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REPORT ON THE FLIGHT PERFORMANCE OF THE Z-93 WHITE PAINT USED IN THE SERT II THERMAL CONTROL SYSTEM

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Abstract

The change in absorptance of the 1.8 square meters of Z-93 white paint applied as the primary component of a passive thermal control system is evaluated by varying the paint absorptance values used in a calibrated computer model of the satellite until the resulting temperatures matched the flight data. This absorptance change is determined for a 6200 hour, constant sunlight period in a 1000 kilometer orbit and found to have increased by 0.04 as expected. It has been concluded that inadvertant contamination has caused higher degradation of one area. This area of higher degradation has not caused thermal problems. Hence, the white paint system is functioning satisfactorily.

Introduction

The second Space Electric Rocket Test satellite (or SERT II) is a long duration, space environment test vehicle for a one kilowatt, mercury bombardment, ion thruster. In its orbiting configuration the SERT II satellite consists of the Agena D booster, a spacecraft support unit (SSU) which houses the command system, telemetry and control moment gyros (CMG's), and a spacecraft section which houses the two ion thrusters and associated experiments (see fig. 1). After reaching a stable orbit, the Agena was shut down and has remained dormant. Power for the satellite is supplied by two solar array wings attached to the Agena.

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The satellite was placed in a 1000 km (540 n. mile), circular, constant sunlight polar orbit on February 3, 1970. The orientation is with the ion thrusters pointing towards the earth and the solar cells facing the sun. The satellite is gravity gradient stabilized and does not spin. The oblateness of the earth has caused the orbit plane to precess so that the satellite remained in constant sunlight until November 23, 1970. At this time the anticipated shadowing of the satellite for part of each orbit began. This periodic interruption of solar power has required that activities with the satellite be minimized until it again drifts into a constant sunlight orbit in February, 1971.

A passive thermal control system is being used on this satellite.(1) This system consists of Z-93 white paint, aluminum surfaces and black paint. The purpose of this report is to present the flight performance evaluation of the major component of the thermal control system, the Z-93 white paint. The evaluation will cover the period from launch until the premature shutdown of the second ion thruster on October 17, 1970, a total time of 6200 hours in a constant sunlight, space environment. This is believed to be the first such evaluation of Z-93 white paint applied as part of a passive thermal control system over this extended period of time in space.

The evaluation of the paint performance in this mission is complicated by the fact that the coated panels are integral parts of a structure and are not thermally isolated. Changes in temperature, therefore, can not automatically be correlated to changes in optical properties of the coating. The procedure used in this evaluation is essentially the same as that used in the design of the thermal control system. The analytical thermal model, using the design optical properties of the coatings, was subjected to the space environmental heating factors and internal heat dissipation at a given time in the mission and these analytical results compared to the flight data. The space environmental heating inputs were evaluated at an early point in the mission to ensure that all heat inputs were included. Differences between the flight data and the analytical model results, then, were proportional to changes in the solar absorptance of the coating.

Since the analytical thermal model is such a necessary tool in this evaluation, it will be discussed in the next section along with the other components of the SERT II thermal system. Then, the flight data will be discussed and the resulting solar absorptance curve for the 6200 hours of flight will be presented.

Discussion of the Thermal System

Analytical Model

The basic tool used in this evaluation of the coating performance is the analytical thermal model of the satellite. Since this model would be so important to the project, considerable effort was expended in making the model accurate and detailed enough to be useful. The computer program chosen to solve the analytical thermal network was the Chrysler Improved Numerical Differencing Analyzer or CINDA.(2)

The model consists of a 109 node representation of the spacecraft section, a 81 node representation of the SSU and the Lockheed analytical model of the forward equipment rack for a total of 534 nodes. Heat conduction tests on the shims used between the SSU and Agena rack had indicated that the SSU would be essentially thermally isolated from the Agena. Hence, these sections were sufficient to treat the thermal considerations for orbiting conditions; the Agena rack was used to satisfy the radiation boundary conditions for the two sections containing all of the heat dissipating components.

The nodal layout of the spacecraft and SSU analytical model is shown in Fig. 2. The majority of the nodes are assigned to the structure. Components on each tray are generally lumped into a sin-

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gle node while the tray itself is assigned a separate node. The two ion thrusters which are essentially thermally isolated are assigned one node each. A joint thermal conductance value of about 316 watts/m^2 was experimentally obtained and incorporated into the model for all structural joints. Internal radiation interchange factors were obtained either by hand calculations or by use of CONFAC II.(3)

Calibration of the Analytical Model

In setting up an analytical model, some of the assumptions which must be made might invalidate the accuracy of the model. In order to guard against this possibility the SERT II analytical model was calibrated against temperature data obtained in a test program.

