N78-10143

Contents

12. The Multiple Applications of Electrons in Space

Réjean J. L. Grard
Space Science Department
European Space Agency
Noordwijk, The Netherlands

Abstract

An electron source such as a simple cathode is a cheap and light device which can serve several technological and scientific purposes in space:

(i) Electrostatic charging of a spacecraft can be limited by releasing electrons accumulated on the conductive elements of their surface. Clamping the reference potential of scientific instruments, such as particle detectors, can significantly improve their performance in a magnetospheric environment.

(ii) The erosion of conductive coatings and the ability of conductive paints to withstand the space environment can be evaluated by monitoring the flow of charged particles impinging on their surface, that is, by simply measuring the rate at which electrons are emitted from the cathode.

(iii) Measuring the current collected by the spacecraft surface as a function of its potential with respect to an emitter is a very sensitive diagnostic technique which can yield a number of plasma parameters, such as density and temperature.

(iv) It is possible to convert the thermal motion of space plasmas into electrical energy by collecting energetic electrons and returning them to the medium as cold particles. This concept may find applications in the magnetosphere of distant planets where solar cells are inefficient.

(v) A wave in a plasma is characterized by a conduction current density which gives rise to fluctuations of the current flowing to the surface. An investigation of the frequency spectrum of the cathode current will therefore disclose the existence of electromagnetic and electrostatic waves without using any antenna.

1. INTRODUCTION

The potential of a body in space is defined by the current balance of the charged particles emitted and collected by its surface. In equilibrium the difference between the flows of plasma electrons and ions equals the rate at which electrons extracted from the surface by photo- and secondary emissions are escaping into the surrounding plasma.

When the random flux of the plasma electrons is relatively large, the surface develops a potential sufficiently negative to limit the incoming flow of these particles and to make it equal to the combined contribution of the other species. The magnitude of this potential is then in direct proportion to the temperature of the ambient electrons. Such conditions are often met in dense or hot plasmas when surface emission cannot match the net flow of ambient particles. In planetary outer magnetospheres, the electron mean kinetic energy is so high that surface potentials of the order of several kilovolts are frequently encountered. This phenomenon, commonly referred to as spacecraft charging, disturbs the particle population in the surrounding plasma and is therefore a source of interferences for scientific measurements.[1]

It has also been observed that different materials insulated from each other do not reach the same floating potential. This differential charging gives rise to large electric fields between adjacent elements and can cause discharges which are responsible for the degradation of spacecraft materials and anomalies in the behaviour of electronic subsystems.[2, 3]

This situation has led up scientists and engineers to compound their effort in an attempt to understand and control these phenomena.[4] The basic remedies are simple. The entire surface of the spacecraft must be made conductive in order to be equipotential and the negative charge accumulated on this body must be released into space through an electron emitter.

A number of materials, conductive coatings, and paints have been developed and qualified, and new testing procedures have been set up to check the ability of the spacecraft to withstand the magnetospheric environment.

This paper recapitulates the principles which govern the interaction of a spacecraft with its environment,[5, 6] and reviews the various techniques which are available for controlling the electrostatic potential of a spacecraft. It is demonstrated that electron emitters are requisite to any scientific mission in a magnetospheric environment, in the vicinity of Jupiter in particular. It is also emphasized that electron sources can be simultaneously used for a number of additional tasks such as monitoring the degradation of spacecraft material surfaces, converting the thermal motion of plasmas into electrical energy, measuring the density and temperature of the ambient electrons, and receiving waves without aerials.

2. INTERACTION BETWEEN A SPACECRAFT AND ITS ENVIRONMENT

2.1 Spacecraft Without Emitter

The variations of the different current components collected by a planar probe in space as function of its potential are schematically represented in Figure 1, where i_e and i_i are the ambient electron and ion contributions, and i_{ph} is the current due to photoelectron emission. Secondary emission is neglected in first approximation and the potential Φ is referred to that of infinity.

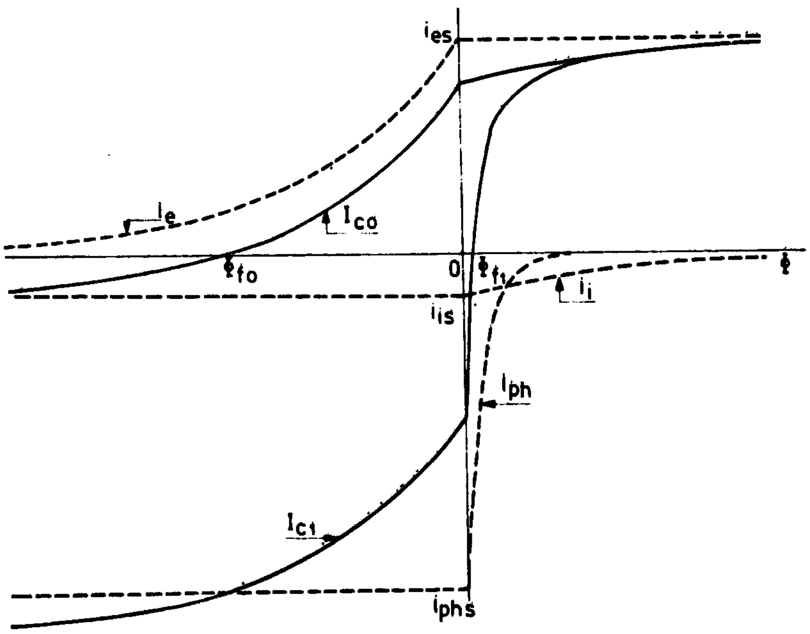

Figure 1. Current-Voltage Characteristics of a Probe in a Plasma

It is assumed that the various species have Maxwellian distributions, and that the magnitude of the saturation current of the plasma electrons i_{es} is much larger than that of the ions i_{is}; the photoelectron saturation current is noted i_{phs}.

