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DETERMINATION OF THE EXTENT OF ION THRUSTER EFFLUX DEPOSITION ON SPACECRAFT SURFACES FROM THE SERT II FLIGHT THERMAL DATA

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DETERMINATION OF THE EXTENT OF ION THRUSTER EFFLUX DEPOSITION ON SPACECRAFT SURFACES FROM THE SERT II FLIGHT THERMAL DATA

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Abstract

In the studies of proposed electric propulsion missions one of the areas of concern is the possible contamination of spacecraft instruments and thermal control surfaces by exhaust particles from an ion thruster. There have been vacuum tank tests conducted in ground facilities to determine the extent of this deposition by thruster exhaust particles, but the application of these results to long term space missions is questionable. The flight thermal data from the SERT II satellite, the only electric propulsion mission with an extensive thruster operational history, has been reviewed specifically to see if there is any evidence of contamination that could be attributed to the 5860 hours of mercury bombardment ion thruster operation. This evaluation of the flight data has shown that the only evidence of deposition occurred on the contamination experiment solar cells which are located at the edge of the thruster exhaust beam. There is no evidence of any deposition of ion thruster efflux on any other surface of the satellite.

Introduction

In the past few years there has been a considerable number of missions proposed that would use electric thrusters for main propulsion devices on spacecraft. These missions range from a technology readiness flight that would use mercury bombardment thrusters to move a payload from a low Earth orbit to synchronous altitude,[1] to flyby and rendezvous missions with the comet Encke.[2,3] The characteristic of all of these missions is that they would require ion thruster operation for periods between one and three years.

One of the areas of concern with such a use of electric propulsion devices is that the thruster exhaust particles might return to the spacecraft and contaminate sensitive instruments or thermal control surfaces.[4] The possible efflux contaminates from the ion thruster exhaust are metallic elements or compounds. In sunlight, surfaces coated with these types of deposits would reach elevated temperatures and this could endanger the thermal control of a spacecraft. Such contaminates can also coat lenses and poison solid state detectors.

There have been contamination tests conducted in ground test facilities.[5-7] These tests indicate that the principal efflux materials from a mercury bombardment thruster are mercury and molybdenum from the accelerator. It has also been found that the deposition occurs only on those surfaces that had a line of sight to the accelerator grids. Other tests have been conducted in which the optical properties of thermal control coatings are examined before and after exposure to propellant fluxes.[8] No significant changes were observed in these ground tests.

As it is with most ground tests, the question can arise as to whether or not these results will apply in space. Tests in a ground facility must be run in vacuum tanks of finite size. The tank walls, then, can trap exhaust particles that could, conceivably, travel great distances in space and return. Hence, it is possible to suppose that the exhaust particles could eventually return to a spacecraft in space and yet not be able to detect such a deposition in a ground facility. Furthermore, since the proposed missions would require a long ion thruster operating period, the probability of deposition can be considered to be increased.

There has only been one long duration electric propulsion mission that can be used to verify the ground test data on surface contamination. That mission is the second Space Electric Rocket Test (SERT II). It is the purpose of this report to review the thermal data from this flight to determine if there is any measurable contamination on the spacecraft surfaces that could be attributed to the operation of the mercury thrusters.

The SERT II Satellite

Background

A brief summary of the SERT II satellite features will be given in this section. The satellite and its mission have been described in detail in the literature.[9] The current SERT II status is documented in reference 10.

In its orbiting configuration the SERT II satellite consisted of the Agena D booster; a spacecraft support unit (SSU) which housed the command system, telemetry, and control moment gyros (CMG's); and a spacecraft section which housed the two ion thruster systems and associated experiments (see fig. 1). The satellite power was provided by two solar array wings capable of generating about 1.5 kilowatts of power and located at the Agena end of the satellite. The ion thrusters were the one kilowatt, mercury bombardment thruster described in the literature.[11]

The satellite was placed in a 1000 kilometer (500 n. mi.), circular, constant sunlight, polar orbit on February 3, 1970. The satellite was gravity gradient stabilized and oriented such that the solar cells faced the Sun while the ion thrusters pointed towards the Earth. The oblateness of the Earth caused the orbit plane to precess at approximately the same rate as the Sun, thereby maintaining a constant sunlight orbit for the satellite for about nine months. After this time, the satellite was shaded by the Earth for portions of its orbit.

Thruster system #2 was started first and operated for a brief period and then shut down. Thruster system #1 was then started and operated at full beam current conditions (250 m amp) for the next

3810 hours with only a brief shutdown at a mission time of 750 hours due to a solar eclipse and an uncommanded shutdown at 2550 hours. At 4060 hours this thruster developed a short in the accelerator grids which terminated its operation. Thruster system #2 was then restarted and operated for the next 2050 hours, until a short in the accelerator also terminated the operation of this thruster.(12)

Flight Thermal Instrumentation

There was sufficient temperature instrumentation on this satellite to characterize the changes of the optical properties of most of the surfaces of the satellite. This review of the thermal data will be limited to four typical areas where deposition might be expected to occur. These areas are the contamination experiment solar cells, the ion thruster neutralizer tanks, the Z-93 white painted surfaces on the sun facing side of the spacecraft and SSU, and the main solar array wings.

