

A NOVEL SPACECRAFT CHARGE MONITOR FOR LEO

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Introduction

Five years ago we introduced a new method for measuring spacecraft chassis floating potential relative to the space plasma ("absolute" spacecraft potential) in low Earth orbit. The method, based on a straightforward interpretation of photoelectron spectra, shows promise for numerous applications, but has not yet been tried. In the interest of testing the method, and ultimately supplying another tool for measuring absolute spacecraft charge, we are producing a flight prototype Spacecraft Charge Monitor (SCM) with support from NASA's Small Business Innovation Research (SBIR) program. Although insight into the technique came from data collected in space over two decades ago, very little data are available. The data indicate that it may be possible to determine spacecraft floating potential to within 0.1 volt each with the SCM second under certain conditions. It is debatable that spacecraft floating potential has ever been measured with such accuracy. The compact, easily deployed SCM also offers the advantage of long-term stability in calibration. Accurate floating potential determinations from the SCM could be used to correct biases in space plasma measurements and evaluate charge mitigation and/or sensing devices. Although this paper focuses on the device's use in low Earth orbit (LEO), the device may also be able to measure spacecraft charge at higher altitudes, in the solar wind, and in orbits around other planets. The flight prototype SCM we are producing for delivery to NASA in the third quarter of 2004 will measure floating potential from 0 to -150 volts with 0.1 volt precision, weigh approximately 600-700 grams, consume approximately 2 watts, and will measure approximately 8 x 10 x 17cm.

Spacecraft Charge Sensing

The phenomenon of spacecraft charging is of concern to those who maintain platforms in space. Spacecraft charge collection can take place in a variety of ways, including photo ionization of spacecraft surfaces by solar radiation and by interactions between the spacecraft and charged particles in space (free electrons and ions – space plasma).

There is a distinction made between "absolute" and "differential" spacecraft charging. Absolute spacecraft charging is the charging of the spacecraft chassis relative to the surrounding space plasma. Differential spacecraft charging is charging between electrically isolated parts of a spacecraft. Large differential charging is thought to be a cause of spacecraft system failures or, in extreme cases, entire spacecraft failures. Differential charging is generally thought to be more of a hazard to spacecraft than absolute charging. Spacecraft designers try to avoid differential charging by providing electrical contact between parts. Sometimes this is done with an electrically conductive coating. Although steps can be taken to limit differential charging on spacecraft, absolute charging is less readily avoided. Since the SCM is designed to measure

absolute spacecraft charge, all of the discussion of "spacecraft charge" or "spacecraft floating potential" that follows will be in reference to the charge of the spacecraft chassis relative to the space plasma.

Since a spacecraft chassis is not normally "grounded" to the surrounding space plasma, it is free to collect charge and reach a floating potential relative to the space plasma. Although the absolute charge on a spacecraft should theoretically be very small (a few volts) under "normal" conditions, geomagnetic-activity-induced and spacecraft-power-system-induced charging can charge spacecraft to tens, hundreds, or even thousands of volts. Although geomagnetic-activity-induced charging might be rare (or mild) for some orbits, spacecraft-power-system-induced charging of nearly the voltage of the power system (e.g., about a hundred volts for the International Space Station) may take place every time a satellite's solar arrays are illuminated.

Spacecraft designers are concerned about absolute charge collection because it can affect the operation of the spacecraft. Even relatively mild (less than 100 volts) absolute spacecraft charge can disrupt the operation of electrically biased instruments, bias the data collected by scientific instruments, attract contaminants to sensitive instrument surfaces, and ultimately cause arc discharges and sputtering that deteriorate spacecraft surfaces or even cause electronics components to fail.

Charge mitigation schemes have been proposed to avoid the problems associated with absolute spacecraft charging. For example, the International Space Station is expected to be prone to significant charging due to its high-voltage (160-volt) DC power system and its orbit in a plasma-rich region of space. Designers predict that constant arc discharges will deteriorate surfaces of the station unless charge collection is mitigated. Therefore, the Plasma Contactor Unit has been designed to harmlessly dissipate charge from the International Space Station.

We believe efforts to study the spacecraft-charging phenomenon and efforts to successfully mitigate spacecraft charging will be advanced by the instrument we are now developing. Although the hazards of spacecraft charging are well known, the actual measurement of spacecraft floating potential has proven difficult. In cases where it is thought spacecraft systems have failed due to problems associated with spacecraft charging, the conclusion has usually been arrived at by very careful deduction rather than by direct measurements of spacecraft charge at the time of failure. Methods now used to measure spacecraft potential depend on relatively bulky, expensive instrumentation and/or complicated data analysis.