The current balance is defined by

$$i_e + i_i + i_{ph} = 0 \quad , \tag{1}$$

and the floating potential is given by

$$\Phi_f = -\Phi_e \ln \left| \frac{i_{es}}{i_{phs} + i_{is}} \right| \tag{2}$$

when $|i_{es}| > |i_{phs} + i_{is}|$, and by

$$\Phi_f \simeq \Phi_{ph} \ln \left| \frac{i_{phs} + i_{is}}{i_{es}} \right| \tag{3}$$

when $|i_{es}| < |i_{phs} + i_s|$. The quantities Φ_e and Φ_{ph} are the mean kinetic potentials of the plasma electrons and photoelectrons, respectively.*

The current-voltage characteristic of a body in shadow ($i_{ph} = 0$) is illustrated by the curve labelled I_{C0} in Figure 1. The floating potential Φ_{f0} is negative and approximately equal to $-3.8\Phi_e$ in the case of an hydrogen plasma in thermal equilibrium.[7] In a magnetospheric environment, Φ_e can be of the order of 1-10 kV, which explains why a geostationary spacecraft develops large negative potentials during eclipses.[8, 9, 10] The same situation also occurs in sunlight when photoemission cannot balance the flow of ambient particles. This condition is occasionally fulfilled in the Earth environment, but it must always be met in the magnetosphere of Jupiter where the photoemission rate is 27 times less than at the Earth's orbit.

In a relatively cold and rarefied plasma, such as the solar wind, the photoemission saturation current is generally predominant; this situation is illustrated by the curve labelled I_{C1} in Figure 1. The corresponding floating potential Φ_{f1} is given by Eq. (3) and is of the order of the photoelectron mean kinetic potential, which is typically equal to 1.5V.[11]

2.2 Spacecraft With Electron Emitter

A spacecraft fitted with an electron source is schematically represented in Figure 2a. The electron source and the conductive elements of the spacecraft surface are referred to as the emitter (E) and the collector (C), respectively. It is assumed that the emitter and collector are sufficiently decoupled, so that their

*The kinetic potential of a charged particle is given by the magnitude of the accelerating voltage associated with its kinetic energy. The kinetic potential, in V, therefore, is measured by the same number as the kinetic energy, in eV.

current voltage characteristics are, in first approximation, independent of their separation.

The voltage-current characteristic of the emitter is schematically represented in Figure 2b and c by the curves labelled I_E; the voltage reference is that of the surrounding plasma, not that of the spacecraft. The area of the emitter is relatively small and its current is, therefore, insensitive to the fluxes of photons and ambient particles. When electron emission is space charge limited, I_E is proportional to $(-\Phi)^{3/2}$. The shape of the characteristic is otherwise defined by the temperature of the emitter, as well as by the magnitude of the electric field at its surface.[12] It is assumed that the saturation current of the emitter is larger than that of the ambient electrons, which can always be easily fulfilled.

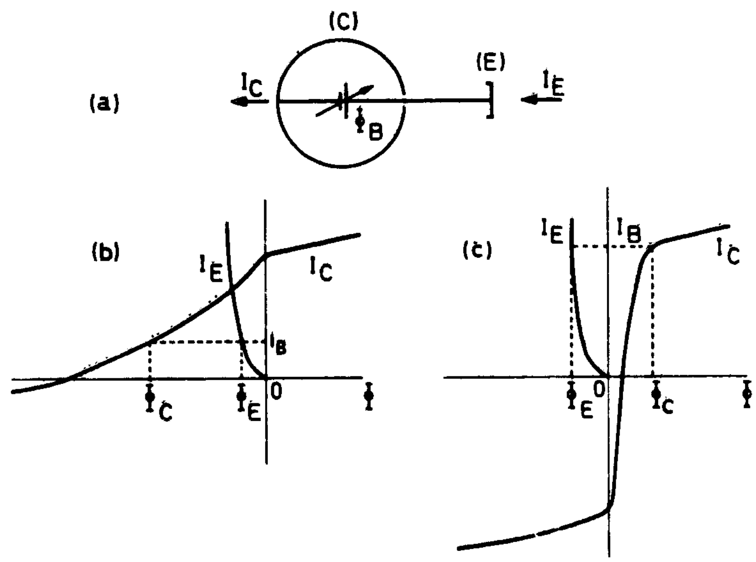

Figure 2. Current Balance and Potentials of Emitter and Collector

When collector and emitter are connected through a voltage source Φ_B, their respective potential Φ_C and Φ_E are linked by the relation

$$\Phi_C - \Phi_E = \Phi_B \quad , \tag{4}$$

and the emitted and collected currents are of course equal,

$$I_E = I_C = I_B \quad , \tag{5}$$

as illustrated in Figure 2.