Contamination experiment. The contamination experiment was designed to investigate the possibility of deposition of mercury on surfaces that were maintained at temperatures corresponding to 1 and 2 AU conditions.(13) The test surfaces were chosen to be solar cell array segments. The degree of deposition could be determined from those segments by monitoring the electrical characteristics (at a fixed load point) which would be a function of the transmission of the cover glass.

A contamination experiment was located behind each thruster (see fig. 2). In each contamination experiment there were two solar cell segments, each one of which consisted of a series string of ten 1×2 cm cells with fused silica cover glass. One of these segments, called the low temperature solar cell segment, was mounted directly on the extended surface radiator structure designed to simulate the 2 A.U. solar cell temperature (about -40° C). The sunside surface of this radiator was covered by an electrically grounded multilayer thermal insulation which had cutouts for the solar cell segments. The other segment, called the high temperature solar cell segment, was mounted to the same radiator plate using thermal isolation techniques which allowed that segment to operate at about 60° C (1 A.U. condition). There was a temperature sensor on each radiator plate (behind the low temperature solar cell segment) and on the back of each high temperature solar cell segment.

Thruster. The thermal instrumentation on the ion thrusters was limited to the surfaces that were at electrical ground. This meant that the thermal performance of the thruster had to be determined from the neutralizer tank temperature. The location of this temperature sensor is shown in figure 3.

Sunside panels. The six panels covering the sunside surfaces of the spacecraft and SSU (identified as the Bay 1, Bay 8, and Bay 7 panels) were coated with a zinc oxide, potassium silicate white paint called Z-93. The white paint was required so that these panels would act as radiators while in the sunlight. The Z-93 was chosen because it was the most stable white paint available for spacecraft thermal control applications. Since the SERT II thermal control was to be accomplished by passive means only, the stability of the white paint while exposed to the ultraviolet radiation in space was a factor of prime importance.

While the Z-93 paint was expected to be stable in space, it was known to be highly susceptible to contamination on the ground. Dust, dirt, and any vapors could get into this porous paint surface and drastically change the coating optical properties. Extreme care had to be taken to avoid this possibility. Hence, it would appear that these surfaces would be ideal sensors for possible contamination from ion thruster exhaust products.

Each of these six panels had an area of about 0.37 square meter (4 ft^2) with the two main sunline panels (Bay 8) completely coated with the white Z-93 while three quarters of each of the other four panels were painted. At the center of the inside of each panel was a temperature sensor.

Main solar arrays. The main solar array wings were mounted at the Agena end of the satellite and generated the 28 volts d.c. power for the satellite housekeeping systems and the 56 volts d.c. power for the thruster operation. The total power generated was 1.7 kilowatts at beginning of life. Each array wing was 1.5 × 5.8 meters (5×19 ft). The position of these arrays was about 6 meters (20 ft) away from the ion thrusters and so could be used to determine if any efflux particles were following a curved trajectory.

Flight Thermal Data

The flight temperature data for the selected satellite surfaces are shown in figures 4 to 7. As can be seen from these figures, there is no catastrophic temperature rise that would indicate deposition of a very pure metallic coating. The detailed discussion of each of the four areas will now be presented.

Contamination Experiment

The flight temperature history of the high and low temperature solar cell segments (radiator plate) are shown in figures 4 and 5. The curves marked experiment #1 refer to the segments behind thuuster #1 while the experiment #2 curves refer to the segments behind thruster #2. Since the temperature trends are more pronounced for the thermally isolated, low mass, high temperature segments, the discussion will be limited to this data. The same conclusions can be reached from the low temperature segment data.

It is obvious from the data (see fig. 4) that the segments are being coated by the ion thruster deposits. When each thruster was started, the segment temperature dropped indicating that the optical properties of the surface were changing. The electrical data for the segments(13) showed that the glass transmission was decreasing such that the glass became opaque within 40 hours of thruster operation. This result verified the ground test data(6) which indicated that any surface that had a view of the thruster accelerator would be coated by thruster efflux at a rate depending upon the view factor to the accelerator. The principal coating products obtained in this ground facility study were sputtered grid material particles which were being randomly distributed.

After the coatings on the cover glass became opaque, the temperatures of both segments rose even though only thruster #1 was operating. This indicated that the coating process was continuing. The rate of temperature rise on both segments was different which would be expected since the view factors to the accelerator of the operating thruster are different. At about 450 hours into the mission (175 hrs of thruster operation), the temperature of the #1 segment started to fall while the #2 segment continued to rise slowly. Stable temperatures were reached on both segments at about 1200 hours. The inflection in the temperature curve for the #1 segment at about 2400 hours in the mission may have been associated with a thruster #1 uncommanded shutdown (arc off) that occurred at about this time. When thruster #1 was shut down and the #2 thruster started (at about 4100 hrs), the thermal situation was reversed with the #2 segment temperature falling while the #1 segment temperature rose. Temperature stability was reached about 1000 hours later for segment #2 but the temperature of the #1 segment continued to rise slowly until the 6000 hour mark in mission time. Thruster #2 ceased operations at about 6150 hours.

