and ± 50 volts. A hot-wire probe was swept through the beam at each bias and zero bias to measure the beam plasma potential and the results are shown in Fig. 2. The beam probe was only in the ion beam approximately $\pm 20°$ about the thruster centerline which coincided with the midpoint of the probe sweep. The balance of the sweep measured the plasma potential in the fringe or wing area of the beam plasma.

The data shown on Fig. 2 agreed with the results of Ref. 6. The SERT II spacecraft tended to float at 0 to 20 volts below space potential with no thruster or neutralizer on. With a thruster or neutralizer on the spacecraft could be maintained near zero potential or biased negatively. Positive bias of the spacecraft was ineffective because the neutralizer emission current was preferentially drawn to the spacecraft rather than space plasma.

Summary

The SERT II spacecraft, designed for 1 year life, remains functional after 5 years in space. Opportunity exists therefore to check the long-term operational status of the on-board ion thruster components, power processors, and other spacecraft ancillary equipment. During the 1974 test opportunity reported in this Note, a notable result was that the high-voltage short was clear on thruster 2 and that normal thruster operation was restored within the limits of available solar power. The cathodes and propellant supply system continued to function normally on thruster 1. Both power processors continued to function without fault after 5 years in space and 3889 and 2175 operating hours, respectively.

In addition to the thruster tests, a neutralizer cathode was operated separately to demonstrate that the electric potential level of a spacecraft could be controlled by the neutralizer. Orbital mechanics predict a continuous sun-lighted orbit in late 1980. If spacecraft reorientation maneuvers are performed, it could be possible to operate thruster 2 continuously in a 1981 test opportunity with the propellant remaining in the thruster reservoirs.

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Role of Thermal Contact Resistance in Pyrotechnic Ignition

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SPACECRAFT have been relying on small charges of pyrotechnic inside squibs to do useful work in space. Such work includes separating spacecraft from their launch vehicle, unfolding solar panels, and releasing communications antennas. A squib usually consists of a threaded metal body that holds a small amount of pyrotechnic in one end and has electrical leads sealed in glass running through the middle to the other end. A small diameter wire, called a bridgewire, is welded across the two electrical leads, and is imbedded in the pyrotechnic. An electrical current supplied from a charged capacitor, or some other source, heats up the bridgewire which in turn heats up the pyrotechnic until it ignites. This Note will examine how a thermal contact resistant can influence the time to ignition and how it can influence the instantaneous temperature distributions within the bridgewire-pyrotechnic system.

It has been shown by Sernas[1] that axial conduction in the bridgewire is negligible for short heating times that were of interest in this study. When it is assumed that bridgewire is completely surrounded with pyrotechnic, the heat transfer can be treated as a one-dimensional radial heat conduction problem in the pyrotechnic and bridgewire domains. A detailed discussion of this assumption can be found in Refs. 1 and 2.

The governing equation in the bridgewire was then

$$\rho_1 c_1 \frac{\partial T}{\partial t} = H + k_1 \left[\frac{1}{r} \frac{\partial T}{\partial r} + \frac{\partial^2 T}{\partial r^2} \right] \tag{1}$$

where ρ is the density, c is the specific heat, k is the thermal conductivity, T is the temperature, and t is the radial coordinate. The subscript 1 is used to denote the thermal properties of the bridgewire. The quantity H is the rate of volumetric heating in the bridgewire due to a constant electric current being applied to the bridgewire at time zero.

In the pyrotechnic the governing equation was taken to be

$$\rho_2 c_2 \frac{\partial T}{\partial t} = k_2 \left[\frac{1}{r} \frac{\partial T}{\partial r} + \frac{\partial^2 T}{\partial r^2} \right] \tag{2}$$

where the subscript 2 denotes the properties of the pyrotechnic. A term representing the self-heating of the pyrotechnic due to an exothermic chemical reaction has been left out to simplify the model.

The boundary condition used at the interface between the bridgewire and the pyrotechnic was

$$k_1 \left[\frac{\partial T}{\partial r} \right]_1 = h(T_1 - T_2) = k_2 \left[\frac{\partial T}{\partial r} \right]_2$$

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Index categories: Heat Conduction; Spacecraft Ground Testing and Simulation (including Components).

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Table 1 Thermal properties

	Tophet A	Pyro.
Density, g/cm	8.4	1.95
Specific heat, cal/gm-K	0.107	0.20
Thermal conductivity, W/cm-K	0.134	0.0088

where T_I and T_{20} represent the interface temperature of the bridgwire and the pyrotechnic respectively. This stipulates that a constant thermal conductance, h, exists between the bridgewire and the pyrotechnic. The remaining two boundary conditions were $\partial T/\partial r = 0$ at $r=0$, and $T = T_{\text{amb}}$ as r approaches infinity.