3. POSSIBLE USES OF ELECTRON EMITTERS

3.1 Spacecraft Potential Clamping

When the bias voltage Φ_B is zero, the equilibrium potential of the collector - emitter combination is defined by the intersection of the curves I_C and I_E. In a magnetospheric environment the floating potential can then be maintained at a few volts, rather than several kV negative, as illustrated in Figure 2b.[13, 14, 15] The potential of the collector can even be adjusted to zero exactly by biasing its potential positively with respect to that of the emitter until a break in the slope of the characteristic is observed.[16] The ability to control the spacecraft potential allows one to minimize the perturbation to the environment and provides a stable voltage reference for scientific instruments.

An electron emitter cannot indeed reduce the floating potential when photoemission is predominant (Figure 2c), but this is a relatively unimportant point since this positive potential is typically of the order of 3V.[17]

3.2 Plasma Diagnostics

It will be seen in the following that the potential of the emitter is practically independent of its current; this system operates like a potential reference with respect to which the collector may be biased. The emitted current may be measured as function of the bias potential. The spacecraft then behaves like a Langmuir probe[18] with a collecting area equal to that of the conductive parts of its surface.[19, 20] This technique for measuring electron density and temperature is extremely sensitive, and irregularities in plasma densities can be detected by investigating the low frequency fluctuations of the collected current.

The electron saturation current can be monitored under all circumstances, and this measurement is absolutely not impaired by photoemission (see Figure 2c). A variable resistor R can replace the voltage source, as shown in Figure 3, but the bias voltage $-R\,I_B$ can only take negative values. This possibility is nevertheless extremely

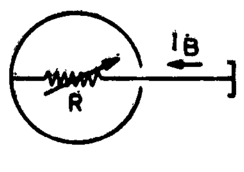

Figure 3. Negative Bias of the Collector Using a Resistor

advantageous in energetic plasmas where voltages of several kV would be required to describe the characteristic down to the floating potential.

3.3 Monitoring of Material Surface Degradation

Spacecraft are partly covered with materials and paints, generally insulators, in order to maintain the temperature within specified limits. Conductive paints and coatings have consequently been developed to cope with the problem of differential charging. Solar cell covers and second surface mirrors can similarly be covered with 100Å ...ck layers of Indium oxide which also insures electric potential uniformity, but are optically transparent.[21]

The ability of these conductive materials to keep their properties in space under the bombardment of energetic particles can be easily tested in situ, as illustrated in Figure 4. The current collected by the test material, I_t, is simply compared to that impinging on a metallic surface, I_r, taken as a reference. This measurement is instantaneous and independent of the properties of the cathode.

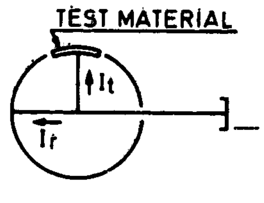

Figure 4. Test on Material Surface Degradation

3.4 Energy Conversion

It has been shown previously that energy can be dissipated in a resistor when the floating potential is negative (Figure 3). It is therefore possible to convert the thermal motion of electrons into electrical power.[22] This concept may find applications in space environments where energetic plasmas are likely to be found, but sufficiently distant from the sun to render the use of photovoltaic conversion unpractical. Such conditions are met in the magnetospheres of distant planets such as Jupiter and Saturn.[23, 24]

The efficiency of this system, that is, the ratio of the plasma energy input to the available electric energy equals 0.37 for a Maxwellian energy distribution. The corresponding maximum power output per m^2 of collecting surface is given as function of the ambient plasma density, N_e, and mean kinetic potential, Φ_e, in Figure 5. If the collecting area of the spacecraft is insufficient, the power output can be increased by the adjunction of a large sail made of metallic foil, for example.

It may be possible to reach specific power equivalent to that of the radioisotope thermal generators presently used for the outer planetary missions,[25, 26] that is, a few Wkg^{-1}, provided the plasma power density input lies in the range 10^{-2}-10^{-1} Wm^{-2}. This figure may be met in the environment of Jupiter, but is has to be established by proper in situ measurements.

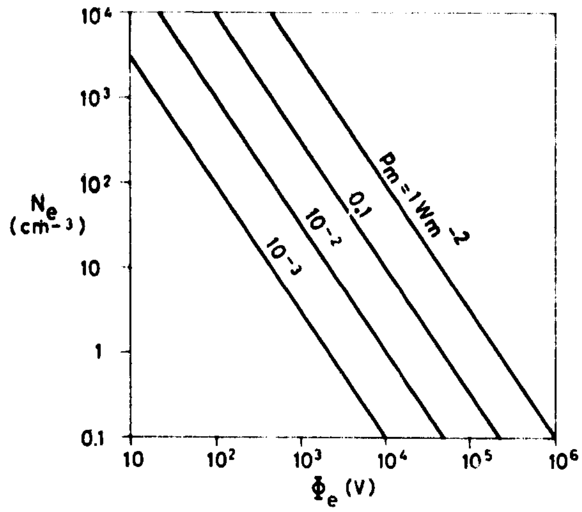

Figure 5. Maximum Power Density Output as a Function of the Electron Density and Mean Kinetic Potential

3.5 Detection of Electromagnetic and Electrostatic Waves

A wave propagated in a plasma is characterized by an electric field $\vec{E}$, a magnetic field $\vec{H}$, and a conduction current density $\vec{J}$. These three quantities are related by Maxwell's equations:

$$\nabla \times \vec{E} = -\mu_o \, \partial \vec{H}/\partial t \quad , \tag{6}$$

$$\nabla \times \vec{H} = \vec{J} + \varepsilon_o \, \partial \vec{E}/\partial t \quad , \tag{7}$$

where μ_o and ε_o are the vacuum permeability and permittivity. Time and space variations are assumed to be of the form $\exp i(\vec{k}\vec{r} - \omega t)$, where $\vec{k}$ and $\omega = 2\pi f$ are the wave vector and angular frequency, $\vec{r}$ and t are space and time variables.

