. Since both "positive" and "negative" optical elements can be produced in electrostatic lenses (magnetic lenses only produce "positive" optical elements), a combination of both elements can to a degree correct this, but only to a lesser extent as compared to the magnetic lenses, and then with a more complex power supply situation. For relatively low-magnification applications, this may not be too great a disadvantage.

Correction of most of the other aberration are carried out in the same manner for both systems. Limiting apertures may be used in the same way and the asymmetries of the lenses may be corrected by the same devices, either electrostatic or magnetic.

Electrostatic lenses are sensitive to magnetic fields and so require additional magnetic shielding from these fields if they are present. Due to the high voltages on the focusing electrodes of electrostatic lenses, they require insulation and greater spacing than the magnetic lenses. This may add appreciably to their weight and also require a longer column.

By "potting" the lenses in a insulating plastic, the insulation problem and the ability to withstand shock and vibration could be greatly improved. Many of the other requirements common to both magnetic systems and electrostatic systems, such as positioning of apertures, stigmators, etc. could be accomplished by this method.

The electron beam for either electrostatic or magnetic lens systems could be deflected (to produce the scanning raster) by either an electrostatic deflection system or a magnetic system. With electrostatic deflection, the potentials on the deflection plates must be quite high, usually in the order or hundreds of volts, to deflects beams of several kV energy. Where solid state amplifiers are being used, and with a 12 Volt power supply, this may require more weight and power than the straight electromagnetic deflection.

Even in considering the disadvantages of the electrostatic lensing system, it could be considered for use in many situations.

To obtain a high resolution image, it is necessary to provide a very symmetrical spot of electrons for the exploring probe. The "stigmator" for adjusting this could be radial shunt screws or small permanent magnets placed in the gap of the objective lens and would be adjusted before launch. The beam confining apertures likewise could be pre-set and (to insure stability and ruggedness) potted during column construction. The parameters for these "stigmator" adjustments do not change appreciably for different magnifications, which allows potting.

Deflection fields for the electron beam to provide the raster on the specimen would be created by coils or electrostatic deflection plates located in the conventional position between the condenser lenses and the objective.

Magnification Control

Magnification control of an SEM is accomplished by changing the relative size of the viewing and the specimen scanning rasters. The means for this is merely attenuating the current to the scanning coils. Here again this lends itself very well to entire electronic-remote control. Probably a series of step controls would be the best. A series of 2 - 5 - 10 steps could provide a convenient arrangement.

High Voltage Supplies

The accelerating voltage for the electron beam can be generated from compact power supplies, since the current requirements are relatively low - in the order of 50 to 75 uA.

The choice of the accelerating voltage would be governed by several factors. Although higher voltages give higher velocity electrons (with shorter wavelengths) and thus a higher ultimate resolution, this is of little advantage at the relatively low magnifications used here. The electron wavelengths are already several hundred times shorter than is necessary. By choosing a high voltage of only 3 kV to 5 kV, the lenses require magnetic or electric fields of lower strength. The deflection system requires less power, and there is still sufficient energy in the electron beam to excite secondary electrons and backscatter electrons from most specimens. With lower accelerating voltages, the problem of charging of the specimen is less likely. This phenomenon usually occurs only when attempting to look at highly dielectric specimens. The surface or volume resistivity for most materials is sufficiently low to provide a reasonable return for beam electrons impinging upon the specimen. Choosing a lower voltage for the accelerating voltage power supply requires less high- voltage insulation. The problem of high-voltage cabling from power supply to electron gun is eliminated by mounting the high- voltage supply directly on the electron gun. The bias of the triode gun would be provided by a self biasing arrangement. The high voltage for the Everhart-Thornley scintillator- photomultiplier secondary electron detector is 10 - 12 kV but is also low in current requirements. The photomultiplier requires an 800 - 900 volt power supply of low current. All other power requirements fit within the normal 5 - 12 volt range.

Field of View Selection

Selecting the field of view of the specimen is an important consideration. Within the limits of the raster area of the lowest magnification, the raster fields of higher magnification could be moved about by applying a constant biasing current on the beam deflecting-scanning coils. For larger movements, mechanical movement of the specimen would probably be desirable, although more complex.

The scanning generator synchronizing signals could be either generated within the unit or transmitted to the instrument from a remote location.

Detector Selection - Types

The detector would depend on the type of signals desired to be obtained from the specimen-electron beam interaction. Backscatter electron detectors could be of the large area silicon diode type. It would probably be worthwhile to utilize the Everhart-Thornley scintillator-photomultiplier type of detector for secondary electrons due to its low power consumption and very good signal to noise ratio. Other types of detectors, or several at once, could be used, depending on factors such as beam energy (accelerating voltage), type of specimen, etc. Some types of detectors are very simple. For example, the absorbed-current detector is merely a resistor to ground in series with the specimen. Cathodoluminescence detectors may become quite complex with their light-focusing devices. Most detectors and their requisite amplifiers are relatively low in power consumption.

If sufficient accelerating voltage is used to provide sufficient energy to the electrons of the beam, an Energy Dispersive X-ray detector can be used to provide elemental analysis information. The conditioned detector output pulses could be sent by telemetry to the analyzer for processing. The main drawback with this detector is that the detector diode and FET pre-amplifier must be kept at liquid nitrogen temperatures to minimize the lithium drift in the detector diode, and to lower the noise of the FET. The only possibility readily apparent for this cooling is to sink the heat to outer space on the shadowed side of the satellite. This detector also needs a fairly high bias voltage - about 300 volts. All things considered, operation of an EDS X-ray system from a small, low-earth-orbiting satellite does not look too promising.

Image Transmission

Probably the best system of transmitting the image signal would be to convert it to a digital signal and then transmit this signal. This has many advantages in that it is easy to transmit, is less subject to noise interference, and also is in a form for easy image enhancement, processing, presentation, and measurement. Another advantage of a digitized signal is the possibility of recording (or capturing) a single image in a storage buffer (frame grabber) at one signal rate, and transmitting this image at some different rate.

Mechanical Design - Operational Procedures

The overall outline of the instrument could be very compact, perhaps with the circuit boards, power supplies, and detector amplifiers located radially around the central electron optics column.

Procedures for operation of such an instrument as this from a remote location would have to be considered. For example positioning of the raster on the specimen or movement of the specimen could be of the form of step-and-repeat, or by means of "escapement" type operation. Because of the time taken to transmit a complete image, it may be necessary to focus by means of steps, while observing the high frequency component of the reconstructed video signal, or by a through-focus series of images.

CONCLUSIONS

The results of this study show that a remotely operated sem, within the limitations of specific design applications, could be successfully built, and by taking advantage of the special environmental conditions available, could be integrated into a small earth-orbiting satellite. Much useful information about materials, material changes, alloys, fine structure degradation of materials, and material properties could be obtained by the use of this type of instrument. Currently there is no instrument with these capabilities available.

