NASA TECHNICAL MEMORANDUM

NASA TM X-73601

NASA TM X-73601

(NASA-TM-X-73601) DEVELOPMENT OF ENVIRONMENTAL CHARGING EFFECT MONITORS FOR OPERATIONAL SATELLITES (NASA) 7 p HC A02/MF A01 CSCL 14E G3/19

N77-17141

Unclas 14920

DEVELOPMENT OF ENVIRONMENTAL CHARGING EFFECT MONITORS FOR OPERATIONAL SATELLITES

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TECHNICAL PAPER presented at the
Spacecraft Charging Technology Conference cosponsored by the
United States Air Force and National Aeronautics and Space Administration
Colorado Springs, Colorado, October 27-29, 1976

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E-8970-5

ABSTRACT

An instrumentation package to monitor the effects of the environmental charging of spacecraft surfaces on the systems of operational spacecraft is being developed at the Lewis Research Center of NASA. This package is to perform two functions: first, the local charged particle flux and the particle characteristic energy will be monitored, and second, transients in the spacecraft electrical harness will be counted as a function of amplitude, with time. This package is considered to be a monitor of the spacecraft system. It will be used to determine the duration and effect of any environmental charging of the spacecraft surfaces. Thus, it will be possible to determine the relationship between the occurrence of any anomalies and the charging phenomenon. Design details and design goals of this package are presented.

1.0 BACKGROUND

Spacecraft charging occurs when spacecraft surfaces react to the charged particles of the geomagnetic substorm environment and charge to negative potentials relative to the space plasma. Such charging occurs predominately when the spacecraft is in the midnight-to-down portion of its orbit. Spacecraft surfaces in sunlight can charge from several tens of volts negative to several kilovolts negative. Spacecraft surfaces in the shade can charge to several tens of kilovolts negative; actual surface potentials depend upon the substorm intensity. Spacecraft surface potentials of this order have been deduced from scientific measurements made by instruments aboard geosynchronous spacecraft.[1,2,3]

The occurrence of anomalies on geosynchronous spacecraft is consistent with charging of spacecraft surfaces in the midnight-to-down quadrant of the spacecraft orbit. Direct relationship of the occurrence of spacecraft system anomalies, primarily changes in the state of electronic logic, and the charging of spacecraft surfaces has been inferred from operations aboard geosynchronous spacecraft.[4] There is scant one-to-one correlation of spacecraft anomalies with the local environment in which the spacecraft finds itself.

Spacecraft surfaces are covered with a variety of materials and can be variously shaped. Differential charging on these surfaces, then, can result.[4,5]

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*Published in the proceedings of the Conference on Spacecraft Charging Technology, NASA TM X-73537, 1977.

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Differential charging can lead to electrical breakdown of dielectric surfaces. These discharges can couple into spacecraft harnessing and may cause interference with electronic circuits.[6]

To date, spacecraft that have experienced anomalies have not had sensors to detect geomagnetic substorm conditions. In addition, there are indications that differences in spacecraft configurations, specifically those differences between spin-stabilized spacecraft and three-axis-stabilized spacecraft, result in different reaction of the spacecraft to its immediate environment.[7,8]

To address the spacecraft charging phenomena more data is needed. Data is needed from many missions. Simultaneous information about the spacecraft internal housekeeping environment and the local geomagnetic substorm environment is required. This can be accomplished with simple instrumentation that is usable on the greatest number of spacecraft. It will then be possible to correlate transient events with environmental conditions. Such instrumentation is being developed at the Lewis Research Center. Design details and some design goals of this instrumentation follow.

2.0 DESCRIPTION OF ENVIRONMENTAL MONITORS

2.1 Design Philosophy

To correlate spacecraft discharge-induced transient events with environmental conditions it is considered sufficient to count transient events induced in the spacecraft housekeeping harness, to make a measurement of the characteristic energy of the incident charged particles, and to measure the average current density of the incident charged particles. These are reasonable engineering measurements to make. The environmental monitors are analogous to a spacecraft system power monitor, a diagnostic device. The information obtained with these monitors can be used to supplement data obtained from scientific instruments.

2.2 Transient Event Counter

The monitor to be used to sense discharge-induced transients in the harnesses of spacecraft is a transient event counter. This is a growth version of the Transient Event Counter (TEC) presently returning data from the Communications Technology Satellite (CTS). Its characteristics are given in figure 1. Four sensors are assembled with the spacecraft harness. These sensors are coaxial cables with one end unterminated and stripped for a length of 30 to 60 cm. Each sensor signal is amplitude discriminated to one of three levels set during the final stage of TEC assembly. Only transients over the preset signal strength, measured at the input to the discrimination circuitry, are counted. The counting circuitry incorporates a 10-microsecond delay after a discharge pulse is counted to avoid counting of line ringing as discrete transient events. The counting circuitry incorporates a ring counter to eliminate transients induced by noise in the spacecraft-to-ground communication link. The four continuous sensor measurements are output to telemetry on separate digital channels. The TEC physical characteristics are summarized in figure 1 and reflect both design goals and what has been achieved to date. The TEC should be located within the satellite interior because of the rf shielding the electrically grounded spacecraft structure and thermal insulation provides.

2.3 Characteristic Energy Sensor

The characteristic energy sensor, described in figure 2, consists of an electrically floating metal plate coupled to a voltage sensor. The voltage sensor is a capacitively-coupled electrostatic voltmeter that operates on a null-balance principle whereby the potential of the voltmeter sensor is brought to the potential of the metal plate by a power supply. This design provides a very accurate local voltage measurement and minimizes large voltage gradients at the measurement location. The sensing range, +50 V to -20 kV, is based on present knowledge of the environment.[9] The frequency response bandwidth, dc to 5 Hz, is based on characteristic charging times that have been observed in testing.[10] The physical characteristics shown in figure 2 reflect both design goals and what has been achieved to date. The output is analog but analog-to-digital conversion can be performed, with the attendant increases in power and weight. It is necessary to locate this sensor on the satellite exterior, preferably not in the sun.