The instruments listed in the following table are used to measure spacecraft potential in low Earth orbit. The table lists the instrument, the method used, and some of the factors that limit the effectiveness of the method. Our comparison of the various methods available today is not made in an attempt to discount the careful work that has been done in the past by specialists in the field of spacecraft charging. By all accounts it has proved difficult to measure spacecraft charge by any method. In fact, the closest we have found to a published comparison of multiple methods of spacecraft floating potential measurement used on a single flight is the following: "the various methods [used to measure spacecraft charge] yielded very different results" [*Myers et al.*, 1990]. Clear statements of accuracy may be rare because it is difficult, if not impossible, to estimate the accuracy of current techniques (there are many factors that must be considered to make such

estimates). It may also be difficult to find statements of accuracy simply because there has not been a demand for reliable spacecraft potential measurements until recently.

Methods Now Used to Measure Spacecraft Potential

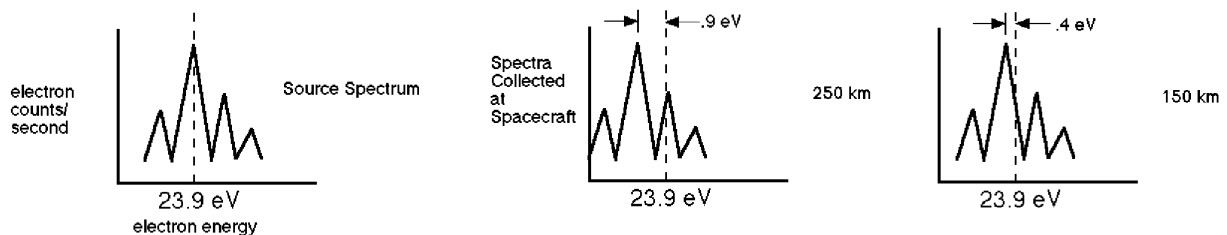
<u>Instrument</u>	<u>Method</u>	<u>Major Limitations</u>
Double Probe	Spacecraft's electric field measured by potential difference between two probes mounted on booms.	Biases due to changes in probe work function, probe photoemission, etc. Booms needed. [Maynard, 1998]
Langmuir Probe	Volt-ampere characteristic of probe immersed in space plasma is measured.	Biases due to changes in probe work function, magnetically induced probe potentials, and so on. [Brace, 1998]
Retarding Potential Analyzer (RPA)	A current-voltage curve from the instrument is analyzed to determine ion drift velocity.	Biases due to uncertainty in expected ion drift for spacecraft at 0 potential. [Anderson, 1994]
Ion Energy Analyzers	Ion spectra of space plasma are analyzed for low energy cutoff.	Biases due to uncertainty in the low energy cutoff from such measurements. [Moore, 1996]

Methods now used to measure spacecraft (chassis) floating potential

Electron-Spectroscopic Charge Sensing

While examining high energy-resolution electron spectra for another purpose, an unexpected shift in the energy location of sharp electron spectroscopic peaks was observed. What was an annoyance in that study (the peak shift had to be compensated for) led the way to a novel method for determining absolute spacecraft floating potential. The data examined were from the Photoelectron Spectrometer Experiment (PES) of Atmosphere Explorer E (AE-E) [Doering, Bostrom, and Armstrong, 1973].

The peak shift, illustrated in the following diagram, was due to the ~0.5 volt change in spacecraft chassis floating potential as the satellite changed in altitude from 250 to 150 km.



Electron-spectroscopic peak shift due to change in satellite floating potential.

The electron-spectroscopic technique of charge sensing is possible because 22.2, 23.9, 25.2, and 27.2 eV electrons are produced by the photoionization of oxygen and nitrogen at 150-300

km altitude by 304Å solar radiation. Simply put, the energy location of the four photoionization peaks (source spectrum) is well defined; therefore, an apparent shift in their energy location is due to deceleration or acceleration of the source electrons by the spacecraft's floating potential. Although the phenomenon was first observed due to a 0.5-volt change in spacecraft floating potential, the phenomenon can be used to determine floating potential of any magnitude.

Spacecraft floating potential can be determined from any electron spectra that contain the characteristic peaks that are produced by the 304Å photoionization of the atmosphere. Such spectra can be collected from all locations in the sunlit atmosphere at low altitudes (from 150-250 km, the "production region" for photoelectrons). In the production region photoelectrons are produced and lost locally and the distribution is isotropic. However, not many spacecraft fly at 150-250 km altitude. Fortunately, the electron spectroscopic method can be extended to higher altitudes, as will be described below.