EXPERIMENT INTEGRATION ON SPACECRAFT LIPS III

J. G. Severns, R. M. Hobbs, R. H. Towsley, and J. G. Winkler
Naval Research Laboratory
Washington, D. C. 20375

The LIPS design effort converted a sheet metal plume shield into a "living" spacecraft, with power supply, R.F. receiver and transmitter, sensors to make experimental measurements and monitor satellite health, etc. LIPS III was built to provide a test bed for new and innovative space power sources, with some 140 photovoltaic experiments being submitted by 18 laboratories, all of which were placed in orbit. This paper describes the LIPS III system with special emphasis on mechanical and electrical integration of the experiments. Results after one year of operation are described, and although most experimental data is proprietary and thus cannot be published here, enough data is included to indicate the general quality of the results.

INTRODUCTION

LIPS III is the third in the series of "Living Plume Shield" spacecraft, all designed and built at the Naval Research Laboratory. In each LIPS project the plume shield, a simple sheet metal cone, was structurally stiffened, and an active satellite was then built around it. The original purpose of the plume shield was to prevent the plume from solid propellent engines, which are fired outside the atmosphere after the aerodynamic shroud is jettisoned, from reaching the primary payload. The surface of LIPS III facing the plume also functioned in this manner, but the anterior surfaces were unaffected, and it was there that all solar arrays, sensors, and experiments were mounted.

The purpose of LIPS III was to provide a test bed for new space power sources. With the long delays projected for schedules of the STS and other major launch systems, it appeared that a decade might pass before long term flight data could be obtained on many new and innovative power sources. The fact that a launch scheduled for early in 1987 required a plume shield was seen as a unique opportunity to obtain some of this data in a timely manner. The LIPS III system will be described below along with the experiments placed on board with emphasis placed on the procedures for electrical and mechanical integration.

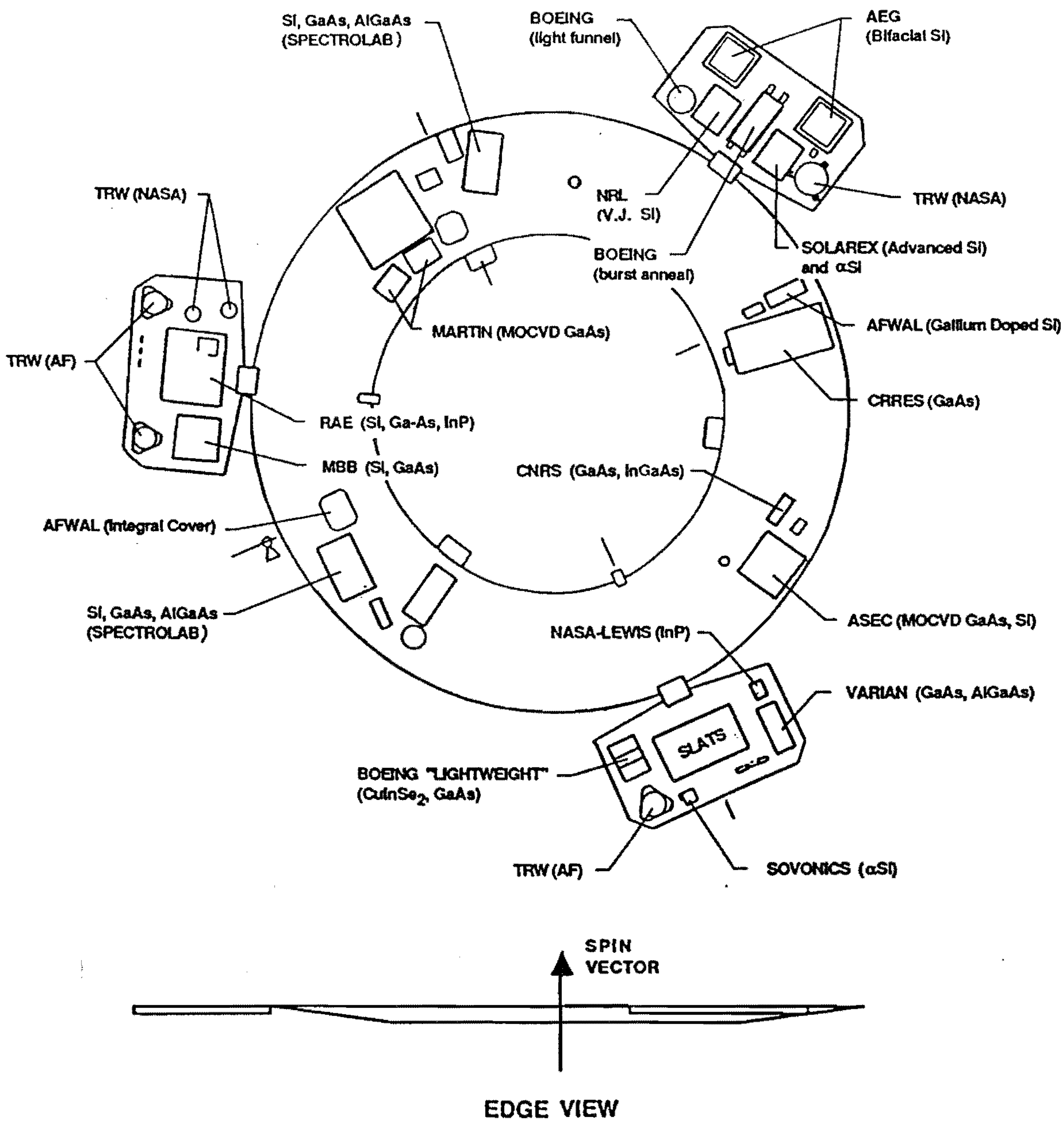

FIGURE 1
LIPS III EXPERIMENT LAYOUT

THE LIPS III SYSTEM

LIPS III was injected into a nearly circular orbit of 1100 kilometer attitude with inclination slightly in excess of 60° late in the spring of 1987. The planned mission life of LIPS III is 3 years, with hopes of attaining 5 years. The original plume shield was a very shallow cone, and the spacecraft structure is similarly shaped, as shown in Fig. 1, with an outside diameter of 74 inches and a height of only 4 inches. This required that electronic boxes and other components to be mounted inside the satellite were indeed "low profile." The three paddles shown were folded over the annular spacecraft body during launch and subsequently deployed as shown. The paddles were included to provide area for mounting experiments.