After Fourier transformation of Eq. (6) and (7), and elimination of $\vec{H}$, the projection of $\vec{J}$ parallel and perpendicular to the wave vector are respectively

$$J_{\parallel} = i\varepsilon_o \, \omega \, E_{\parallel} \tag{8}$$

and

$$J_{\perp} = i\varepsilon_o \, \omega \, (1 - \mu^2) \, E_{\perp} \quad , \tag{9}$$

where $E_{\parallel}$ and $E_{\perp}$ are the projections of $\vec{E}$ along directions respectively parallel and perpendicular to $\vec{k}$, and μ is the refractive index of the medium.

The wave conduction current which can be intercepted is

$$I = A_c J \quad , \tag{10}$$

where J is the modulus of $\vec{J}$ and A_c is the cross section area of the collector in a plane perpendicular to $\vec{J}$. This current is superimposed on the random plasma ion and electron currents. The information carried by the ambient charged particles is consequently best detected when the spacecraft is collecting the saturation current of these species.[27, 28] The collector potential must therefore be maintained at a value near to zero, or possibly be biased at a few volts positive if photoemission is preponderant (Figure 2b and c). Neglecting the ion contribution this dc current is then approximately given by

$$i_{es} = A J_{\dot{e}} \quad , \tag{11}$$

where A is the entire collecting area, and

$$J_{\dot{e}} = N_e \dot{e} \left(\frac{e \Phi_e}{2 \pi m} \right)^{1/2} \tag{12}$$

is the electron random current density, in the case of a Maxwellian energy distribution. The quantities e and m are the electron charge and mass respectively.

The level of the smallest detectable signal is, of course, limited by the shot noise resulting from the random arrival of plasma particles to the spacecraft surface and the random emission of electrons from the cathode into the environment. The root mean square deviation of i_{es} in a frequency bandwidth B therefore defines the lowest measurable wave conduction current[29, 30, 31]

$$I = (2 e i_{es} B)^{1/2} \quad . \tag{13}$$

The capability of this technique can be assessed by comparing its performance to that of an electric aerial in a given bandwidth. This is simply achieved by equating Eq. (10) and (13), and expressing the sensitivity in terms of electric field E, rather than conduction current J.

Consider a wave with a transverse electric field ($E = E_{\perp}$), for example, an electromagnetic wave in an isotropic plasma or a longitudinal wave in a magnetoplasma. Combining Eq. (9)-(13) yields

$$\frac{E}{B^{1/2}} = \frac{4}{(2\pi)^{3/4}} \frac{(me)^{1/4}}{\varepsilon_o^{1/2}} \frac{\Phi_c^{1/4}}{\rho} \frac{f_p}{f} \frac{1}{|1-\mu^2|} \tag{14}$$

where the electron plasma frequency is defined

$$f_p = \left(\frac{N_e e^2}{2\pi \varepsilon_o m}\right)^{1/2} \tag{15}$$

and the collector is taken to be a sphere of radius ρ, that is, $A = 4\ A_c$.

Using the MKSA system of units and replacing various quantities by their numerical values yields

$$\frac{E}{B^{1/2}} = 2 \times 10^{-7} \frac{\Phi_e^{1/4}}{\rho} \frac{f_p}{f} \frac{1}{|1-\mu^2|} \quad , \tag{16}$$

The sensitivity to longitudinal electric fields and to electrostatic waves in general ($E = E_{\parallel}$) can be evaluated if Eq. (8) is used, the equivalent sensitivity then becomes

$$\frac{E}{B^{1/2}} = 2 \times 10^{-7} \frac{\Phi_e^{1/4}}{\rho} \frac{f_\rho}{f} \quad . \tag{17}$$

Under most conditions and provided the sensitivity and frequency response of the current measuring device permits, this concept allows the detection of waves with electric field spectral density of the order of 10^{-6}-10^{-8} V m^{-1} Hz$^{-1/2}$ up to frequencies equal to a few times that of the local plasma resonance. In this frequency range, these performances are certainly comparable to those of other types of antenna used in space.

4. EXPERIMENTAL ARRANGEMENT

4.1 Location of the Emitter

It is desirable to optimize the plasma diagnostic measurements by mounting the electron source at some distance from the collector. Coupling and mutual interactions are indeed the result of the direct interception by the spacecraft of a fraction of the emitted current. Trajectories of electrons emitted with zero velocity are illustrated in Figure 6. It can be seen that the role of the spacecraft is minimal even when its potential is more positive than that of the emitter.