This calibration was accomplished by instrumenting the prototype spacecraft, SSU and simulated Agena forward rack with 144 thermocouples and collecting data for a total of 160 hours under a variety of external heating conditions and internal power levels. As a boundary condition on the computer model calculations the Agena rack temperatures and the heat receiving panels of the spacecraft and SSU were all held at the experimentally determined temperatures. By a laborious iterative process involving changes in solid conductors, radiation interchange and heat storage capacities the agreement between the computer model results and the experimentally obtained temperatures was made excellent for both transient and steady state results. The details of the test procedure can be found in Ref. 1. The results for the areas of interest in this report for 100 hours of the test program are shown in Figs. 3 (a), (b), and (c). As can be seen from the curves the agreement is within 4°C with the predictions generally colder than the thermistor readings. Hence, the analytical thermal model was calibrated for internal heat exchange and radiation losses to the environment provided the heat receiving panel temperatures are known.

Thermal Control System

The analytical thermal model was used to determine the thermal control pattern that would satisfy the component temperature constraints for the mission. This pattern is shown in Fig. 4. The Z-93 white paint is used on the six sunside panels; an area of about 1.8 square meters. The main sun axis panels of both the spacecraft and SSU (bay 8) are completely covered with Z-93 paint while the other four panels are coated with a pattern of three-quarters white paint and one-quarter polished aluminum. The optical properties of the Z-93 paint used in designing the thermal pattern were:

Solar absorptance of 0.16±0.03
Emittance of 0.90±0.05

The Z-93 paint was chosen as the solar reflector coating because of its reputed stability in sunlight in near Earth orbits (4-7) where particle included damage would be minimized. It was felt that the paint disadvantages of easy contamination and difficulty of application could be overcome. This confidence was based on the knowledge that the panels to be painted were removable ones that could be replaced if contaminated and that experimental work with the paint indicated there would be no difficulty in applying the paint. The paint was kept from being contaminated up until launch by maintaining two complete sets of panels; one set which underwent all of the flight acceptance testing with the structure and a flight set which were attached to the satellite just prior to shroud installation. The paint on the flight set of panels was applied within a week of the test set by the same people, using the same procedures as the test set. The flight panels were subjected only to a thermal-vacuum test in an ion-pumped facility and then packaged together for shipment to the launch site. Paint evaluation samples that were painted at the same time as the flight panels were monitered for optical properties at various stages of the flight panel test, packaging and final installation. There were no changes detected in the optical properties of these paint samples.

The paint's reputation for being difficult to apply became apparent in the early prototype phase of the program. The paint applied to the magnesium panels began to flake off under ambient conditions about one week after application (see fig. 5). The panel substrate was changed to aluminum and a program undertaken to relearn the secret of successful application of Z-93 paint. The first lesson learned was that the cleanliness of the substrate is a major contributor to good paint adhesion. This alone did not solve the application problems for the paint texture would vary between rough, sandy coatings and smooth, glossy finishes. The glossy coating looked well but did not exhibit the adhesion of the sandy coating in the bend adhesion test (see fig. 6). The difference was finally found to be in the drying of the paint: in periods of high humidity the water did not evaporate and the paint dried to a smooth, glossy texture. The addition of a bake out immediately after application resulted in a panel that was uniformly coated with a rough, sandy textured coat of Z-93 paint that had excellent adhesion. The details of the successful application of Z-93 paint can be found in Ref. 8.

Flight Thermal Instrumentation

Of the 57 flight thermistors on the SERT II satellite, 23 are attached to the spacecraft and SSU structure. The evaluation of the total thermal control system requires the matching of the computer results with all 23 of the flight temperatures. However, the performance of the Z-93 paint can be demonstrated by an inspection of only 5 flight readings consisting of two panel temperatures and three internal temperatures (see fig. 7).

The flight temperatures are digitally coded in the spacecraft, returned to Earth by the telemetry system and then computer converted to temperatures. This flight thermistor system was calibrated against thermocouple readings at the same time that data was being collected for the calibration of the analytical computer model. It was found that the thermocouple-thermistor agreement was generally within ±3°C. This corresponds to a ±1 count which is the estimated accuracy of the telemetry system.