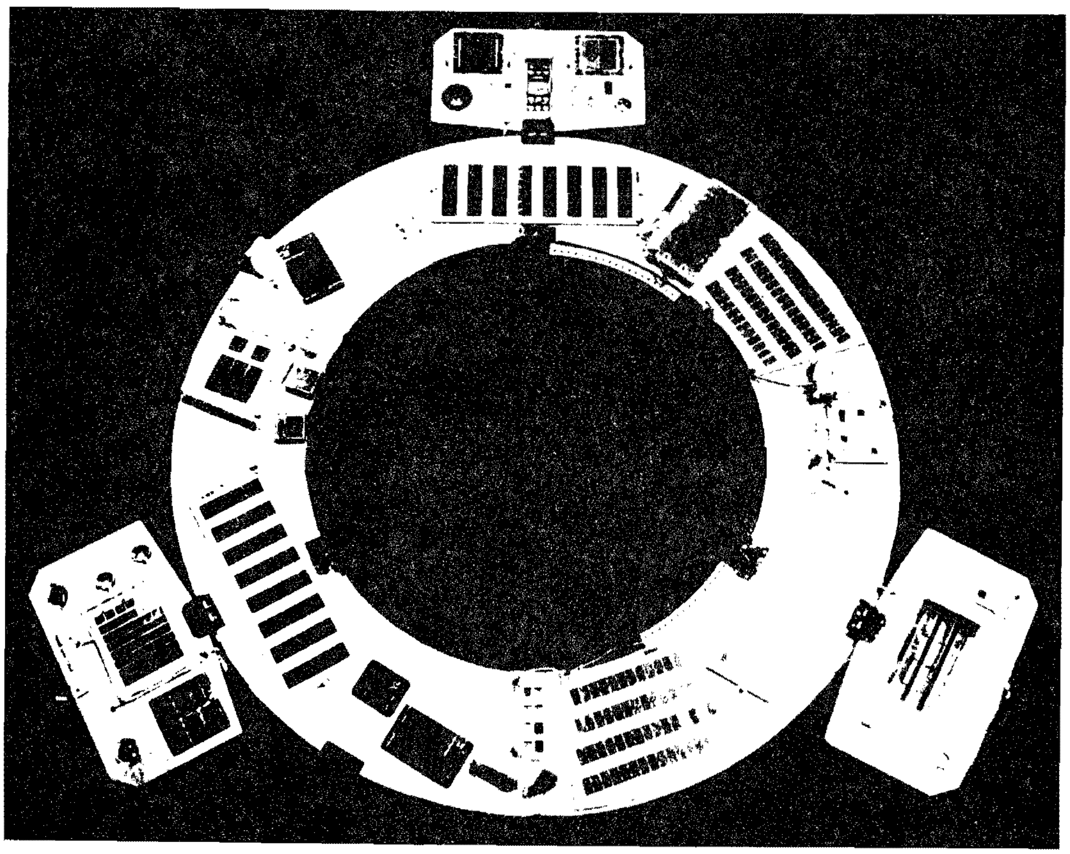

FIGURE 2
LIPS III PADDLES IN FLIGHT CONFIGURATION

The spacecraft is spin stabilized with the spin axis pointed toward the sun so that all experiments receive constant illumination. To meet the sun pointing requirements of those experiments involving solar concentrators, the attitude error between the spin vector and the sun vector was limited to ± .5°. The spacecraft EPS is powered by five silicon solar panels mounted on the annular spacecraft body and generating 25 watts (BOL). This solar array drives an unregulated bus operating at 14.5v ±2v with a 6 A-H NiCd battery for storage. The telemetry system encodes analog voltages between 0.0v and 5.1v, producing an eight bit "word" for each. Digital quantities (with only values of "0" or "1") are also included in the data stream, typically with eight digital values lumped together for transmission in one 8 bit telemetry word. These digital words are interleaved with the analog quantities in the telemetry format. Also interleaved with housekeeping information is data from the experiments. Every housekeeping quantity is transmitted once during each 68 word telemetry frame, along with one value of current, voltage, and temperature sensor resistance from one experiment in each of three experiment data channels. Each of these data channels is submultiplexed so that a complete set of experiment data is eventually transmitted to the ground. The data is transmitted via an L band downlink at a rate of 2441 bits/sec. No provision was made for on-board data storage, so data can only be obtained in real time when the spacecraft is in view of a tracking station.

The thermal control system is an all passive design, with multilayer thermal blankets covering most of the satellite surface. The paddles were not blanketed, but were coated with silver-teflon tape. The blanket on the non-illuminated side of LIPS provided freedom for thermal properties of the outer blanket surface to change markedly during impingement of the plume without affecting subsequent thermal performance. The interior ring of the structure (which is 4 in. high) was painted black and served as the waste heat radiator.

The operation of the attitude control system (ACS) will now be described. For a more detailed discussion of the design and operation of the ACS, see Ref. 1. This system was designed to operate in two modes. At orbit injection, the spacecraft was expected to be spinning at 30 RPM with its spin vector pointed 120° from the sun vector. A hydrazine thruster was designed to be pulsed each time the spacecraft rotation put it in the appropriate position to reduce the angle between sun and spin vectors. The pulse timing is determined from a wide angle sun sensor with a "fan shaped" field of view (FOV=4° x 180°). This sun sensor was mounted on the outer edge of the spacecraft with the plane of the "fan" containing the spin axis. As LIPS rotates the fan sweeps across the entire celestial sphere and when the sun crosses the field of view, the angle between the spin and sun vectors is measured. From the time between consecutive measurements, the next hydrazine pulse time is calculated. Once the spin vector was brought within five degrees or less of the sun, the second mode of operation began which relies on a magnetic interaction to control attitude. A solenoid (Ref. 2) was mounted in the spacecraft so that when a pulse of current flowed through its windings the resultant dipole moment was perpendicular to the spin axis. This dipole couples to the earth's magnetic field to generate a torque on the spacecraft which will change the spin rate if the torque is parallel to the spin vector, or change the spin vector direction if the torque and spin vector are perpendicular. Thus, depending upon the instantaneous direction of the earth's field, either spin rate or direction (or both) can be changed, and thus controlled. To implement this magnetic control system, a three axis fluxgate magnetometer and a fine angle sun sensor were placed aboard the spacecraft. The design ensured that measurements of the earth's magnetic field are accepted from the magnetometer only when the solenoid is not energized. The fine angle sun sensor has a useful field of view of 5° with a resolution of .05° . Since this magnetic control system is currently operating with errors ≤.1° , we plan to rely on this mode of operation for the remainder of the mission.

THE EXPERIMENTS

With the exception of a study of material properties of selected thin films submitted by the Martin Marietta Astronautics Group, all of the experiments aboard LIPS III were photovoltaic in nature. It is unfortunate that the small size of the spacecraft and very tight schedule precluded integration of other types of energy conversion experiments, such as solar dynamic systems. As can be seen from Table 1, experiments were submitted by 18 laboratories involving cells of silicon (both crystalline and amorphous), GaAs, AlGaAs, InGaAs, and $CuInSe_2$. One of the primary goals of the LIPS III effort was to realistically test photovoltaic concentrators in space; there are three concentrator concepts represented on LIPS III. In all, 140 separate I-V characteristics are measured. These span short circuit current values of 1.2 $X10^{-3}$ amps up to 2.0 amps, with open circuit voltages varying from .35 volts to 6.0 volts. Temperatures are sensed by measuring the resistance of a sensor, and including resistance values of the non-photovoltaic thin film experiment from the Martin Company, resistances from 40 ohms to 24 megohms are measured. These experiments are housed on some 30 panels or fixtures, nearly all manufactured by the experimenters and integrated by NRL.