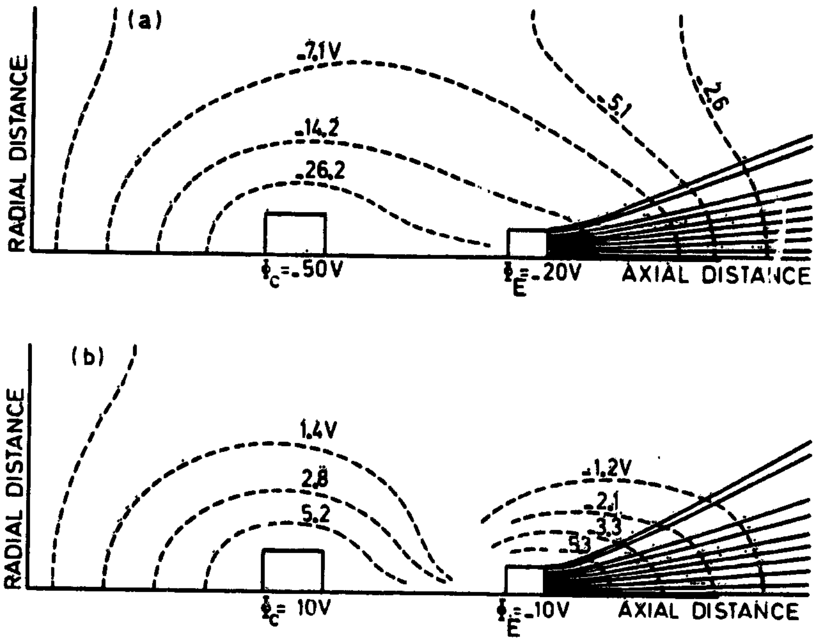

Figure 6. Electron Trajectories in an Axial Symmetrical Field

Potential clamping is more efficient if the electric properties of the electron source environment are as little as possible influenced by the proximity of the spacecraft. The potential developed by the collector in the vicinity of the emitter has so far been taken equal to zero, but this assumption is only valid at large distances from its surface. Approximating the collector by a sphere of radius ρ, the potential at a distance r from its center is given by

$$\Phi_r = \Phi_C \frac{\rho}{r} \quad . \tag{18}$$

The effect of probe separation is graphically demonstrated in Figure 7. When the potential referenc[illegible] e emitter characteristic is shifted by the amount given by Eq. (18), the colle[illegible] ential becomes

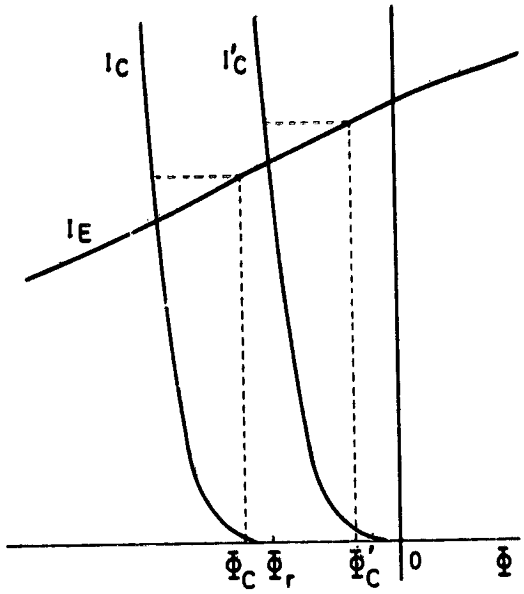

Figure 7. Effect of Collector-Emitter Separation

$$\Phi_C = \Phi'_C \frac{1 - dI_C/dI_E}{1 - dI_C/dI_E - \rho/r} \approx \frac{\Phi'_C}{1 - \rho/r} \tag{19}$$

where Φ'_C is the spacecraft potential for infinite separation. The quantity dI_C/dI_E which represents the ratio of the collector and emitter current variations for a given voltage increment is close to zero in energetic plasmas. The result given by Eq. (19) is indeed approximate since it does not account for direct electron flow between emitter and collector, but it shows the tendency for the spacecraft potential to increase with decreasing separation.

Potential clamping can be achieved without bias voltage. It can be anticipated, however, that limitations caused by space charge near the emitter and potential barriers resulting from differential charging, as observed on ATS 6[32] and possibly Pioneer 10,[33] are more important when the electron source is mounted too close to the surface (Figure 8a). Nevertheless, it has been demonstrated that spacecraft potentials could be controlled with an emitter mounted in an open cavity under the surface, but only for limited periods of time (Figure 8b). In fact, if electrons are emitted with zero velocity, their injection into the plasma is impossible unless the emitter is biased negatively with respect to the collector. This type of consideration naturally leads to the concept of electron guns with grid system (Figure 8c), such as those mounted on the ISEE-A spacecraft.[34, 35]

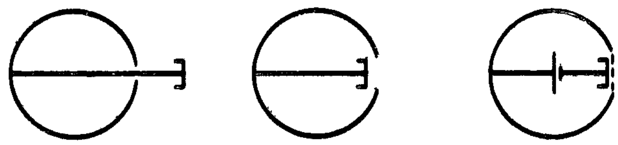

Figure 8. Three Emitter-Collector Configurations

4.2 Types of Emitter

4.2.1 ACTIVE EMITTERS

Electron emission from a metal in vacuum is a function of its temperature and of the electric field existing at its surface.

Thermoionic emission describes the situation where temperature is the most important parameter. A directly heated tungsten filament is the simplest type of emitter but it dissipates a power of several W. An indirectly heated cathode impregnated with a barium compound requires less than 1W of heating power and offers in addition a uniform surface potential (Figure 9a and b).