Equations (1) and (2) with their described boundary conditions were solved numerically with a transient heat conduction computer library program for a typical pie-shaped section of a squib which used a 50 μm (0.002 in) diam Tophet A bridgewire to ignite a boron, potassium perchlorate, Viton mixture. The bridgewire was broken up into two nodes of equal mass, and the pyrotechnic into seventeen nodes of varying mass. The twentieth node at $r = 155$ μm was a heat sink kept at the ambient temperature. Node 3 was the pyrotechnic node that was in contact with the bridgewire, and it is this node that becomes the hottest and reaches ignition temperature first. The thickness of node 3 was taken as 2.5 μm (10^{-4} in.) after it was shown that a smaller grid size did not change the pyrotechnic surface temperature.

The computer program could produce a complete temperature history of each node when the following information was specified: a) the ambient temperature; b) the thermal properties of the wire; c) the thermal properties of the pyro; d) the bridgewire heating rate, H; and e) the thermal contact conductance, h.

The values for the thermal properties used in this study are shown in Table 1. The properties were assumed constant and independent of temperature. The ambient temperature and the heating rate were made to fit the problem at hand.

The only information that was not available was the thermal contact conductance because it has never been measured experimentally nor estimated analytically. It has been usually assumed that the temperature of the bridgewire is the same as the temperature of the pyrotechnic in contact with it. This assumption is of course equivalent to assuming h equal to infinity. But it is well known that all solids in contact with each other have some finite value for their thermal contact conductance. Thus, it was necessary to estimate the magnitude of h.

An estimate for h was obtained by trial and error by guessing at a value for h and letting the computer program predict the temperature at node 3 at some fixed time which corresponded to the experimentally measured ignition time corresponding to the current level used in the program. The h for which the program predicted the ignition temperature at node 3 at the ignition time was then taken as the correct estimate for h. The critical ignition temperature for this pyrotechnic was estimated to be 672 K (750°F). Twenty squib firings at room temperature were conducted at a 3.5 amp level and it was found that ignition occurred at an average of 1.59 ms after the beginning of the pulse. Thus an h was found by trial and error that made the computer program predict a temperature of 672 K in node 3 after 1.6 ms of the constant 3.5 amp input into the bridgewire. The only value for h that satisfies these conditions (when used with the previous values for the thermal properties) was found to be 31,200 W/m^2-K (5,500 BTU/hr-ft^2-F). A few of the trials for h are shown in Fig. 1. It is clear from Fig. 1 that an h of infinity would yield a much higher temperature for node 3 at 1.6 ms, and as a result the predicted time to reach an ignition temperature of 672 K would be much shorter than 1.6 ms. It is interesting to note that the value of 31,200 W/m^2 for h is considerably higher than the values reported[3] for metal-to-metal contact of machined surfaces.

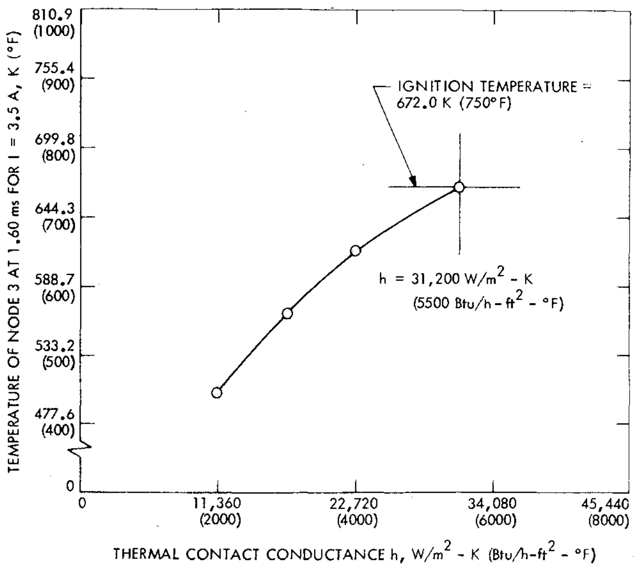

Fig. 1 Predicted node 3 temp at 1.6 ms after the beginning of a 3.5 amp pulse as a function of an assumed thermal contact conductance.

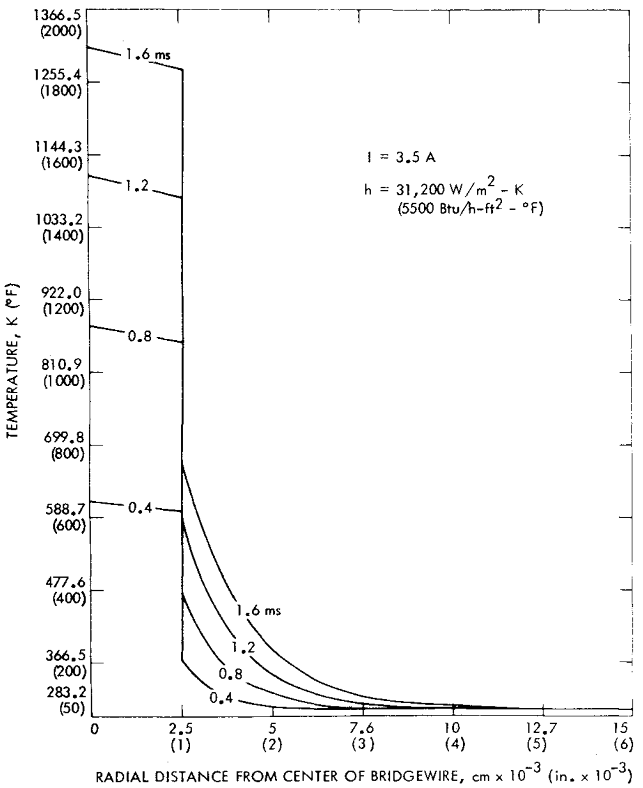

Fig. 2 Radial temp distribution within the bridgewire-pyrotechnic system at 0.4 ms intervals.