It should be pointed out that the LIPS experimenters met an extremely tight schedule, building and testing their experiments in a time so short that many had no chance to obtain official support for their efforts from their respective

Table 1

GaAs, AlGaAs, InGaAs

Applied Solar Energy Corp	2cm x 2cm and 4cm x 4cm GaAs Cells: GaAs cells on Ge substrate - four cells in series
AFWAL/POOC-2	CRRES Ambient Panel (Backup) - 10 four-cell strings comparing coverslips, coatings, and adhesive
Boeing	Small Area "Concentrator" Cells: LPE (Spectrolab); MOCVD (ASEC); MOCVD (Kopin)
CNRS (France)	MBE GaAs - two cells; LPE InGaAs - two cells
MBB (Munich, FRG)	Two cells (MELCO)
Royal Aircraft Estab. (Farnborough, Hants, UK)	Four cells in series (MARCONI)
Spectrolab	Two GaAs (Spectrolab) and two AlGaAs/GaAs (HRL)
VARIAN	GaAs, AlGaAs - nineteen cells in all: InGaAs under inactive AlGaAs; AlGaAs grown on inactive InGaAs
Martin Marietta Astronautics Group	MOCVD GaAs - compares welded and soldered interconnects

Single Crystal Silicon

AEG (Wedel, FRG)	Bifacial Cell Array - four 5cm x 5cm cells; Two very light weight support structures tested
Applied Solar Energy Corp	2 mil Cells (2cm x 4cm)
AFWAL/POOC-2	Integral Coverslip covering eight cells; Gallium Doped Silicon Cells
Boeing Co.	Four cells with high temperature contacts ("Burst" Annealable)
MBB (Munich, FRG)	Five advanced design cells
NRL/Solarex	Six Vertical Junction Cells (Comparing various coverslips & adhesives); Advanced Design Planar Cells
Royal Aircraft Estab. (Farnborough, Hants, UK)	Four cells comparing coverslips & coatings; Two designs for high radiation resistance
Spectrolab	Two advanced design cells

InP

Royal Aircraft Estab. (Farnborough, Hants, UK)	Four cells fabricated by Newcastle University
NASA-LeRC	Four cells fabricated by Rensselaer Polytechnic

Thin Film Cells - $CuInSe_2$ and Si

Boeing Co.	Three series strings of four $CuInSe_2$ cells each
Sovonics	Two α Si Cells
Solarex	Two α Si Cells

Concentrators

Boeing Co.	Six light funnels

managers before starting work. Integration meetings were usually held by telephone and telecopier. Throughout this hectic time, high standards of workmanship were adhered to as evidenced by the almost complete success (to date) of the experiments.

MECHANICAL INTEGRATION

Equipment layout on LIPS III was dictated by several requirements. The need of electronics boxes and the battery for structural support, waste heat removal, and electrical connection to a wiring harness suggested mounting these components to the inner ring which as discussed above, was designed to operate as a waste heat radiator. A view of the interior of the spacecraft during

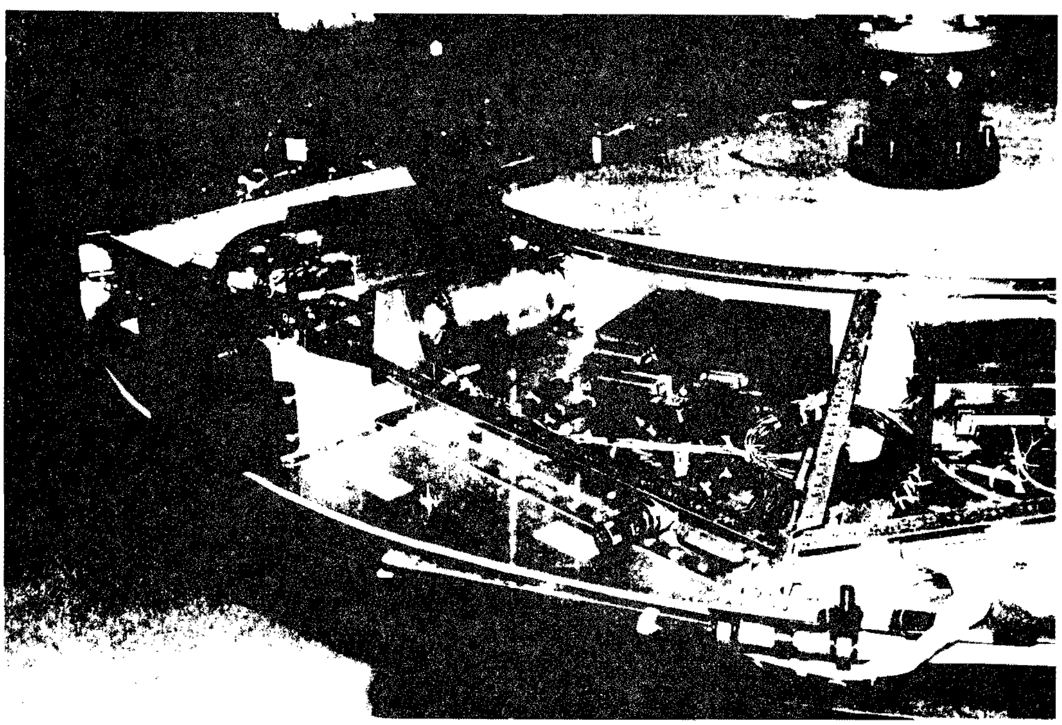

FIGURE 3 LIPS III INTERIOR

LOWER FOREGROUND SHOWS PADDLE HINGE AND CABLE. ELECTRONICS BOXES AND A HYDRAZINE TANK ARE MOUNTED TO THE INNER RING IN THE BACKGROUND.

assembly is shown in Fig. 3, where it can be seen that electrical connectors all face radially outward for ease of coupling into the ring like harness. Placement of experiments was driven by their need for constant illumination and requirements to reject waste heat. As mentioned above, several experiments involve solar concentrators which require both accurate alignment to the sun and highly efficient waste heat radiation to space. These sun pointing needs required both a tight tolerance on the attitude error and careful alignment of the paddles, on which the concentrators were mounted, so that in their deployed configuration they would remain perpendicular to the spin vector. The waste heat rejection requirement was met by mounting the concentrator experiments over holes cut in the paddles for this purpose, allowing a clear view of space for radiation from the back (nonilluminated) surfaces. The shallowness of the LIPS structure was a distinct advantage here, since it eliminated any radiative interaction between the spacecraft body and the concentrator waste heat radiators. Another advantage of mounting experiments on the paddles is the improved knowledge that this gives of the charged particle radiation environment. The back shielding of the spacecraft structure is negligible except for that due to the paddle, which can be characterized quite accurately. It was not possible to mount all experiments on paddles, so some of them were mounted on the annular spacecraft body. Most experiments placed on the body were mounted over the multilayer thermal blanket on standoffs that passed through the blanket. This procedure effectively eliminates radiative heat removal from the back of these experiments, causing them to operate at higher temperatures (80-100 °C) compared to the paddle mounted experiments which operate at 20-50 °C.