An upper boundary of 250 km has arbitrarily been chosen for the photoelectron production region. It can extend to 300 km if the density of neutrals (mostly atomic oxygen at those altitudes) is great enough. Above 250-300 km the density of neutrals is so low that the photoelectrons observed at those altitudes are almost exclusively escaped photoelectrons from the top of the production region (from 250-300 depending on atmospheric density). The photoelectrons travel from the region of production to higher altitudes along geomagnetic field lines. Fluxes of photoelectrons to higher altitudes are great enough that spectra that contain the characteristic peaks due to 304Å photoionization of the atmosphere can readily be obtained with the SCM. If the photoelectrons pass through a region of low thermal plasma density, the spectra will be little changed from spectra collected at lower altitudes [Lee *et al.*, 1980b], and it will be possible to make very accurate spacecraft potential determinations at that location. Spectra collected by PES exhibit the 304Å photoionization peaks at all altitudes for which data is available (PES collected no data above 1000 km). However, there is evidence that the electron-spectroscopic method could be used to determine spacecraft floating potential at very high altitudes. Research indicates that conjugate photoelectron spectra (spectra of photoelectrons that have traveled along geomagnetic field lines to the nighttime ionosphere) might contain the peaks needed to determine floating potential. Studies of satellite data show that conjugate photoelectrons, even after travelling great distances along field lines, do not change significantly in energy (they had no discernable change in energy within the limits of the experiment) [Peterson *et al.*, 1977a and b].

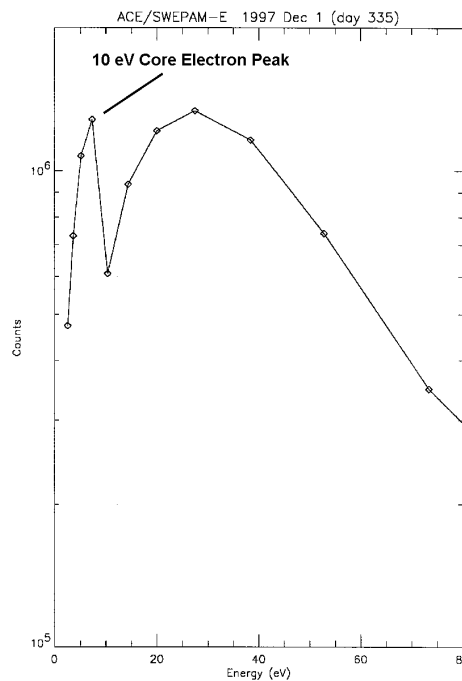
Data from PES of AE-E indicate that it should be possible to make spacecraft potential determinations that are accurate to 0.1 eV at local times from 6 to 9 at altitudes to 1000 km without the use of correction factors, since spectra from those local times will resemble those collected in the production region. Thermal plasma densities are too low at early local times to degrade the spectral peaks through coulomb scattering. At other local times, where the photoelectrons from the production region may need to pass through a region of high thermal plasma density before detection by the SCM, the photoelectron peaks will be broadened and somewhat lowered in energy due to coulomb scattering. It may be possible to use a correction factor in the analysis of spectra that exhibit peak broadening (and the coinciding lowering in peak energy) in order to preserve the 0.1 volt accuracy of the electron spectroscopic method.

Such a factor could be derived from electron transport theory (e.g., as exemplified in the work of Jaspers and Smith [1978] and Link [1992]).

A noteworthy advantage of the electron-spectroscopic method of charge sensing is the promise of an accurate measure of spacecraft charge with few effects that will reduce the reliability of the measurement over time. A predominant method now used (i.e., the use of Langmuir probes) suffers from calibration drift due to a change in probe surface work function with exposure to atomic oxygen. Since the electron-spectroscopic method is based on the determination of the energy location of electron-spectral peaks, drift in the floating potential measurement over time can only be due to a drift in the energy calibration of the electron spectrometer. With careful electronics design, such drift for even a space flight electron spectrometer can be made negligible.

Although this paper focuses on the use of the SCM in low Earth orbit (where all of the PES data was gathered), it may be possible to extend the technique not only to higher orbits (as was just described) but to orbits around other planets as well. The same technique that the SCM will use to determine floating potential above the Earth's atmosphere could be used to determine floating potential above other planets since the dominant 304Å ionizing solar radiation will produce peaks in photoelectron spectra from other atmospheres.

A somewhat different phenomenon will enable the SCM to determine spacecraft charge in the solar wind. There is a distinct peak in electron abundance in the solar wind at about 10 eV. By measuring how much the 10 eV electrons have been accelerated or decelerated by the spacecraft, the floating potential can be measured. In fact, the method is already used, as will be described in detail below.



Energy spectrum of solar wind electrons from ACE. Courtesy of Ruth Skoug.

The spectrum shown above is typical of the data now used to determine spacecraft floating potential in the solar wind. It is data from the SWEPAM instrument on the ACE spacecraft. Curve fitting is used to determine where the 10 eV "solar core electron" peak appears. Note that the peak of interest, at about 10 eV, consists entirely of 5 data points, over a range of about 10 eV. The SCM, with twenty-fold the geometric factor of the SWEPAM instrument at 10 eV, would have much greater sensitivity to the signal used to determine floating potential. Given the same amount of time, the SCM would be able to collect much more data in the critical 0-20 eV range. The quality of the data the SCM would collect would enable a more accurate and/or higher frequency determination of the spacecraft floating potential. The same performance characteristics that enable the SCM to measure floating potential from atmospheric photoelectrons (i.e., energy resolution and data gathering rate) will allow the SCM to excel at measuring floating potential in the solar wind. However, at this time our efforts are primarily focused on flying the SCM in low Earth orbit, where the need for the device is immediate and chances of obtaining a timely first flight are better.