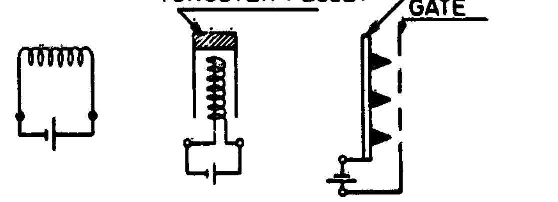

Figure 9. Three Types of Active Electron Emitter

Electron field emission occurs when an electric field of the order of 10^9 Vm^{-1} exists at the surface of a metal. Cathodes working on this principle, that is, without any heater, have recently been developed using thin film technology.[36] Electrons are extracted from an array of sharply pointed cones and a voltage of the order of 100 V is applied on a perforated electrode called gate, located at about 1 μm (Figure 9c). Controlling the energy of the emitted electrons, and thus the potential of the spacecraft, requires the existence of an additional grid placed in front of the gate and electrically biased with respect to the collector. The structure of this emitter then resembles very much that of a miniature electron gun with typical dimensions of 1 mm; energy is only required for accelerating the electrons.

The saturation current of these cathodes, of the order of several mA, is always larger than the plasma electron saturation current collected by the spacecraft. The saturation current density in a plasma characterized by $N_e = 1\ cm^{-3}$ and $\Phi_e = 1$ kV, for example, is less than $1\ \mu A\ m^{-2}$. Electron emission is therefore space charge limited like in a diode and the current is of the form

$$I_E = K\Phi_E^{3/2}/2 \quad . \tag{29}$$

If I_E is measured in A and Φ_E in V, $K = 3 \times 10^{-5}$ for spherical symmetry and α is function of the ratio r_a/r_c of the anode-to-cathode ratio.[37]

In first approximation, infinite collector-emitter separation and spherical symmetry are assumed; r_c is given by the physical dimension of the cathode and r_a is the distance over which space-charge neutrality is restored in the plasma, that is, a distance of the order of the Debye length.

The cathode current is represented by straight lines in Figure 10 for values of r_a/r_c ranging from 10 to 10^6. Also shown is the electron current collected by a conductive sphere of radius 2 m, in various plasma environments, photoelectron and ion currents are neglected. The clamping potential in absence of any biasing voltage is defined by the intersection of two of these curves; it is seen that this potential is not much influenced by the ratio r_a/r_c, and is of the order of 0.1-10V negative for electrons emitted with zero energy.

4.4.2 PASSIVE EMITTER

An entirely passive emitter can be simply made of sharp-pointed filaments, electrically connected to the spacecraft but positioned at a distance equal to a few times the typical dimension of the vehicle, as shown in the insert of Figure 11.[19] The separation is requisite since it ensures that the strength of the electric field at the tips is not reduced by the charge induced in the surface of the main body.[38]

It is assumed that the collector is a sphere, 1 m in radius, immersed in a Maxwellian plasma with $N_e = 1\ cm^{-3}$ and $\Phi_e = 1$ kV. The separation between the emitter and the surface is considered to be much larger than the sphere radius. In fact, for a distance of 3 radii, the clamping potential is within 25 percent of the value obtained for infinite separation. The emitter is made of 100 tips with a curvature radius $a = 0.1\ \mu m$; the emissive area is taken to be $2\pi a^2$ and the strength of the electric field at the tips is of the order of $|\Phi_E/a|$.

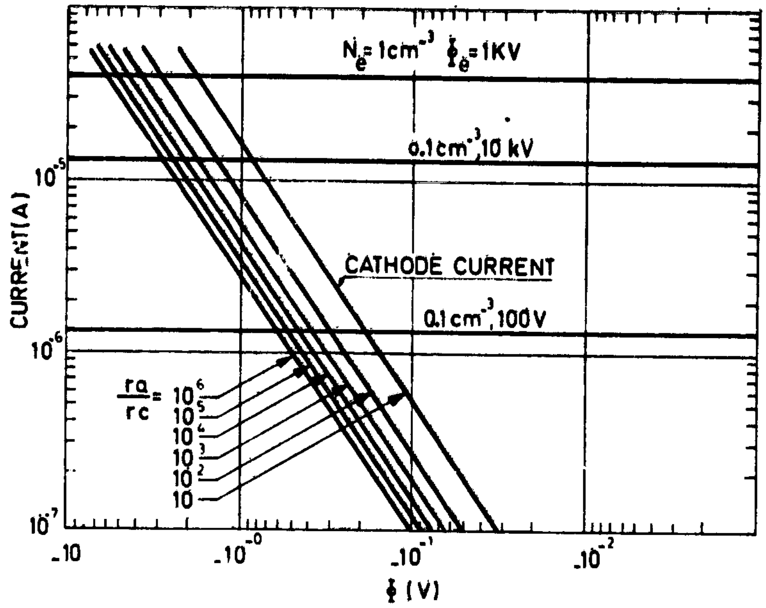

Figure 10. Determination of the Clamping Potential with an Active Electron Emitter