The precise pointing needed by the concentrators required that each concentrator experiment be aligned with the spacecraft spin axis before the vibration test, and then checked for changes after vibration. The alignment procedure was facilitated by requiring that a small mirror be placed on each concentrator module with its normal parallel to the optical axis of the experiment. The addition of a very small mirror added almost no weight and was not difficult to implement in most cases. The alignment procedure was as follows: The spacecraft was mounted on the spin balance machine in such a way that the satellite spin axis was colinear with the axis of the machine. A mirror was then mounted on the end of the spindle with the plane of the mirror perpendicular to the machine axis. The spacecraft was then rotated until the paddle with the concentrator of interest was pointed straight down; i.e., gravity being perpendicular to the paddle hinge axis. It was convenient that the force of gravity was a good approximation of the centrifugal force that would act on the paddle with the spacecraft rotating in space at 30 RPM. As shown in Fig. 4, a He-Ne laser was mounted to an Ealing optical rail so that it could travel ±5 meter along a highly repeatable horizontal line perpendicular to the laser beam. This rail was mounted on a large optical quality Brunson

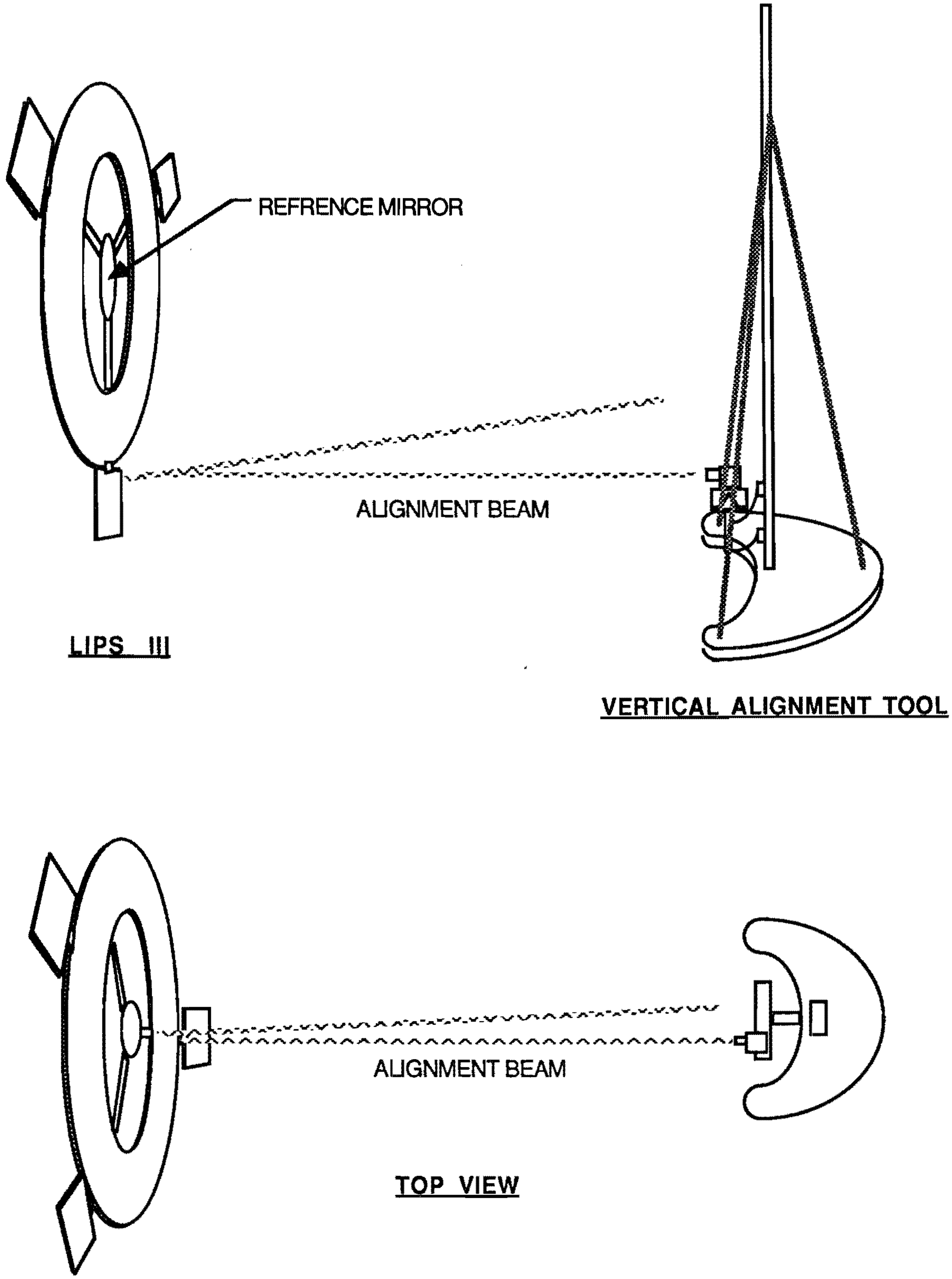

FIGURE 4
CONCENTRATOR ALIGNMENT

upright stand. The stand permitted vertical motion of the rail with the laser mounted on it. This arrangement made it possible to establish a family of parallel lines, all fairly distant from each other. The alignment procedure was begun by autocollimating the laser beam from the mirror mounted to the spin balance machine, establishing that beam as parallel to the spacecraft spin axis. The laser was then transported along the rail and upright stand until its beam struck the alignment mirror on the concentrator. The return beam gave a direct measure of the misalignment of the concentrator, guiding the process of shimming the experiment mount. The beam was periodically rechecked by returning it to reflect from the central mirror. From independent measurements made on the rail and stand, the 3σ error in parallelism between two lines is less than 5 minutes of arc. The goal for this effort was to assure that all concentrators remained aligned to the sun with less than 0.25° error, a goal which appears to be met nearly 100% of the time, exceeding that value only on the one occasion when the attitude control error grew larger than this.

ELECTRICAL INTEGRATION

The electrical subsystem design was dictated by the restrictions on size, weight and schedule. Fortunately, several of the subsystems could be adapted from the LIPS I & II spacecraft. These included power, ordnance and RCS electronics. The telemetry subsystem employed the same multiplexors and power supply as LIPS II but a microprocessor was utilized to expand its function to include the ACS processing and control. The command subsystem was essentially the same as LIPS II but redesigned to accommodate serial type commands. The RF subsystem employed the same type receiver as LIPS II with redesigned antennas. The transmitter was a new design. The experiment data acquisition electronics was arranged in three channels. Each channel consisted of a relay multiplexor unit, a measurement unit and a power supply. Each channel serviced a deployable paddle and the body mounted experiments contiguous to it. This allowed for the shortest possible conductor paths.

The support subsystems were integrated first and a breadboard of the telemetry subsystem was used to check them out. As experiments and experiment channel electronics became available they were integrated and checked out.