An analysis of this thermal behavior was conducted to deduce the apparent changes in the optical properties of the cover glass due to the deposits. The goal of this analysis was to determine whether the deposits were a pure metallic coating which would have a ratio of solar absorptance to emittance (α_S/ε) of greater than 10, or a metallic compound with an α_S/ε of between 1 and 5. The contamination experiment was not designed as a thermal control experiment and so, there are insufficient controls to be able to determine absolute values of solar absorptance and emittance.

The analysis of the temperature data required the solution of the following simplified steady-state heat balance equation:

Absorbed solar flux + Absorbed Earth thermal flux = Radiated flux + Losses

$$I_S\alpha_S A_F + I_T(\varepsilon_F F_F A_F + \varepsilon_B F_B A_B) = \sigma(\varepsilon_F A_F + \varepsilon_B A_B)T_S^4 + Q_{LOSS}$$

where

I_S	incident solar flux on segment
I_T	incident Earth thermal flux
α_S	solar absorptance of deposited coating
ε_F	emittance of the cover glass and coating
ε_B	emittance of back of segment (fixed at 0.17)
F_F & F_B	view factors of front and back of solar cell segments to space
A_F & A_B	area of front and back of solar cell segments
σ	Stephen-Boltzmann constant
T_S	absolute temperature of solar cell segments
Q_{LOSS}	generalized heat loss term

The Q_{LOSS} term was evaluated from the temperatures and the initial properties of the solar cell segment before the thruster start. This was found to be a strong function of the adjacent radiator plate temperature (see fig. 5). The heat balance equation assumes that the segment is in thermal equilibrium at each point in time. This is a reasonable assumption since each solar cell segment had a low thermal mass. Transient data obtained during flight eclipses verified this assumption. This equation lumps such additional heating terms as the absorbed albedo flux and the absorbed spacecraft reflected flux into the heat loss term.

After the thruster was turned on, the solution of the heat balance involved two unknowns: the solar absorptance (α_S) and the emittance (ε_F). The equation was solved for α_S over a range of values of ε_F at various times in the mission. Then, by assuming that both of these optical properties were smoothly varying parameters, it was possible to construct a set of curves for these values. The result of this analysis is shown in figure 8 and will be discussed in the following paragraphs.

The thruster start results in the deposition of a coating on the solar cell segment cover glass causing a change in both the solar absorptance and the emittance. However, the initial change in the solar absorptance is more rapid than the change in the emittance, which results in a decreasing temperature. This decrease in temperature corresponds to the decrease in the transmission of the cover glass in the first 40 hours of thruster operation. After this deposited coating becomes opaque, the process continues resulting in larger changes in the emittance and smaller changes in the solar absorptance. Hence, the segment temperatures rise. Finally, both properties tend to stabilize. The coating deposition rate occurs much more rapidly for the segment behind the operating thruster as expected.

When the thruster #1 shut down and the #2 thruster started, there appeared to be an acceleration in the deposition on the #2 segment. This changes the solar absorptance of the coating to approximately the value reached by the #1 segment more rapidly than the emittance change resulting in an initial temperature drop. Both optical properties of the coating stabilize in about 500 hours. The #1 segment temperature rise during this period is assumed to be due to an increase in the solar absorptance rather than a decrease in the emittance. There is insufficient data to verify this assumption.

For both segments the ratio of solar absorptance to emittance of the deposited coating reaches a value of 2.25 to 2.50 after the 5860 hours of thruster operation. This value corresponds approximately to that of a rough coating of molybdenum, the accelerator grid material. While the deduced values of α_S and ε resulting from this analysis are not absolute, a valid conclusion can be reached that the deposits from an operating ion thruster on glass will be of the rough surfaced, metallic type with an α_S/ε less than 3 and not a pure metallic coating with high values of α_S/ε. In addition, this flight data verifies the results obtained in vacuum tank testing.

Ion Thruster

The neutralizer tank temperature history is shown in figure 6. The thruster #1 neutralizer tank temperature reached equilibrium prior to the thruster start. Then, the temperature rose to an equilibrium value and remained at this value throughout the thruster #1 operational life. When the thruster shut down, the neutralizer tank temperature returned to its original value. The same trends were found with the #2 thruster neutralizer tank temperatures. Since the neutralizer tank temperature is a function of its optical properties, and since the tank temperatures returned to the same initial values that it had after the thruster operation, it can be concluded that there were no thermally significant coatings deposited on the neutralizer tank.

Independent testing of the thruster in a solar simulator and a detailed computer thermal analysis of a thruster have shown that a change in the optical properties of the external surfaces of the thruster of a factor of two would cause approximately 20° C change in the neutralizer tank temperature. Since the flight thermal instrumentation could discriminate a change of temperature of about 1.5° C, gross changes in the thruster optical properties of about 20 percent could be determined from the neutralizer tank temperatures. Therefore, it can be further concluded that the neutralizer tank temperatures indicate that the thruster external surface optical properties did not change appreciably due to the ion thruster operation.

Sunside Panels

As stated previously, the sunside panels of the spacecraft and SSU were coated with Z-93 white paint. This paint is known to be highly sensitive to contamination from dust, particles, and vapors, and therefore, should be a good sensor for possible deposition of ion thruster exhaust particles. The flight thermistors are located in the center of each panel. This placed the thermistors about 76 centimeters (2.5 ft) from the thruster beam axis and about 45 centimeters (1.5 ft) and 110 centimeters (3.5 ft) below the thruster.