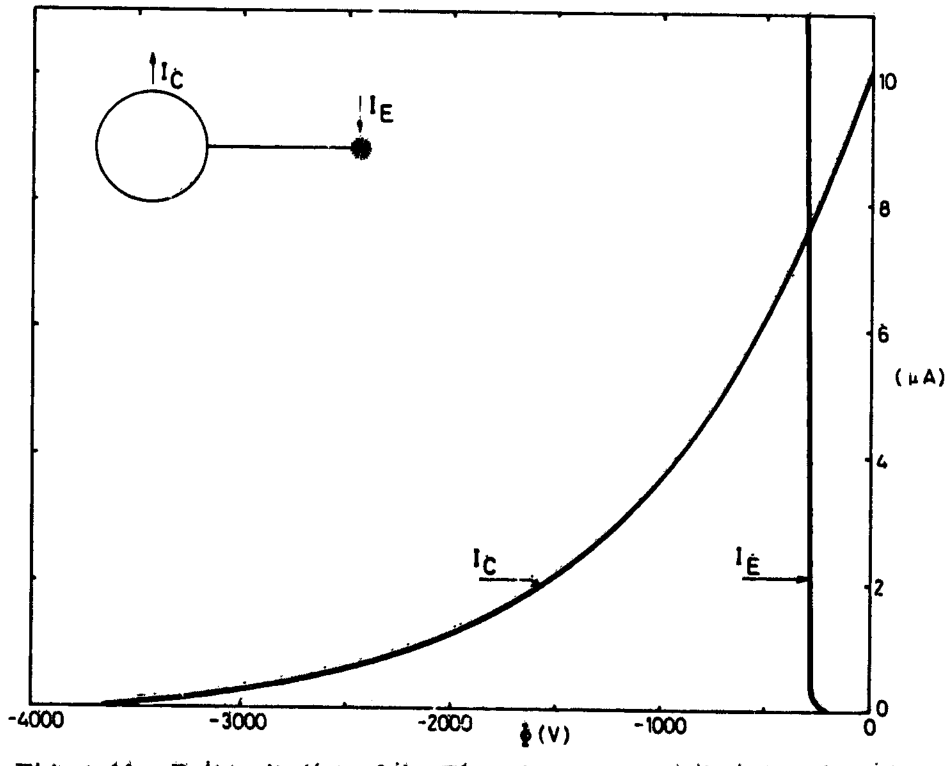

Figure 11. Determination of the Clamping Potential With Passive Electron Field Emitter

The current characteristic of the emitter is given by the Fowler-Nordheim equation,[39] and the potential of the sphere equal to -3600 V without emitter ($-3.8\Phi_e$) is increased to -316 V when connected to the emitter; potential closer to zero are certainly possible with sharper points. The current that can be emitted from such a probe is limited by thermal dissipation and is of the order of 3 mA.

5. CONCLUSION

Electron emitters can prevent a spacecraft from accumulating negative charges and clamp its potential close to zero, provided its surface is conductive. As such, they should be part of any scientific magnetospheric payload because they significantly increase the value of field and particle measurements, in particular.

Information on the ambient medium can be obtained at a very little extra cost; such a diagnostic technique is very sensitive in rarefied plasma and can disclose small scale irregularities in the electron density. The existence of electromagnetic and electrostatic waves can also be detected, without any antenna, by investigating the frequency spectrum of the emitted current, that is, observing the current fluctuations associated with the alternative motion of the ambient particles.

Electron emitters also have interesting technological applications; they can monitor the degradation of conductive paints and coating in space and transform the thermal agitation of a plasma into electrical energy.

The choice of a system for space applications is motivated by considerations on reliability, weight, and power consumption. A thin-film field emission needs no heating but requires a grid system for electron extraction and energy control. A passive electron emitter is very simple but must be mounted on a boom with a length larger than the typical dimension of the collector. Presently, indirectly heated dispenser cathodes seem to offer the best comprise between these various requirements. They have been extensively tested in the laboratory and have lifetimes longer than one year. Provided they are mounted on an appendage of moderate length, say 0.5 m, they can clamp a spacecraft at a potential between -10 and -1 V and perform most of their functions without any grid or polarization system.

Acknowledgments

The author wishes to thank the European Office of Aerospace Research and Development for arranging and supporting his participation at the Conference.

References

1. Grard, R.J.L., Knott, K., and Pedersen, A. (1973) The influence of photoelectron and secondary electron emission on electric field measurements in the magnetosphere and solar wind, in Photon and Particle Interactions with Surfaces in Space, R. Grard, Editor, p. 163, Reidel, Dordrecht, Holland.

2. Fredricks, R.W., and Scarf, F.L. (1973) Observations of spacecraft charging effects in energetic plasma regions, in Photon and Particle Interactions with Surfaces in Space, R. Grard, Editor, p. 277, Reidel, Dordrecht, Holland.

3. Nanevicz, J.E., Adamo, R.C., and Shaw, R.R. (1975) Electronic discharges caused by satellite charging at synchronous orbit altitudes, in the Proceedings of the 1975 Conference on Lightning and Static Electricity, Paper V2, The Royal Aeronautical Society, London.

4. McPherson, D.A., Cauffman, D.D., and Schober, W. (1975) Spacecraft charging at high altitude - the SCATHA satellite program, AIAA Paper 75-92, AIAA 13th Aerospace Sciences Meeting, Pasadena, Ca.

5. Grard, R.J.L. (1975) Effect of the ambient medium upon the electric properties of the spacecraft surface and environment, in the Proceedings of the 1975 Conference on Lightning and Static Electricity, Paper VI, The Royal Aeronautical Society, London.

6. Rosen, A. (1975) Spacecraft charging: environment induced anomalies, AIAA Paper 75-91, AIAA 13th Aerospace Sciences Meeting, Pasadena, Ca.

7. Self, S.A. (1963) Exact solution of the collisionless plasma sheath equation, Physics of Fluids 6:1762.

8. Knott, K. (1972) The equilibrium potential of a magnetospheric satellite in an eclipse situation, Planet Space Sci. 20:1137.

9. DeForest, S.E. (1972) Spacecraft charging at synchronous orbit, J. Geophys. Res. 77:651.

10. DeForest, S.E. (1973) Electrostatic potentials developed by ATS. 5, in Photon and Particle Interactions with Surfaces in Space, R. Grard, Editor, p. 263, Reidel, Dordrecht, Holland.