The most difficult testing operations involved the experiments and the ACS. The experiments required a well collimated solar simulator and careful alignment. The subcommutated data made this a lengthy process. The ACS testing required that the spacecraft be rotated and that the existing earth magnetic field be utilized. A solar pulse was also required. Finally, after all testing was completed, the PROMs in the telemetry unit could be programmed and installed.

After carefully spin-balancing the spacecraft it was subjected to thermal-vacuum and vibration tests. Field tests included an illumination of all experiments using the sun as a source. The RCS tanks and fuel were installed in the spacecraft, final mechanical and thermal closeouts were performed and the spacecraft was installed on the host vehicle preparatory to final spin-balance.

EXPERIMENT DATA ACQUISITION

Provision was made for measuring the current-voltage (I-V) characteristics of each experiment both in sunlight, and the forward characteristic in darkness for those experimenters requesting it. A four terminal measurement was planned so that only very small currents would flow in the pair of wires monitoring voltage. Due to the large number of experiments, the grounding scheme became quite complex, so that one half of the experiment current is inadvertently permitted to flow in the voltage ground lead. Since this is a short (length ≤2 ft.) AWG#22 wire, the resulting underestimation of cell voltage was deemed unimportant. A given I-V curve is measured by varying a dynamic load through 24 points, measuring current and voltage values of each point. One value each of voltage, current, and temperature sensor resistance are measured simultaneously during each telemetry frame. The illuminated measurements were made in the following way:

a. First, the open circuit voltage of the cell is measured and the value is stored.

b. Next, the short circuit current is measured by driving the dynamic load to zero. The cell is slightly back biased to compensate for line losses between the cell and dynamic load.

c. Once I_{sc} is measured, values of voltage are selected at which the current is to be measured. The voltage points were planned to form an ascending array of values which steps out across the I-V curve from short circuit to open circuit. The target voltages are chosen in the following way: V_{oc} is divided into three parts, as shown in Fig. 5, corresponding to three regions of the I-V curve; region one runs from 0.0 volts to .4 x V_{oc}, and corresponds to the low voltage region of approximate constant current operation. Region two runs from .4 x V_{oc} to .65 x V_{oc}, well into the "knee" of the curve. The third region, running from .65 x V_{oc} to V_{oc}, represents the region of rapidly falling current as the open circuit condition is approached.

Region I is divided into four equal segments, region II into five equal segments, and region III into fourteen. The ends of these segments are the voltage values targeted for measurement. After each desired voltage value is calculated, the dynamic load is varied until the experiment operating voltage is equal to the calculated value. To improve resolution of the voltage measurement, two possible gain settings are provided, and the most appropriate setting is automatically selected for each measurement.

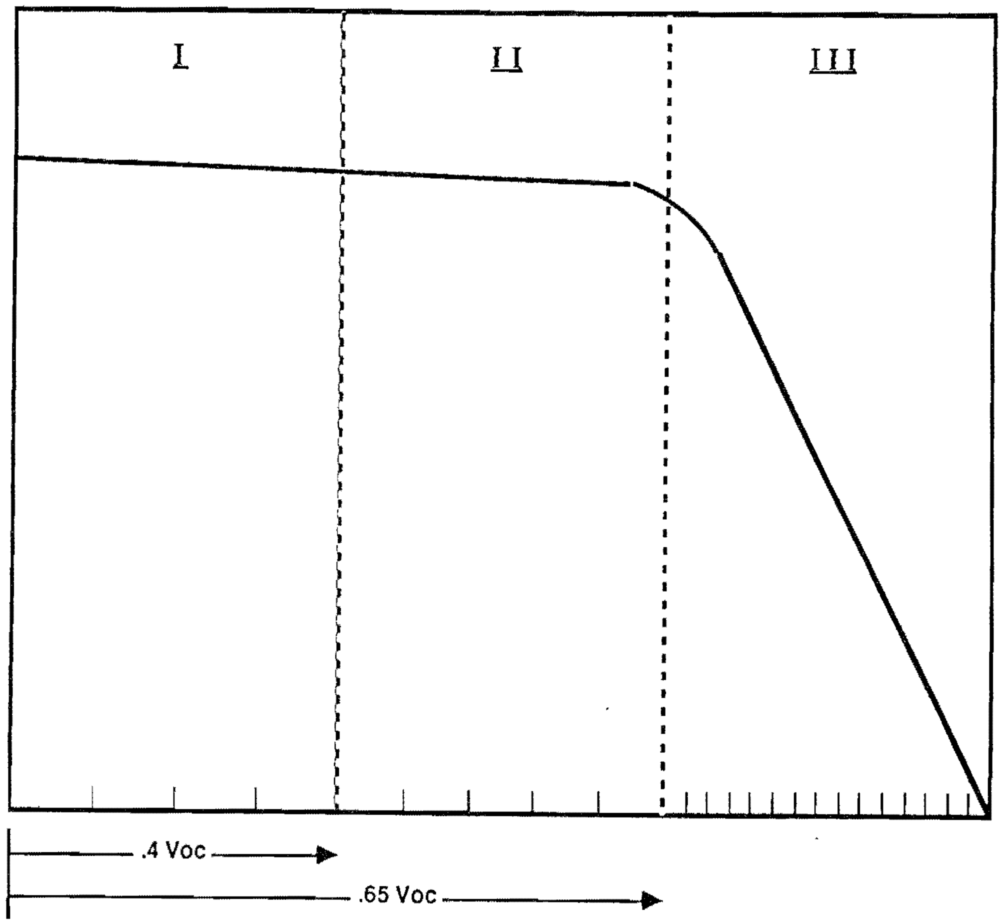

FIGURE 5
TARGET VOLTAGE POINT SELECTION

d. With the voltage value set, the current is measured. Since the experiment currents span such a wide range, there are seven possible gain settings for current measurement. An autorange circuit selects the gain giving the best resolution, and this gain setting with the analog values of current, voltage, and voltage gain are then available for telemetry.

e. The resistance of the appropriate temperature sensor is measured for every current-voltage point. This much temperature data is overkill, but it was quite inconvenient to do this any other way. In this way, 24 points (including I_{sc}) are measured for each I-V curve, the last point being V_{oc}, according to plan at least. Actual operation in orbit will be described in the next section.

The dark forward I-V characteristic is measured in a very similar way, except that the maximum voltage was chosen by the experimenter and stored in a ROM on the spacecraft. The 24 points are evenly spaced between V=0 and V_{max}.

There are three data channels each of which make the above measurements independently. These can each be turned on or off by ground command. Each channel has a compliment of up to 64 experiments through which it steps and a counter to indicate which experiment is being measured at any given time. Unfortunately, the values of these three counters were not included in the telemetry list. A master counter is included however, which can indicate the current status of each multiplexer. When this master counter runs through its complete sequence, it resets and issues a reset pulse that forces all of the data channel counters to reset, so thereafter, they are all in sync with the master counter. This causes some operational complexity since when a data channel is commanded on, the master counter must then be reset before any of the data is meaningful. This complexity is mitigated by use of a command that forces the master counter to reset on demand whatever its current value.