With the value of h fixed at 31,200 W/m^2-K and the ignition temperature fixed at 672 K the computer program was run with different ambient temperatures and higher current levels for which test data were available. The purpose of these runs was to test this model's ability to predict ignition times at those experimental conditions. The results are given in Table 2 which shows a comparison between the predicted ignition time and the experimentally measured time to bridgewire burnout for Mariner Mars '71 squib firing tests ranging from 144-366°K (−200- +200°F) for both 3.5 and 5 amp levels. The agreement is quite good. In this comparison it was assumed that bridgewire burnout occurred at pyrotechnic ignition.

Table 2 Mariner Mars '71 firing data comparison

Constant current, amp	Ambient temperature, °K (°F)	No. of firings	Measured average time to burnout of bridgewire ms	Range, ms	Model prediction of time to reach 672°K (750°F), ms
3.5[a]	294 (70)	20	1.59	1.45-1.79	1.6
3.5	144 (-200)	10	2.02	1.8 -2.2	2.27
5	294 (70)	3	0.84	0.80-0.88	0.82
5	144 (-200)	4	1.08	1.05-1.1	1.1
5	366 (+200)	4	0.58	0.51-0.71	0.70

[a]Test conditions used for initial estimate of h (see Fig. 1.)

The good agreement between the predicted time to ignition and the measured time to bridgewire burnout seems to indicate that this model has some merit in spite of the many simplifying assumptions that were incorporated into the model. One of the significant aspects of this model is the interfacial boundary condition that stipulates a thermal contact conductance between the bridgewire and the pyrotechnic. This boundary condition greatly affects the temperature distribution and the heat transfer within the bridgewire-pyrotechnic system. Figure 2 shows the predicted temperature distribution within the system at 0.4 ms time intervals for the 3.5 amp firing at room temperature. A number of important points about the bridgewire-pyrotechnic system can be demonstrated with this figure. First, it can be seen that a sizable temperature difference can occur between the bridgewire and the pyrotechnic even when a very large thermal contact conductance is used. This results primarily because of the large heat flux out of the bridgewire. Second, there is very little temperature difference between the center of the bridgewire and the outside surface of the bridgewire. Finally, the heat does not diffuse past 100 μm (0.004 in.) within the pyrotechnic during the first 1.6 ms.

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Reliability Simulation for Solar Electric Propulsion Missions

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Introduction

CANDIDATE planetary, cometary, and geosynchronous missions with solar electric propulsion (SEP) are being studied for missions in the 1980's. Important aspects of any such mission studies are the system tradeoffs and reliability analyses. Such studies of alternate configurations and designs are needed to provide guidelines for hardware and subsystem requirements.

On simpler ballistic missions, a number of approximations and realistic assumptions can be used to simplify mission reliability tradeoff studies. In solar electric propulsion missions, the mission requirements are so complex that tradeoff studies become more difficult, and any simplifications must be considered with great care, for often they can affect the results. Indeed, many simplifications can no longer be made. In a typical SEP mission, a number of complexities influence mission tradeoff studies. Many of these complexities evolve from the mission requirements of providing thrust that is variable and dependent on a number of parameters. These parameters include such factors as available solar power (dependent on spacecraft-solar distance) as well as variations in hardware performance, hardware interfaces, and hardware wearout and reliability.

Some of the complexities can be summarized as follows: 1) Multiphase Missions—where different sets of hardware are required to work during the various portions or phases of the missions; 2) Complex Switching—where switching is needed for redundancy and for normal operations; and 3) Wearout Failure Modes of Components—including such items as switches and thrusters. The simplifying assumption of the constant failure rate for electronic components is not applicable to many of the components used on a SEP mission. This is because the constant failure rate assumption implies that there is no wearout.

Because of these and other complicating factors, analytical techniques are often insufficient for performing mission reliability tradeoff studies of alternate designs or configurations of a SEP spacecraft. Monte Carlo simulation provides an alternative and viable method for performing many of these analyses.

The objective of this paper is to very briefly describe the Monte Carlo method and elaborate on the complicated assumptions that can be modeled when using Monte Carlo simulation. This will be done principally by illustrating an application to an Encke Comet rendezvous mission reliability tradeoff study for interconnecting power processors and ion thrusters, the major elements of a thrust subsystem.

Monte Carlo Method

The Monte Carlo method provides a means, through the use of pseudo-random numbers, for simulating both probabilistic events and deterministic events that could occur during an actual mission. Examples of probabilistic (or random) events would be component failures and trajectory errors. Examples of deterministic events would be the

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Index categories: Computer Technology and Computer Simulation Techniques; Reliability, Quality Control and Maintainability; Electric and Advanced Space Propulsion.

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