The Spacecraft Charge Monitor

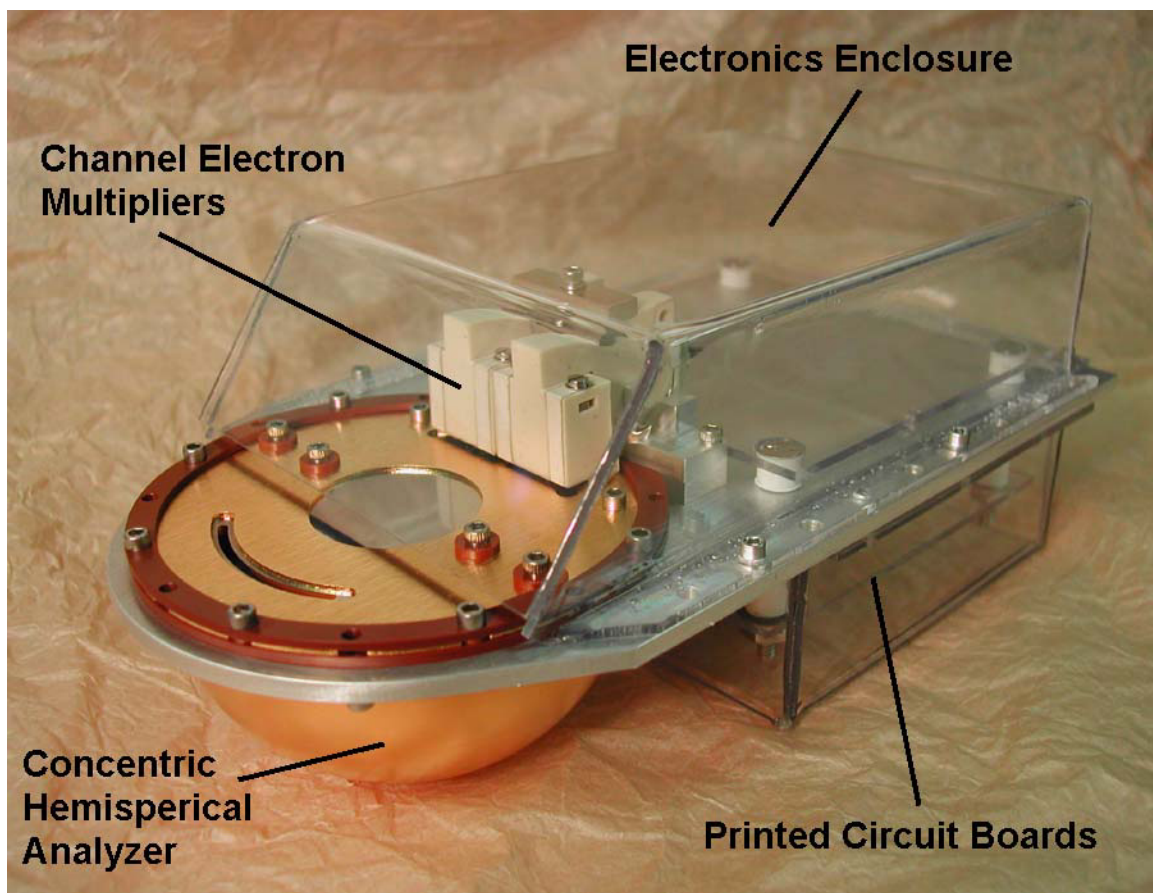
Although the electron spectroscopic method of spacecraft charge sensing shows great promise, its feasibility has only been demonstrated through theory and a single, relatively small data set - the photoelectron spectra from PES. Although the PES instrument gave us the first clues to the method, PES collected data that is difficult to interpret for our purposes (more about this appears in Goembel and Doering [1998]). This is understandable because the method was not published until 22 years after the last PES instrument was launched. The PES instrument was designed to make a broad survey of photoelectrons over a large energy range (0-500 eV). The next step in learning how to best exploit the method is to collect electron spectra from various orbits with an instrument designed specifically for electron-spectroscopic charge sensing.