The temperature history for these two sunside panels is shown in figure 7. A detailed computer thermal model of the SERT II satellite was developed and calibrated[14] to be able to determine the change in the optical properties of the thermal control coatings. The analysis was accomplished and reported for the Z-93 white paint.[15] The resulting changes in the solar absorptance for the paint on the sunside panels is shown in figure 9.

The change in the solar absorptance of the spacecraft sunside panel (bay 8) is well within the expected results based on laboratory ultraviolet degradation tests.[16-19] The SSU sunside panel (bay 8) has a more pronounced change in the solar absorptance. This degradation is believed to be due to the proximity of the paint on this panel to the shroud mounting ring which was coated with lubricants to guarantee shround separation. The degradation is not due to the ion thruster operation, since the change in solar absorptance started at the beginning of the mission well before the first thruster start. The other two panels of the SSU (bays 1 and 7) that were coated with a pattern of Z-93 paint and polished metal had the polished metal portions next to the shroud mounting ring. The optical properties of the Z-93 paint of these panels followed the same trends as the spacecraft sunside panel.

Therefore, there is no evidence of thruster exhaust particles curving back to the satellite body and causing degradation of sensitive thermal control coatings.

Main Solar Array

The temperature for the midpoint thermistor of one solar array wing is shown in figure 6. The other five flight thermistor readings agree with this data point varying slightly in magnitude due to temperature sensors. The temperatures indicate that there are no significant changes in the surface optical properties of either the covered solar cells or the paint on the rear surface of the solar array substrate. The variation in the temperature is due to the changes in the solar flux caused by the differences in the angle between the Sun and the array normal as the satellite orbited. The electrical characteristics of this array have been reported as being within the expected values for this mission.[20] Therefore, it can be concluded that no ion thruster exhaust particles have followed a curved trajectory such that they could deposit on the main solar array.

Concluding Remarks

The flight thermal data from the SERT II mission has been reviewed to determine if there is any evidence of contamination from ion thruster efflux particles. This mission was flown in a 1000 kilometer (500 n. mi.) polar orbit so that the space conditions are not identical to those found at synchronous altitudes. However, this mission is the only one in which the ion thrusters have operated continuously for extended periods of time in space.

The thermal data from the solar cell segments of the contamination experiment does indicate that these segments were coated with ion thruster products. The deposition was such that the transmission through the cover glass went to zero after about 40 hours of thruster operation. Since these segments have a line of sight view of the operating thruster accelerator grids, degradation can be expected. Thermal analysis to deduce the optical properties of this deposited coating indicate that it could be molybdenum sputtered from the accelerator grids. This coating has a ratio of solar absorptance to emittance of about 2.5 and no evidence of a higher ratio that would be anticipated from deposition of a pure metallic coating could be found. Both the segment behind the operating thruster and the segment behind the nonoperating thruster were coated with these deposits apparently at rates dependent upon the view factor to the respective operating thruster accelerator. Coating in this manner is in agreement with ground test data.

There is no evidence of contamination from the thruster exhaust on the ion thruster external surfaces, the satellite body, or the main solar array. This result also is in agreement with the ground test data that has been obtained.

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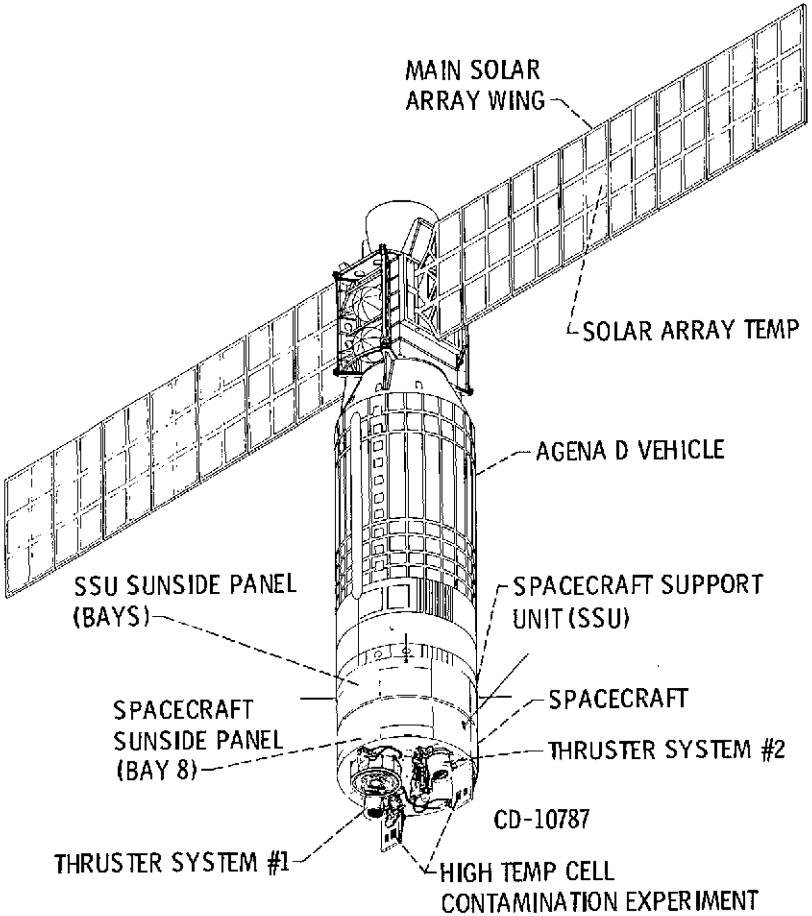

Figure 1. - Component & Temperature Sensor Locations Sert II Satellite.