11. Grard, R.J.L. (1973) Properties of the satellite photoelectron sheath derived from photoemission laboratory measurements, J. Geophys. Res. 78:2885.

12. Murphy, E.L., and Good Jr., R.H. (1956) Thermoionic emission, field emission and the transition region, Phys. Rev. 102:1464.

13. Bartlett, R.O., DeForest, S.E., and Goldstein, R. (1975) Spacecraft charging control demonstration at geosynchronous altitude, AIAA Paper 75-359, New Orleans, La.

14. Goldstein, R., and DeForest, S.E. (1976) Active control of spacecraft potentials at geosynchronous orbit, in Spacecraft Charging by Magnetospheric Plasmas, A. Rosen, Editor, AIAA Progress in Astronautics and Aeronautics Series, Vol. 47, p. 169.

15. Grard, R.J.L., Gonfalone, A., and Pedersen, A. (1976) Spacecraft potential control with electron emitters, in Spacecraft Charging by Magnetospheric Plasmas, A. Rosen, Editor, AIAA Progress in Astronautics and Aeronautics Series, Vol. 47, p. 159.

16. Polychronopoulos, B., and Goodall, C.V. (1973) A system for measuring and controlling the surface potential of rockets flown in the ionosphere, in Photon and Particle Interactions with Surfaces in Space, R. Grard, Editor, p. 309, Reidel, Dordrecht, Holland.

17. Grard, R.J.L., and Tunaley, J.K.E. (1971) Photoelectron sheath near a plasma probe in interplanetary space, J. Geophys. Res. 76:2498.

18. Chen, F.F. (1965) Electric probes, in Plasma Diagnostic Techniques, R.H. Huddlestone and S.L. Leonard, Editor, Academic Press, New York.

19. Grard, R.J.L. (1975) Spacecraft potential control and plasma diagnostic using electron field emission probes, Space Science Instrumentation, 1:363.

20. Grard, R.J.L. (1976) Spacecraft charging control by field emission, J. Geophys. Res. 81:1805.

21. Köstlin, H., and Atzei, A. (1973) Present state of the art in conductive coating technology, in Photon and Particle Interaction with Surfaces in Space, R. Grard, Editor, p. 333, Reidel, Dordrecht, Holland.

22. Grard, R.J.L. (1977) The magnetospheric plasma as a source of energy for space instrumentation, Planet. Space Sci., in press.

23. Scarf, F.L. (1975) The magnetospheres of Jupiter and Saturn, in The Magnetosphere of the Earth and Jupiter, V. Formisano, Editor, p. 433, Reidel, Dordrecht, Holland.

24. Baker, D.N., and Van Allen, J.A. (1976) Energetic electrons in the Jovian magnetosphere, J. Geophys. Res. 81:617.

25. Brittain, W.M., and Christenbury, S.T. (1974) SNAP 19 Viking RTG flight configuration and integration testing, in 9th Intersociety Energy Conversion Engineering Conference Proceedings, p. 185, The American Society of Mechanical Engineers, New York.

26. Russo, F.A. (1974) Operational testing of the high performance thermoelectric generator (HPG-02), in 9th Intersociety Energy Conference Proceedings, p. 193, The American Society of Mechanical Engineers, New York.

27. Grard, R.J.L. (1976) Plasma diagnostic and wave detection with electron emitters, in the Proceedings of the Symposium on European Programmes on Sounding Rocket and Balloon Research in the Auroral Zone, SP 115, p. 221, European Space Agency, Noordwijk, Holland.

28. Grard, R.J.L. (1976) A new concept for detecting electromagnetic and electrostatic waves in space plasmas, Planet. Space Sci. 24:1097.

29. Bell, D.A. (1960) Electrical Noise, Van Nostrand, London.

30. Tunaley, J.K.E. (1970) The effect of noise on probe measurements in the magnetosphere, Ann. Geophys. 26:853.

31. Petit, M. (1975) Bruit radioélectrique dû aux photoélectrons, A.Télec. 30:351.

32. Whipple Jr., E.C. (1976) Observation of photoelectrons and secondary electrons reflected from a potential barrier in the vicinity of ATS 6, J. Geophys. Res. 81:715.

33. Intriligator, D.S. (1975) Pioneer 10 observations of the Jovian magnetosphere: plasma electron results, in The Magnetosphere of the Earth and Jupiter, V. Formisano, Editor, p. 313, Reidel, Dordrecht, Holland.

34. Mozer, F.S., Fahleson, U.V., Fälthammar, C.G., Kelley, M.C., Knott, K., and Pedersen, A. (1972) A Proposal to Measure Quasistatic Electric Fields on the Mother/Daughter Satellites, UCBSSL 454, University of California, Berkeley.

35. Gonfalone, A. (1976) An Electron Gun for Spacecraft Potential Control, EWP 1016, European Space Agency, Noordwijk, Holland.

36. Spindt, C.A., Brodie, I., Humphrey, L., and Westerberg, E.R. (1976) Physical properties of thin-film emission cathodes, J. Appl. Phys. 47:5248

37. Spangenberg, K. (1948) Vacuum Tubes, McGraw-Hill, New York.

38. Jeans, Sir James (1963) The Mathematical Theory of Electricity and Magnetism, p. 196, Cambridge University Press, New York.

39. Gomer, R. (1961) Field Emission and Field Ionisation, Harvard University Press.