To that end, Goembel Instruments has been developing a device that will use high energy-resolution electron spectroscopy to determine spacecraft charge. To the best of our knowledge, PES has been the only satellite instrument able to collect data with sufficient energy resolution in the energy range needed to determine spacecraft charge. Its energy resolution (2.5% $\Delta E/E$) was unusually high: most charged-particle spectrometers flown have energy resolution that is far worse (typically 10%, 15% or even 30%). Those instruments would not be able to produce spectra that exhibit the well-defined peaks needed to determine spacecraft charge. The primary reason only low-energy-resolution charged-particle spectrometers are now flown is that there is a trade-off between instrument resolution and data gathering rate for instruments of traditional design. If one wanted to improve the data gathering rate of a traditional concentric-hemispherical-analyzer (e.g., PES), one could simply increase the size of the device. Doubling the size (diameter) of the PES concentric-hemispherical-analyzer, and scaling up all other dimensions (such as aperture diameter) would increase the data-gathering rate of PES 8-fold while retaining its 2.5% energy resolution. Of course, doubling the size of the concentric-hemispherical-analyzer would add a great deal to the bulk of the device, an action to be avoided in space flight instrument design.

The SCM contains a charged-particle spectrometer of distinctly non-traditional design. In fact, the SCM includes a unique, patent-pending collimator/aperture arrangement that will for the

first time allow the collection of high energy-resolution charged-particle spectra at a high rate with a small diameter concentric-hemispherical-analyzer. The unique collimator/aperture arrangement will enable the SCM to collect the data needed to determine spacecraft floating potential in about one-sixtieth the time it would take PES to collect the same data. Whereas PES weighed 3,160 grams [Doering, Bostrom, and Armstrong, 1973], the flight prototype SCM will weigh approximately 600-700 grams. The flight prototype SCM will have a data-gathering rate per gram two orders of magnitude greater than that of PES even though it will have the same extraordinary energy resolution of PES.

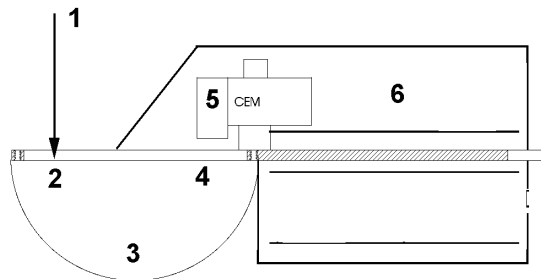
The following photograph is of the flight prototype SCM in its current state, 13 months into its 24-month development through Phase II SBIR funding. All that remains to complete the device is the fabrication of the electronics and electronics housing, which are represented by clear plastic in the photograph.



The Goembel Instruments Spacecraft Charge Monitor (incomplete assembly).

The function of the SCM is now described with reference to the following numbered diagram. The SCM consists of a concentric-hemispherical electrostatic charged-particle energy analyzer, two large aperture channel electron multipliers, instrument electronics, and supporting structure. The concentric-hemispherical-analyzer contains three electrostatic surfaces: the inner hemisphere, the outer hemisphere, and the aperture/collimator. Electrons (1) enter the SCM

through the $\sim 60^\circ \times \sim 20^\circ$ fan-shaped field-of-view of the entrance aperture/collimator (2) and enter the hemispherical shell-shaped space (3) defined by the inner and outer hemispheres. Electrons with kinetic energy in the narrow range allowed to pass through the exit aperture (4) strike the CEMs (5) and are "multiplied" into detectable pulses at the output anode of the CEM. The SCM electronics (6) control the electrostatic surfaces of the concentric-hemispherical analyzer, process the pulses from the electron multipliers, receive commands from the spacecraft, and output data to the spacecraft. The pulses are counted at each energy and, thus, an electron energy spectrum is produced. The SCMs field-potential-gate-array processor (FPGA) will receive commands from the spacecraft to set the data-gathering mode of the device and will output data to the spacecraft. All of the device's electronics will reside on three printed circuit boards within a 1.5 mm-walled aluminum enclosure. The complete flight prototype SCM fits within a volume of 8 x 10 x 17cm. The device requires approximately 2 watts (28 volts, unregulated) to power, will communicate with the spacecraft through a standard RS 422 interface, and is designed to operate over a -24°C to $+61^\circ\text{C}$ temperature range.