2.4 Current Density Sensor

The current density sensor is described in figure 3. It consists of a plain metal current collecting plate and current measurement circuitry. Current measurement is by means of an electrometer. The sensor range, 0.01 to 5 nA/cm^2, is based on present knowledge of the environment.[9] The frequency response bandwidth is dc to 1 Hz. The physical characteristics given in figure 3 are a combination of goals and what has been achieved to date. The output is analog; conversion to digital output can be performed, with the attendant weight and power increases. Sensor location is required to be on the spacecraft exterior, preferably not in the sun.

3.0 DISCUSSION

Utilization of the spacecraft charging monitors, briefly described herein, on the greatest possible number of operational spacecraft will serve to broaden the base of engineering data on the spacecraft charging phenomenon. Their use will also enable a heretofore unobtained one-to-one correlation of transient events with environmental activity.

The spacecraft charging monitors can serve as a warning system. The occurrence of a substorm electron injection into the local environment can be sensed within some fraction of a minute. The response of the dielectric surfaces of a spacecraft to the charging environment would be on the order of minutes.[10] Thus, the differential charging that can lead to breakdown could be dealt with by means of some active charge control or action could be taken to put the spacecraft in some fail-safe or standby mode of operation until the danger had passed.

The spacecraft charging monitors can serve as a diagnostic tool. Any anomalous behavior can be isolated as charging-induced or the result of other causes. The real-time capabilities the monitors can provide in this regard should be particularly beneficial to operational spacecraft.

4.0 CONCLUDING REMARKS

The Lewis Research Center is developing a package of engineering instruments to serve as spacecraft charging monitors. They sense the local spacecraft environment (the characteristic energy and flux of charged particles) and transients induced into spacecraft harnessing (as a result of breakdowns resulting from differential charging). The monitors are simple and conservatively designed. Their use can serve to warn of adverse changes in the local spacecraft environment. The data they can return will broaden the data base upon which spacecraft system design and test criteria specifications rest.

It is in the interest of all those involved in the use of operational spacecraft to integrate spacecraft charging monitors into their spacecraft systems. The state of knowledge of the spacecraft charging phenomenon is such that the warning and diagnostic functions that they would perform would serve to insure a successful mission.

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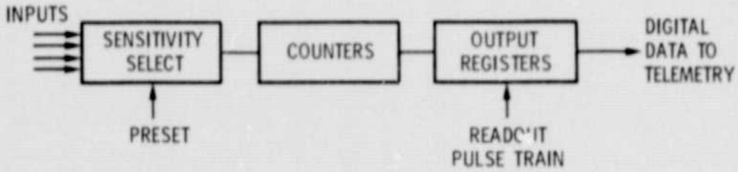

CHARACTERISTICS

4 SENSORS

AMPLITUDE DISCRIMINATION, 3 LEVELS, PRESET

PHYSICAL: APPROX $1\frac{1}{2}$ W, 0.5 KG, 100 CM^2 FOOTPRINT

OUTPUT: 4 DIGITAL TELEMETRY CHANNELS

LOCATION: SATELLITE INTERIOR

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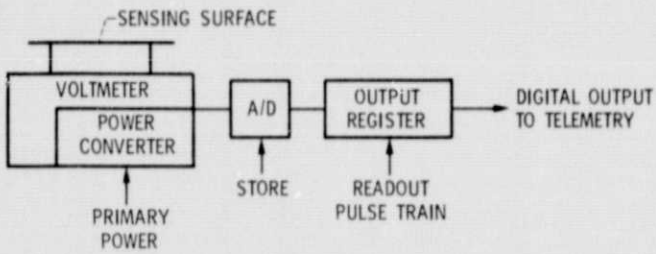
Figure 1. - Summary description of the transient event counter.



CHARACTERISTICS
SENSOR RANGE: +50 V TO -20 kV
BANDWIDTH: DC TO 5 Hz
PHYSICAL: APPROX 3/4 W, 0.5 KG, 100 CM^2 FOOTPRINT
OUTPUT: 8 BIT SERIAL TTL COMPATIBLE
LOCATION: SATELLITE EXTERIOR, NOT IN SUN

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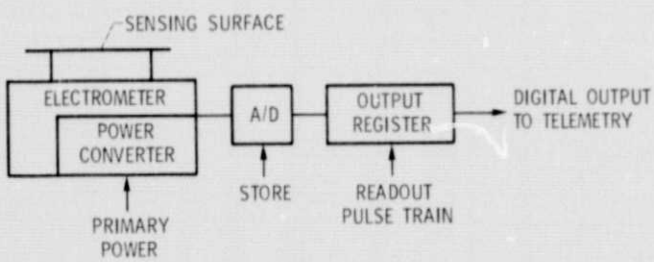
Figure 2. - Summary description of the characteristic energy sensor.



CHARACTERISTICS
SENSOR RANGE: 0.01 TO 5 nA/CM^2 CURRENT FLUX
BANDWIDTH: DC TO 1 Hz
PHYSICAL: APPROX 1 W, 0.5 KG, 80 CM^2 FOOTPRINT
OUTPUT: 8 BIT SERIAL TTL COMPATIBLE
LOCATION: SATELLITE EXTERIOR, NOT IN SUN

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Figure 3. - Summary description of the current density sensor.

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