Discussion of Flight Results

Heat Dissipation

The direct solar flux is the most important of the external heating fluxes. The angular variation of the orbit plane with respect to the Earth-Sun line was known for the mission life. This angle varied from 25° at the time of launch to a minimum of 4° at about 1600 hours after launch, then increased to 15° at 3300 hours, decreased to 6° at 5000 hours and then increased to 25° again at 6200 hours. This information coupled with the seasonal variation of the solar flux was sufficient to compute the incident solar flux perpendicular to the orbit plane (see fig. 8). Average values for Earth thermal radiation and albedo were obtained from the literature.[9]

The SERT II orbit is fixed with respect to the Earth at a 9° inclination. The satellite is always oriented so that the ion thrusters face Earth and the solar array points toward the Sun. With the satellite in this configuration, there are two oscillations of the incident solar energy over each orbit (see fig. 9). There is a longitudinal oscillation in the solar energy incident to the Earth facing surface of the satellite over the polar regions (see figs. 9(a), and (c)). With the Sun in the southern hemisphere (as it was at the time of launch), the Earth facing surface is sunlit over the North Pole and shaded over the South Pole. The situation is reversed when the Earth moves so that the Sun is in the northerm hemisphere. In addition there is a lateral oscillation of incident sunlight on the sides of the satellite over the equator (see figs. 9(b), and (d)). This oscillation of incident flux has caused the flight temperatures of the four polished metal Z-93 painted panels to vary over a wide range complicating the analysis of this temperature data. The two, fully coated panels were centered on the main axis of the satellite and remained in the orbit plane. Hence, neither oscillation affected the solar flux incident to these two panels and, therefore, did not cause orbital temperature fluctuations in the flight data. The data from these two panels forms the basis of this analysis. The flight temperatures of the other four panels are used to support the conclusions derived from the analysis of the two main panels. For the analysis the oscillations in the incident solar energy were averaged over the orbit and applied to the appropriate panels as heat inputs.

The internal heat dissipation depends on the activity status of the satellite. The heat dissipation for the 6200 hour period considered by this report is shown in Fig. 10.

Comparison of Flight and Analytical Results

The analytical model was used to compute the expected steady-state temperatures for conditions of internal heat dissipation and environmental heating corresponding to 80 hours after launch. Design optical properties were used. These first computations resulted in predicted temperatures that were about 15° C below the measured flight temperatures. Further corrections to the environmental heat input were required. The solar heating terms were expanded to account for such items as: interior heating through the holes in the ion-thruster side of the satellite during the times when this surface was in sunlight, heating due to the numerous screwheads, and heating due to uncoated metal in the gaps between the panels and between the spacecraft and SSU. The value used for the solar absorptance of the Z-93 paint was increased to 0.17 which was within the tolerance of the design optical properties. After these factors were added the agreement was improved, but the SSU section was still too cold. An additional heat input to the SSU main panel was required in order to obtain reasonable agreement between the flight data and the analytical model results. After considering the possibilities, it was concluded that the solar absorptance of the SSU main panel must be increased above the value used for the spacecraft main panel paint. The agreement then became within expectations. This implied that the SSU panel paint had become damaged in some manner. This point will be discussed more fully in the next section.

This evaluation of the Z-93 paint is based on the comparison of flight temperatures and computer model steady-state results at eight points over the mission using the environmental heating and internal heat dissipation values corresponding to the point under consideration. All solar heating term corrections found to be necessary to obtain good agreement in the 80 hour case were modified to account for solar incidence angle and included in the computation. The paint emittance was assumed to be constant.

Parametric computer runs at each of the eight points were made using a range of values of paint absorptance to determine how the absorptance changed. Agreement of the panel and the internal temperatures was required to increase the confidence in the results. Agreement of the panel temperatures alone was not sufficient to insure the validity of the resulting change in absorptance.

The results of the comparison between the flight temperatures and the analytical model results for the bay 8 panels are shown in Figs. 11 (a), and (b). The analytical temperatures used in these figures are the best fit values to the flight data and, hence, determine the solar absorptance behavior of the Z-93 paint from the 80 hour point used to calibrate the environmental heating of the satellite. The solid line represents a smooth curve between the eight points used in the analysis. Thus, possible temperature fluctuations between the computed points due to changes in internal heat dissipation are disregarded. With this excellent panel temperature agreement then, one would expect that the internal temperature trends would be similar to those obtained in the calibration tests. The comparison of these internal temperatures for the flight conditions is shown in Figs. 12(a), (b), and (c). Again, the analytical model curve is drawn through the computed points. The agreement for these cases is also very good. The tendency for the battery temperature to be low in the analytical model results was noted in the calibration tests when the battery was not being charged or discharged. The fit in temperatures, as presented here, is felt to be the best possible within the accuracy of the model and the flight data.