Functional diagram of the SCM

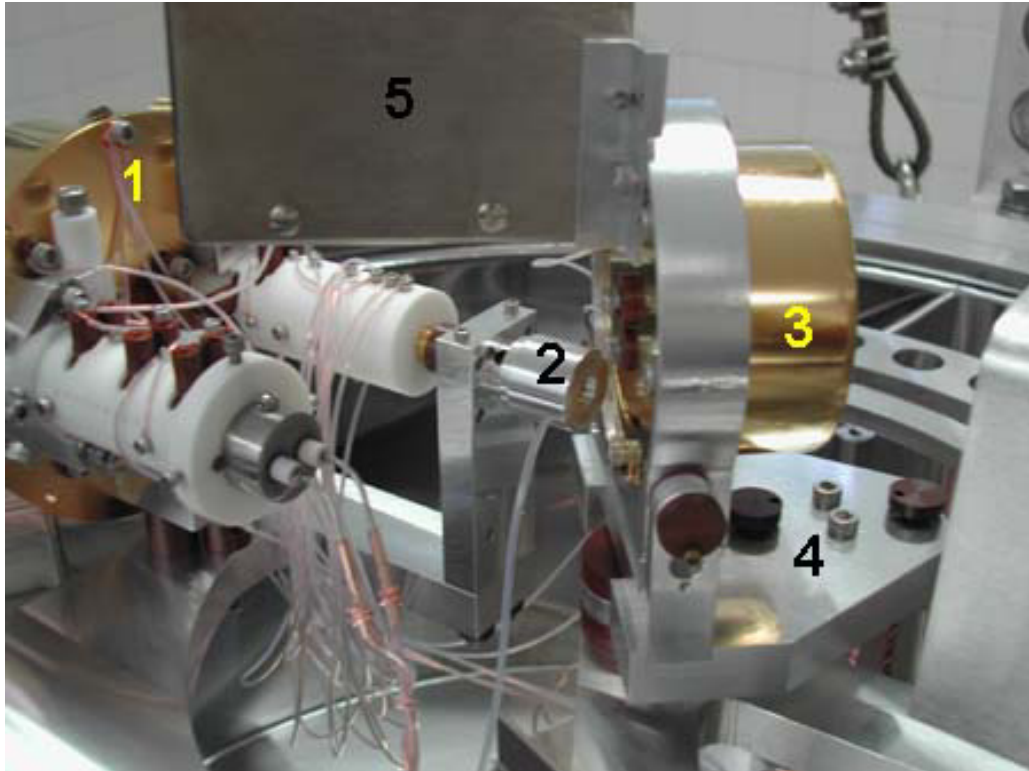
Laboratory Tests of the SCM

The Goembel Instruments electron spectroscopy laboratory is equipped with a high vacuum chamber, precision electron monochromator, and all of the apparatus needed to performance test the SCM. A hybrid turbomolecular/molecular drag pump and an oil-free rough pump evacuate the vacuum chamber. Custom-built electronics control the electron monochromator. The vacuum chamber and electron monochromator control electronics are shown in the following photograph.



SCM test and calibration apparatus

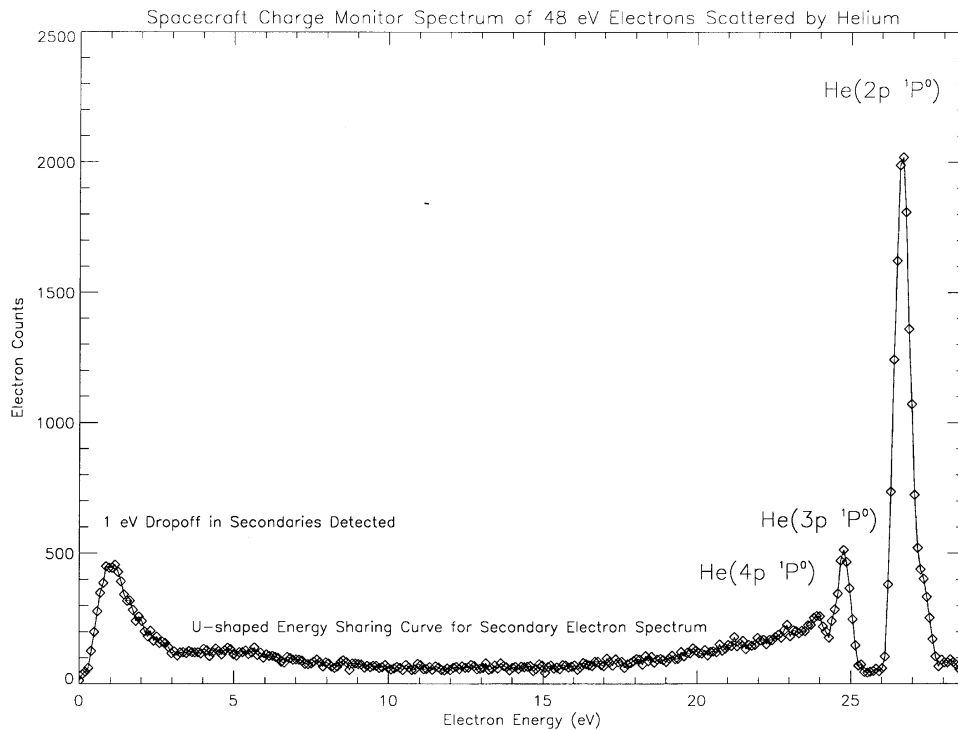
The following photograph illustrates the arrangement of the electron source (electron monochromator) and laboratory prototype SCM in the vacuum chamber (top section of chamber removed).