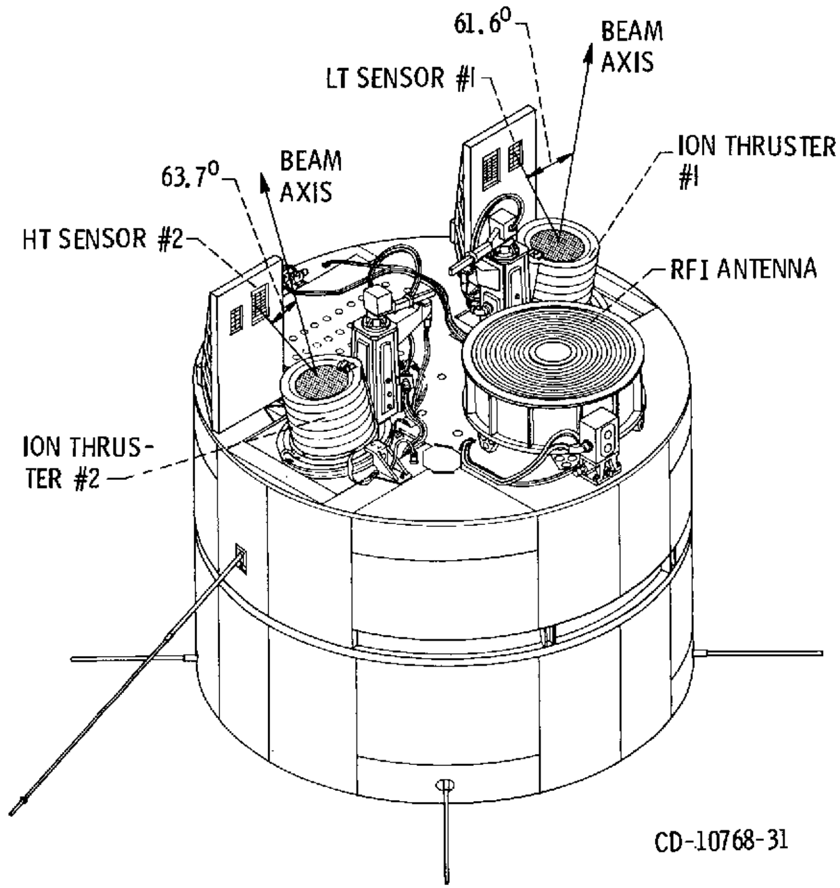

Figure 2. - SERT II spacecraft showing contamination sensor locations.

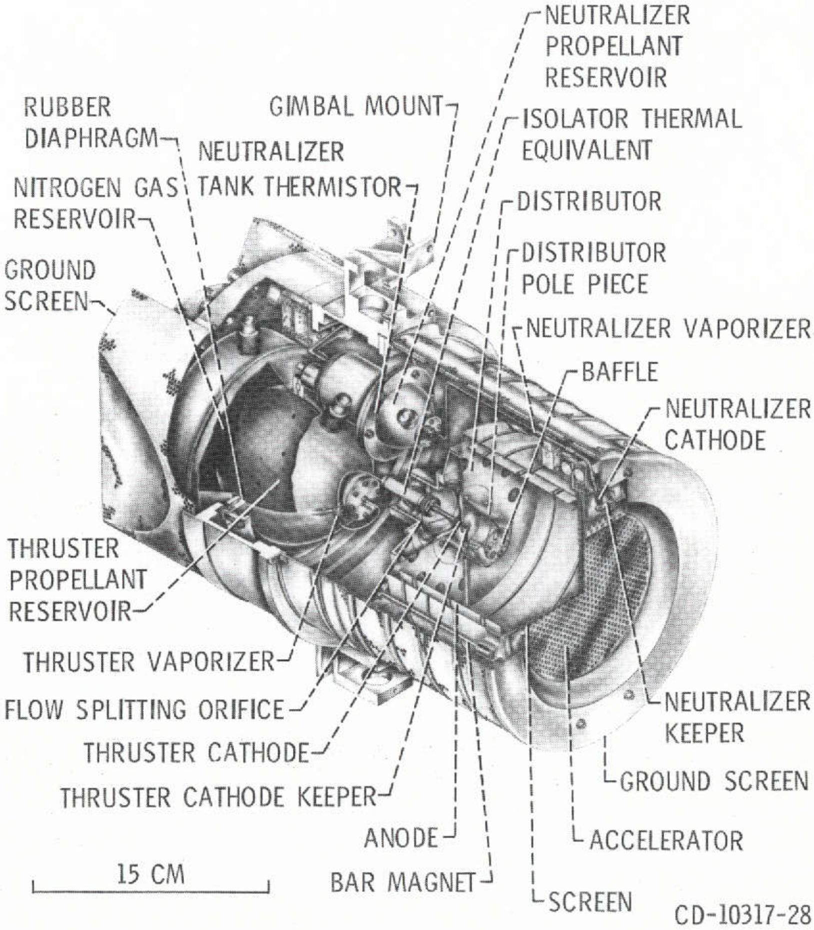

Figure 3. - SERT-II thruster.