RESULTS

With the exception of the attitude control and experiment data acquisition subsystems, the spacecraft has operated for more than one year exactly as planned. The attitude control problems began when the spacecraft separated from the primary payload, a large nutation being introduced in the motion at that time. By judicious use of the hydrazine thruster during the first ten days of operation, the nutation was reduced sufficiently to permit use of the magnetic mode of the attitude control system. The attitude error was quickly reduced to small values; i.e., error <0.25°. The attitude control system has operated in this way ever since except for a two week period during which the geomagnetic field vector was not aligned in a direction to efficiently participate in correction of spin vector direction. For correction of spin direction, an appreciable component of the field must be along the spacecraft spin axis most of the time. Occasionally the precessing orbit puts the spacecraft in a position where the spin direction magnetic component is small

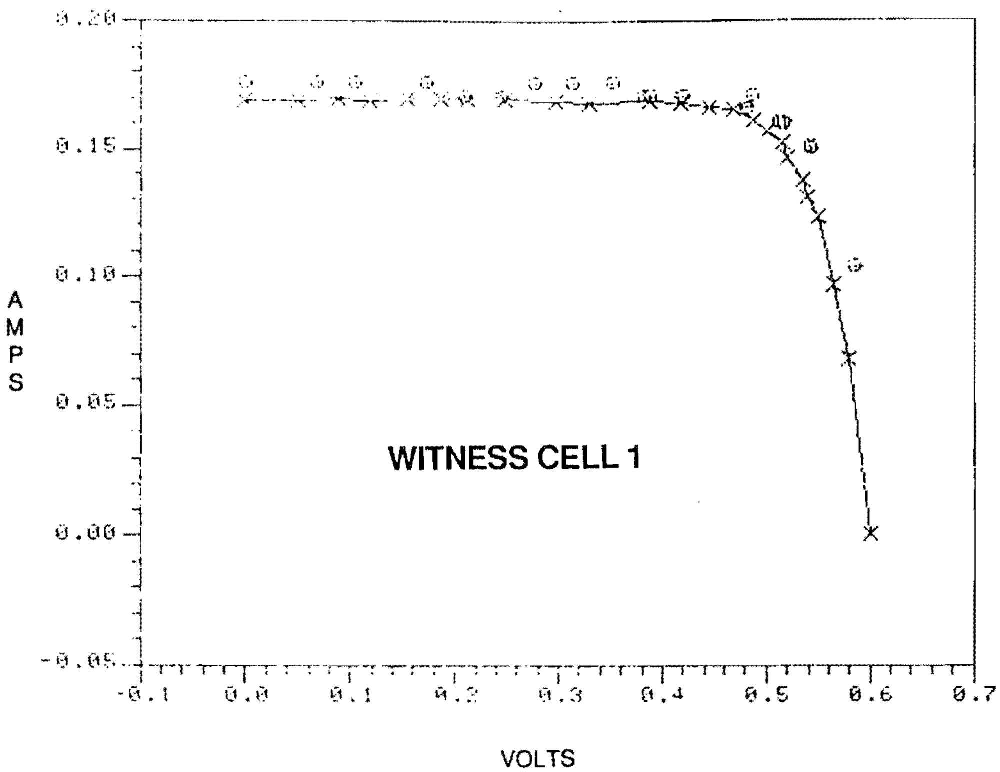

FIGURE 6

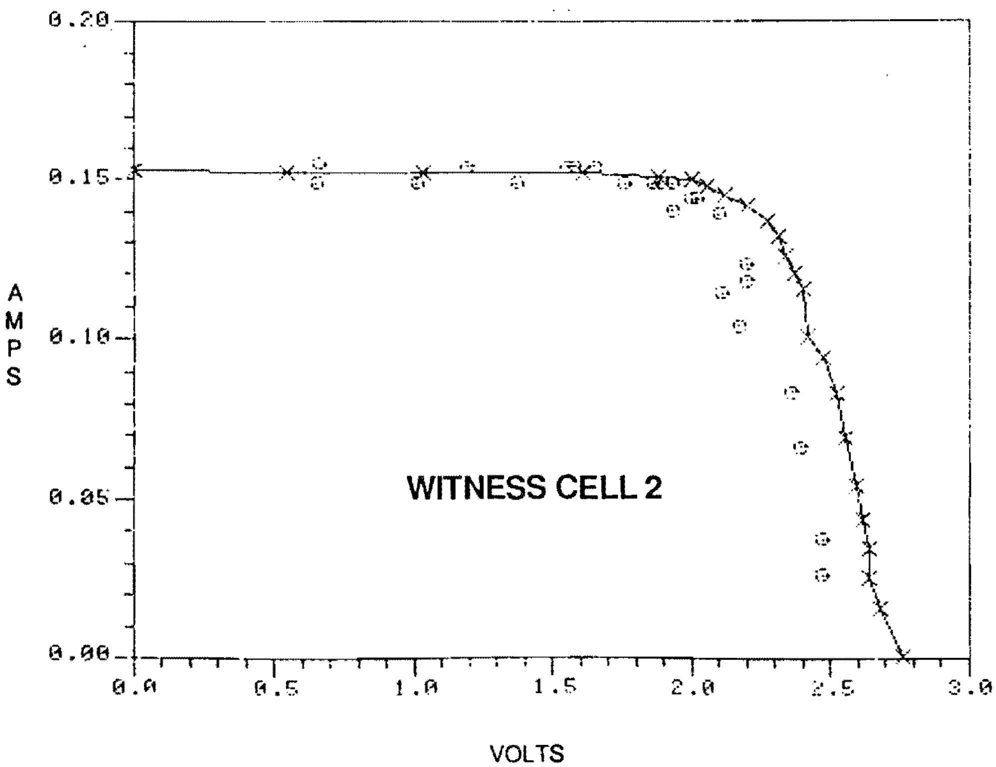

FIGURE 7

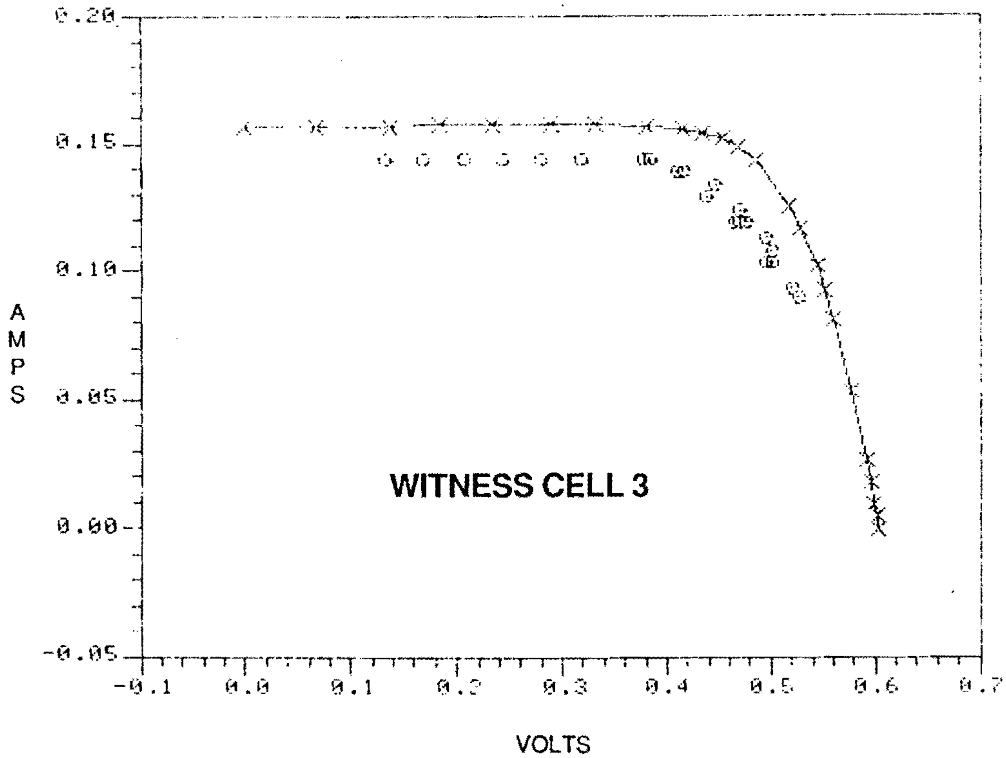

FIGURE 8

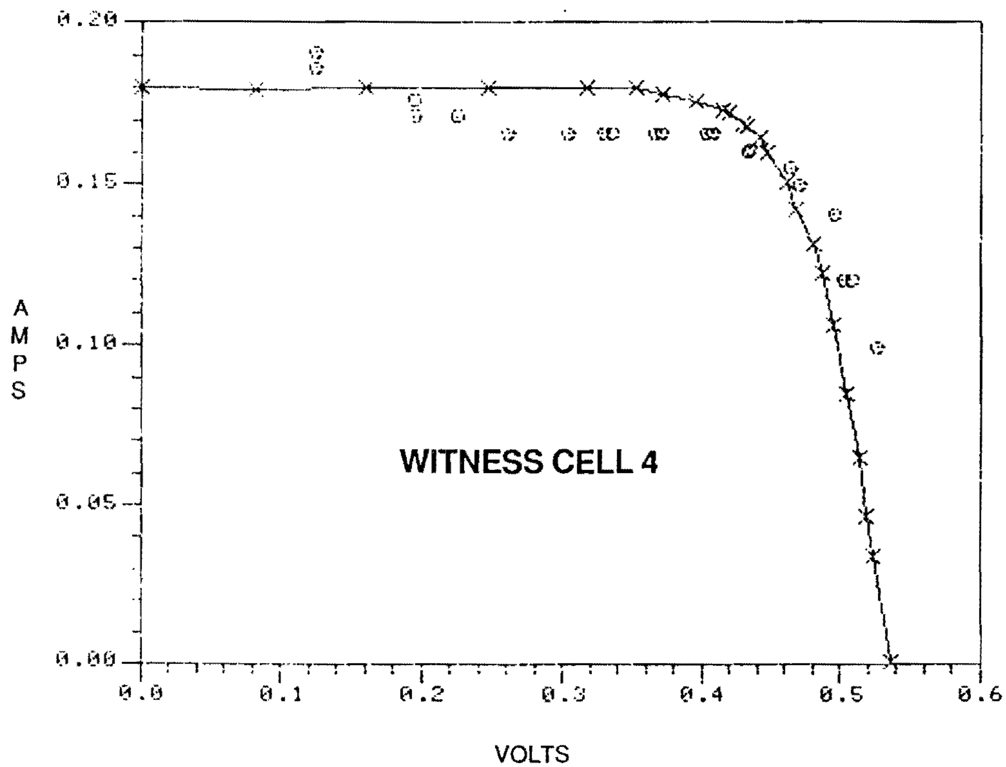

FIGURE 9

most of the time. This happened in June 1988, and the attitude error degraded to ~2.0° but the magnetic control system was still able to perform marginally during this period of inefficiency. Within two weeks of onset of this period, the attitude error was again <0.25°.

Difficulties have been experienced in collection of experimental data as discussed in Ref. 4. After integration with the launch vehicle the air conditioning introduced a layer of dust over the spacecraft which could not be cleaned off before launch. Some anomalies attributable to this dust layer have been observed in the experimental data, but fortunately only a few experiments seem to be affected. All other data problems can be lumped under the heading of electrical noise which causes a loss of short circuit current and open circuit voltage measurement. However, the remainder of the measured I-V characteristics appear to represent the real experiments with suitable accuracy. Shown in Figs. 6-9 are flight data from four so called witness cells, each plotted on the same axes with data taken before launch. The flight data shown is from shortly after launch, thus the flight data and ground data should be in close agreement. As seen from the figures, the agreement is not too bad when account is