Photograph of the electron monochromator and laboratory prototype SCM

The numbers that appear in the photograph will aid the reader in the description. Keep in mind that the laboratory prototype SCM concentric-hemispherical-analyzer (3) has the identical electron-optical design of the flight SCM and is being used in laboratory tests of the SCM until the completed flight prototype electronics are delivered. The electron monochromator (1) produces a well-collimated beam of electrons with an energy spread of less than 0.1 eV. The electron monochromator is described in detail in Goembel and Doering [1995]. The electron beam passes through a gas cell (2), where gas is admitted so that electron scattering spectra of the gas can be gathered by the SCM. The laboratory prototype SCM concentric hemispherical analyzer (3) is held in an angular-controlled mount (4). The mount includes an angular control so that we can measure the SCM field-of-view. An AMPTEK MD-501 electron detector module (5) is mounted at the exit aperture of the SCM hemispherical analyzer. The electron detector module (5) and the FPGA-based laboratory prototype control electronics (not shown) serve the same purpose as the integral CEMs/electronics of the flight prototype SCM.

The following spectrum was gathered with the laboratory prototype SCM.



SCM Electron Impact Spectrum of Helium

The spectrum covers an energy range of 0-30 eV in 0.1 eV steps. The electrons detected by the SCM are those that result from the scattering 48 eV source electrons (from the electron monochromator) by helium. The peaks that appear at energies greater than 23 eV are due to the electron impact excitation of helium to various electronic (quantum) states above the ground state. The electrons that have been detected by the SCM have lost some of their 48 eV of kinetic energy in collisions with the helium atoms. The final state of the electrically excited helium atoms produced is labeled above each of the peaks in the spectrum. The tallest peak in the spectrum (at about 27 eV) is the first in the Rydberg series. The Rydberg series converges to a continuum to the left of the peaks at ~23.4 eV: 48 eV minus the ionization potential of helium

(I.P._{He} = 24.58 eV). The table, below, gives the final state of the He produced, and the energy lost by the 48 eV electrons in the inelastic collision with the gas.

The ionization of helium to certain final electronic states produces the peaks in the spectra.

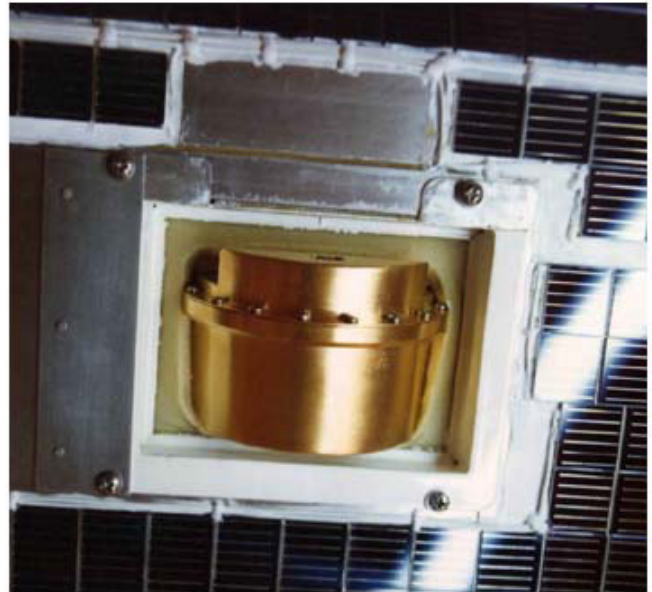
He⁺ Electronic State Produced by Collision	Energy Lost by Primary (48 eV) Electron
4p ¹ P ⁰	23.73 eV
3p ¹ P ⁰	23.08 eV
2p ¹ P ⁰	21.21 eV
2s ¹ S	20.61 eV
2s ³ S	19.81 eV

At energies less than 23 eV the spectrum exhibits the u-shaped “energy-sharing curve” expected for secondary electron production through the collision of the 48eV electrons with a gas. The sudden reduction in intensity that appears at 0-1 eV in the spectrum is also expected. It is due to the inability of concentric-hemispherical-analyzers configured as this one is to pass electrons with less than ~1 eV energy. The spectrum collected with the SCM laboratory prototype confirms that the patent-pending design of the SCM retains the 2.5% energy resolution of the PES device in precisely the energy range needed to determine spacecraft floating potential. This and other spectra collected from the SCM demonstrate that the SCM has the high energy resolution and excellent internally-produced-secondary-electron (noise) suppression needed to collect spectra for the determination of spacecraft floating potential.

Tests of the complete flight prototype SCM are planned to commence after the delivery and integration of the flight electronics. Instrument field-of-view, energy calibration, temperature stability, etc. will be determined. Vibration and thermal vacuum tests will also be performed.