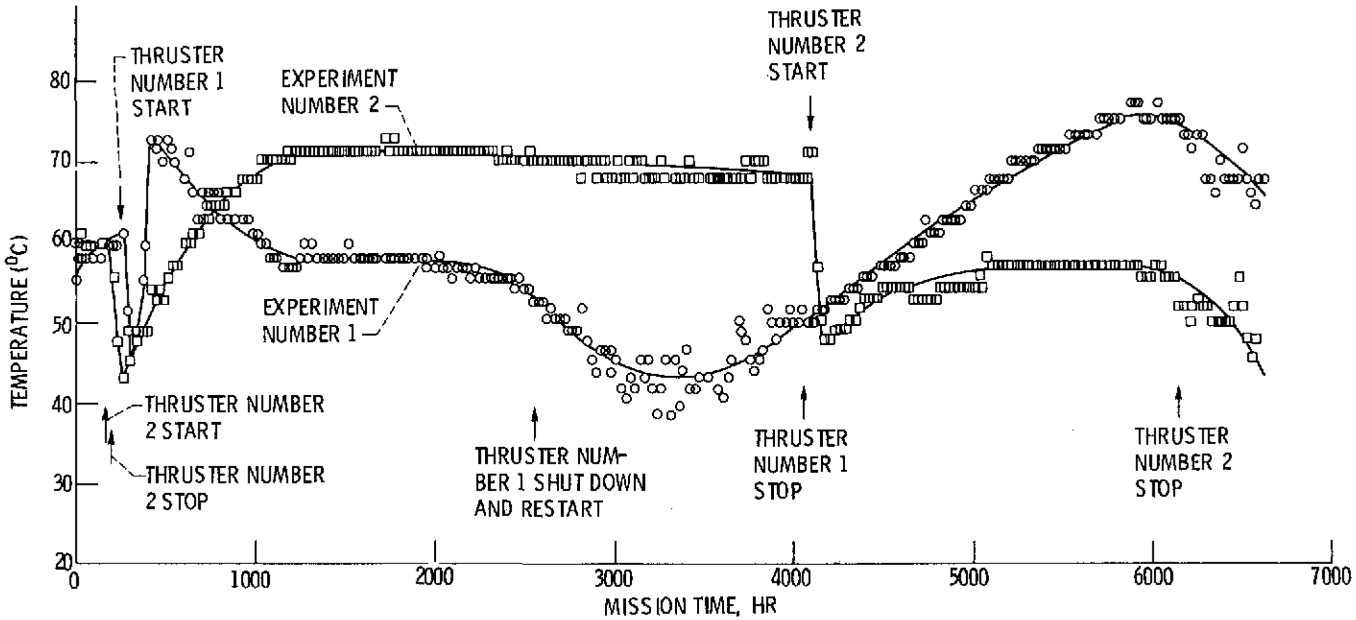

Figure 4. - Contamination experiment solar array temperatures - high temperature cells.

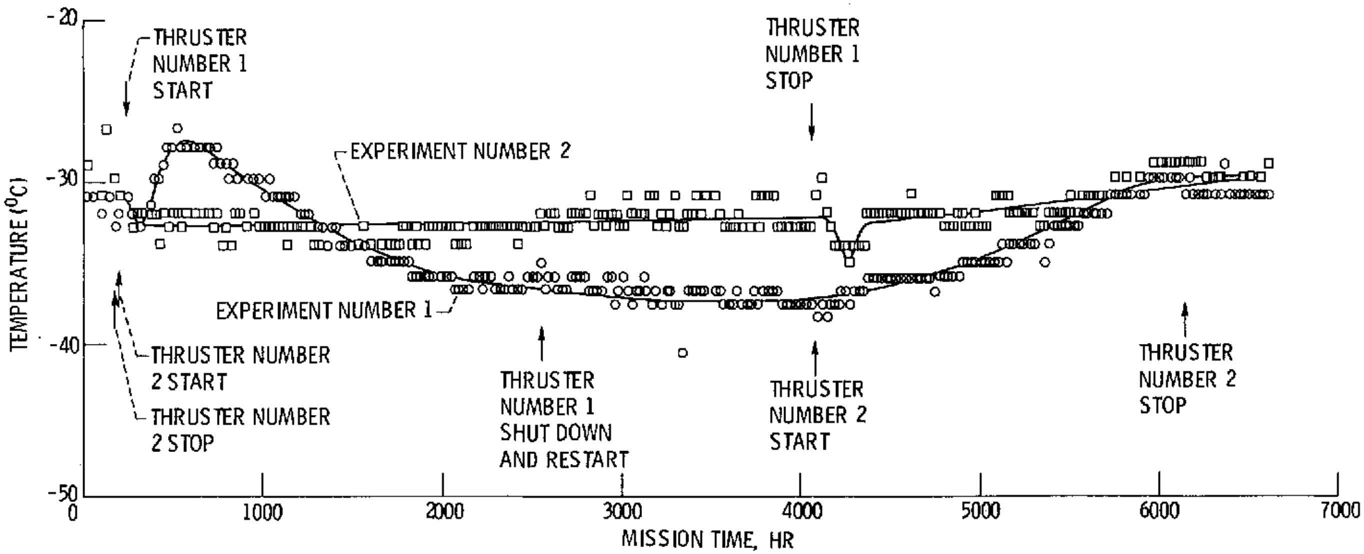

Figure 5. - Contamination experiment solar array temperatures - low temperature cells.