taken of the fact that the lamps used to generate ground data are only an approximation of the real air mass zero solar spectrum. It can be seen that, as mentioned above, the short circuit current and open circuit voltage measurements are rarely obtained from the spacecraft. The dust problem mentioned above could be the cause of the low current seen in Fig. 8 since the voltage does not seem to be as strongly affected as the current. In summary, the data appears to be of high enough quality to make it quite useful in assessing the long term stability and durability in space of the various experimental power sources studied here.

CONCLUSIONS

Even though, as mentioned in the last section, the data acquisition electronics does not operate as planned, the quality of the flight data is, in general, quite good. Almost all photovoltaic technologies for space (circa 1987) are represented on LIPS and, with the exception of four temperature sensors and two cells, the large number of experiments here survived integration, launch, and the first year of flight operation. Thus, after one year of operation an assessment can be made that, overall, LIPS III with its compliment of experiments, is a success.

The effort to design, fabricate, and test LIPS III was made difficult by the short time available for the task. Many people made outstanding efforts under the most trying of circumstances, without which success would have been impossible. The long list of those who contributed experiments is one of true distinction. Of particular importance to the spacecraft in general were the

contributions of experimenters Ted Stern, Micky Cornwall, Allan Dollery, Christopher Goodbody, and especially Gary Virshup. Members of the NRL staff whose efforts were exemplary are far too numerous to be listed here, but, aside from the coauthors, this paper would be incomplete without expressing special thanks to Robert Burdett, command system design and flight computer "Guru", Christopher Herndon, AGE design, Mark Johnson, telemetry design, David Hastman, thermal design, George Gregory, mechanical layout and design, Robert Morris, RCS design, Christopher Garner and Wilbert Barnes, NiCd Battery, Charles Morgan, ordnance design, William Webster, RF design, Joseph Valsi, wiring layout, Eric Eisler, RCS valve control design, Michael Mook, ACS system, Robert Conway, tracking station operations, Robert Grant and James Mills, system test conductors, and Joseph Delpino, launch integration specialist.

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