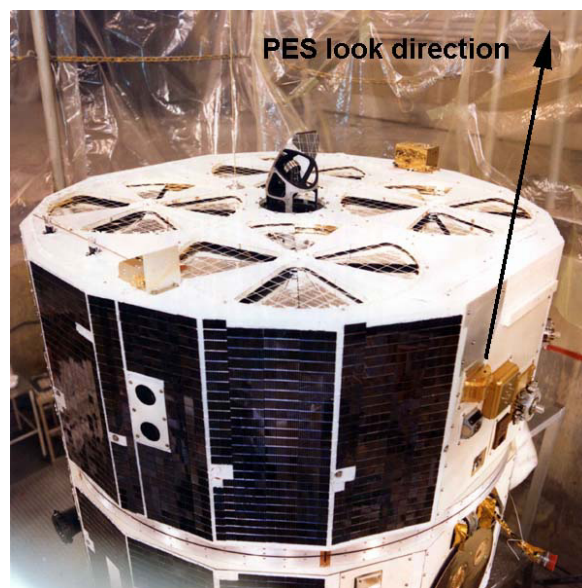
Integration of the SCM onto a Spacecraft

Experience gained from the deployment of the six PES instruments on three Atmosphere Explorers (AE-C, -D, and -E) will guide us in the deployment of the SCM. As the following photographs illustrate, the PES instruments were mounted on the AE satellite so that the entrance aperture/collimator of the device protruded slightly from the satellite's outer surface.



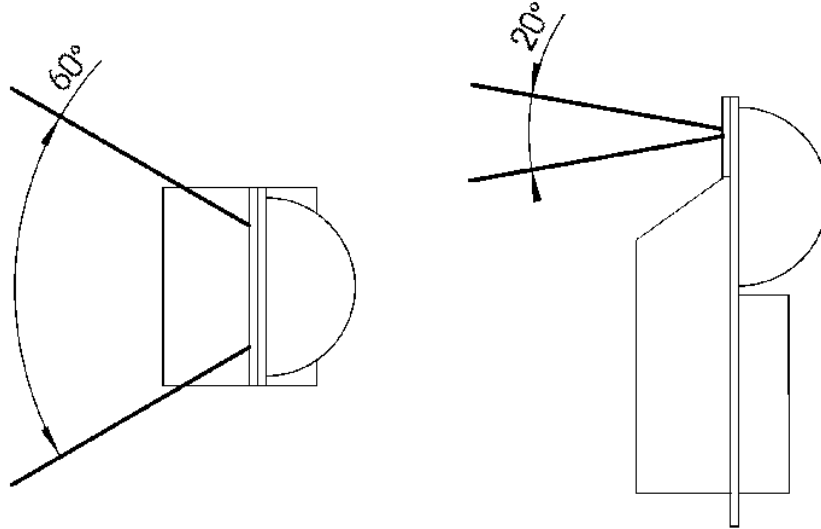
PES sensor 1 (left) and sensor 2 (right) mounted on AE-E, circa 1976.

It is desirable to orient the look direction of the SCM so that it points to within 55° of geomagnetic field lines that pass through the photoelectron production region. The " 55° " guideline results from the experience gained from the orientation of PES sensor 1 of the AE-E satellite. PES sensor 1 was mounted on the ram side of the despun spacecraft looking along the spin axis (AE-E orbited at an inclination of $\sim 20^\circ$). Interested readers can learn more about how to best mount the SCM on a spacecraft by referring to Goembel and Doering [1998] and Appendix 1 and 2 of Lee et al., [1980a]. One of the great advantages offered by the electron spectroscopic method of charge sensing is that regardless of the vagaries of the orientation of the sensor, if a signal is received (peaks in the photoelectron spectrum), the floating potential can be determined. Misalignments from an ideal orientation will not bias the measurement.



The look direction of PES sensor 1 on AE-E, circa 1976.

Whereas PES has a field of view of approximately $20^\circ \times 9^\circ$, the SCM has a field of view of approximately $60^\circ \times 20^\circ$, as illustrated by the following diagram.



$60^\circ \times 20^\circ$ field of view of the SCM

The field of view is actually a more complex shape than that shown due to the patent-pending electron optical design of the SCM, but for spacecraft integration purposes the $60^\circ \times 20^\circ$ approximation suffices. The larger field of view contributes to the greater data-gathering rate (larger geometric factor) of the SCM. It also allows more flexibility to mount the device in an orientation that maximizes the portion of an orbit that photoelectron spectra can be gathered.

To recap what was said in section in 4 of this paper: the flight prototype SCM will measure floating potential from 0 to -150 volts with 0.1-volt precision, weigh approximately 600-700 grams, require approximately 2 watts (28 volts, unregulated) to power, and have the dimensions of approximately 8 x 10 x 17cm. The SCM will communicate with the spacecraft through a standard RS 422 interface, and is designed to operate over a -24°C to $+61^\circ\text{C}$ temperature range. The flight prototype we are developing will be delivered to NASA in the third quarter of 2004. We are looking forward to retrieving data from the flight of the prototype and are already considering options for subsequent versions of the SCM.

Acknowledgements

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