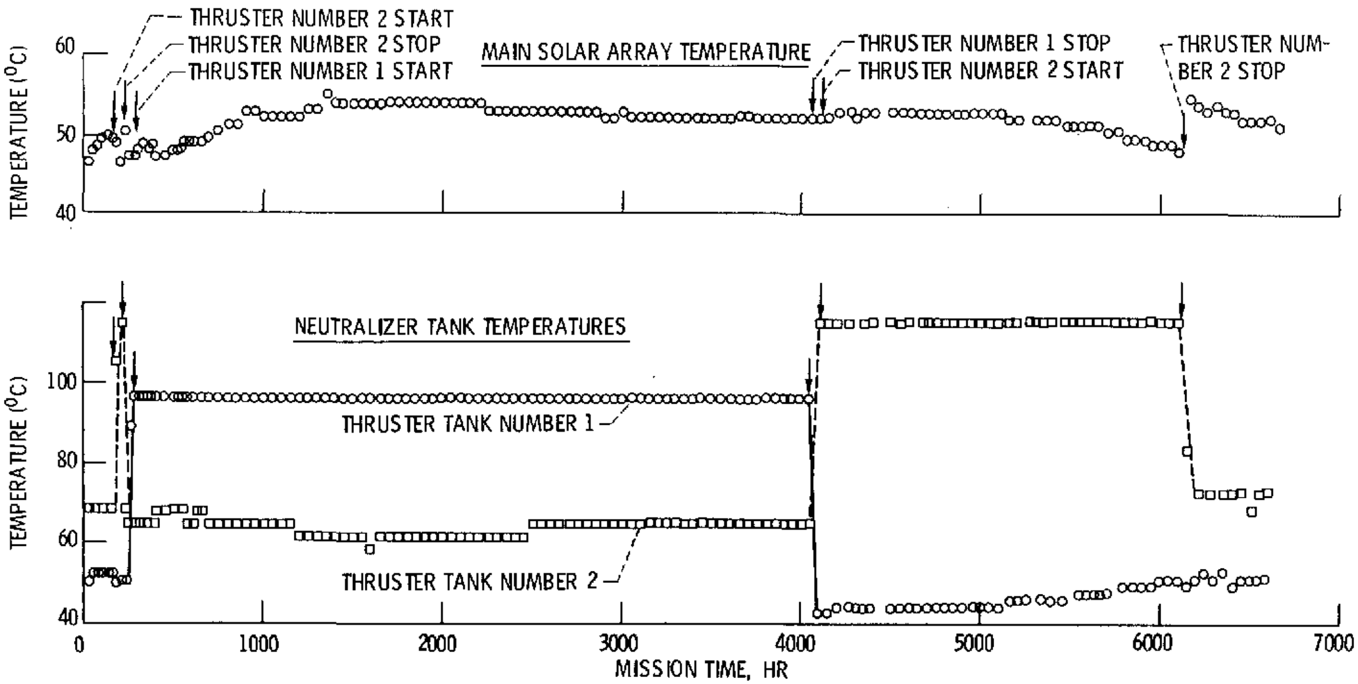

Figure 6. - SERT II temperature data.

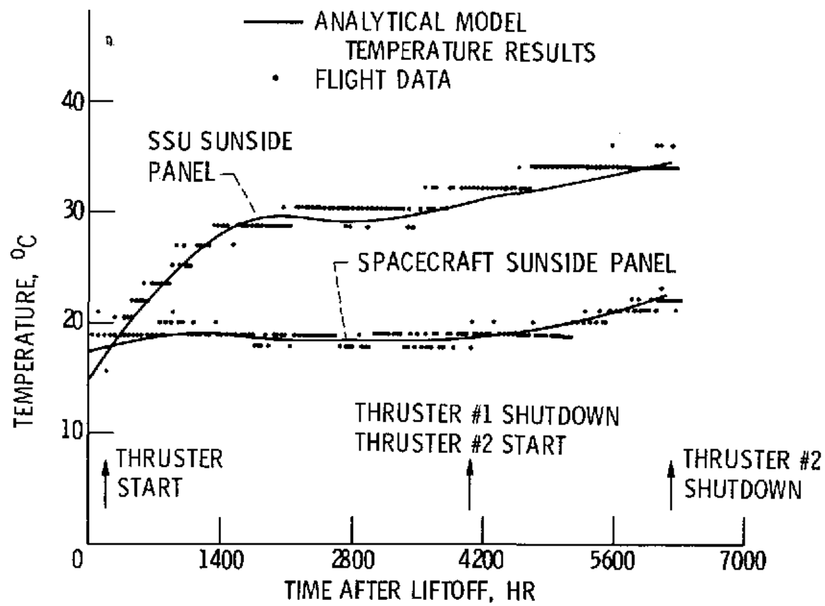

Figure 7. - Flight temperatures of the sunside panels.

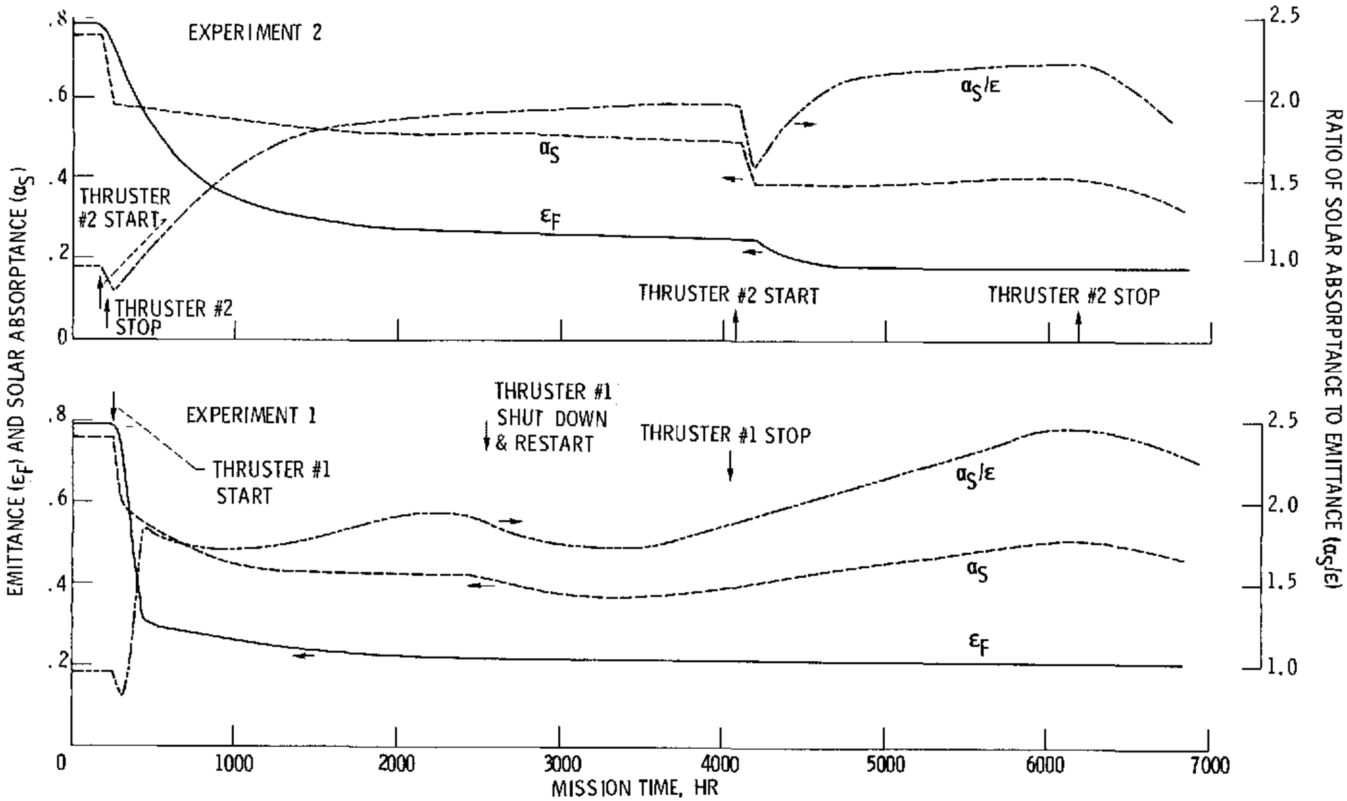

Figure 8. - Estimated optical properties of coatings deposited on high temperature solar cell contamination experiment SERT II.

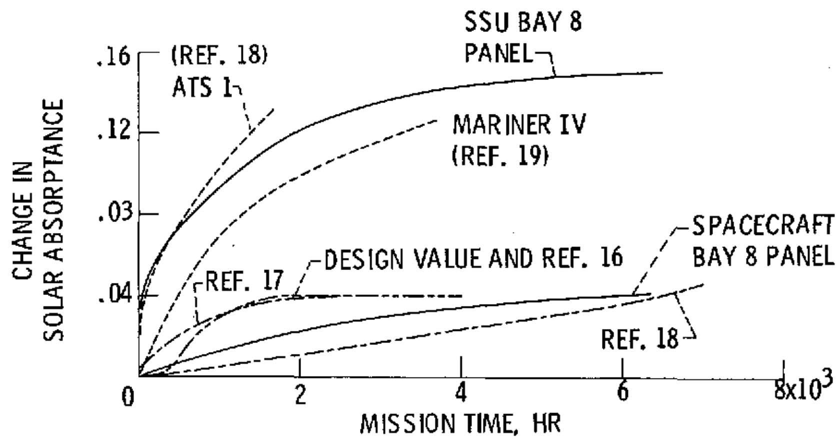

Figure 9. - Change in solar absorptance of SERT II Z-93 paint.