Resultant Z-93 Paint Absorptance

The resultant change in the solar absorptance for the Z-93 paint used on the SERT II satellite is shown in Fig. 13. The paint on the spacecraft bay 8 has exhibited a slow, continual increase in solar absorptance over the initial value of 0.17. The nominal change for the 6200 hour period is 0.04. The design expectation for absorptance change was a 0.04 increase in 2000 hours based on the work reported in Refs. 10 and 11. The results reported in Ref. 12 is also shown on Fig. 13. The agreement with these literature values is reasonable.

The SSU bay 8 panel Z-93 paint absorptance change however, has shown the characteristics of the Z-93 paint samples flown on ATS I[13] and Mariner IV.[14] This SSU panel paint absorptance curve was generated by assuming that the absorptance of both the spacecraft and SSU panels was initially at the same value. This latter assumption is believed to be logical since both panels were painted within 5 minutes of each other on the same day, by the same man, with the same batch of paint, and cured, tested and packaged together. No significant difference in procedure can be found up to the final installation of the flight panels. After the assembly of the flight panels to the structure, both the spacecraft and SSU were wrapped in paper to protect the surfaces. The next day the paper was removed and the shroud put in place. With the shroud installed the environment until launch should have been the same for all panels. Paint samples were left in the gantry from the time the panels were removed from their containers until the shroud was installed. The optical properties of these samples were measured and found to be same as the values obtained when the panels were first painted.

The search for possible explanations for the degradation of the Z-93 paint on this SSU bay 8 panel started within a week after launch when the rising panel temperatures indicated the unexpected behavior. It is known that Z-93 paint can be degraded by charged particle fluxes. The change in the solar absorptance for this SSU panel paint appears to follow the data from the ATS I and Mariner IV flight experiments where particle induced degradation was considered a likely explanation for the damage mechanism. However, it is hard to visulize different particulate environments for the two bay 8 panels whose temperature sensors are only 0.6 meters apart. Degradation of the SSU panel from a charged particle flux from the ion thrusters was ruled out also since the degradation started before the thruster was turned on. Particle fluxes as a source of increased degradation for one of the panels were ruled out for the above reasons.

It has been postulated that the paint may have cracked and peeled off in a manner similar to that experienced by the prototype panels. Experience with the application of Z-93 paint has shown that cracking and peeling of the paint is a phenomena that takes place within a week or two of application. No such cracking or peeling of any of the flight panels was noted from the time of application in September, 1969, throught the testing and packaging, up to the final installation in February, 1970. Once the paint has survived the initial few weeks of drying without cracking, it usually remains firmly attached to the substrate. In addition it would be very fortuitous for the paint to crack off in such a fashion that would give the degradation characteristics noted in flight. Therefore, cracking and peeling of the paint is not believed to be a probable cause of the degradation of the SSU panel Z-93 paint.

After reviewing these and other machanisms to account for the SSU panel flight temperatures, it was concluded that the paint on this panel had become contaminated in spite of the precautions taken. The source of the contamination in unknown. However, a possible explanation of how this one panel might have been contaminated does exist. The shroud used was a clamshell variety that mounted on a ring immediately below the SSU. The shroud split line was in the center of the bay 8 panels. The clearance between the shroud and the SSU panel was about 2 to 3 cm. The spacecraft bay 8 panel was set back about 1 cm from the SSU so that, in installing the shroud, care had to be taken not to physically damage the paint on the SSU panel. In addition this SSU panel was painted down to the shroud mounting ring. A final connection had to be made at this split line, in front of the white paint, after the shroud was in place. A dry lubricant was also used on the mounting ring to insure that the shroud would fall away. Special precautions were taken to prevent damage to the paint from these causes. However, it is conceivable that contaminations could still have been introduced and localized in the SSU bay 8 panel paint when the shroud was installed.

The absorptance of the Z-93 paint used on the other four panels of the SERT II satellite can only be inferred from the flight panel temperatures because of the orbital fluctuation in solar heating mentioned previously. Based on a comparison of the flight temperatures with the computer model results it is believed that the solar absorptance of this Z-93 paint is following the trends of the paint on the spacecraft bay 8 panel rather than the SSU bay 8 panel (see figs. 14 (a), (b), (c), and (d)). The solid line represents the computed temperatures when the Z-93 paint degradation follows the spacecraft bay 8 absorptance. The dashed line in figs. 14 (c) and (d) represents the analytical results for degradation on these panels corresponding to that of the SSU bay 8 paint. An analytical investigation of this higher degradation rate was conducted for only the SSU panels. However, an increase in the paint absorptance for the spacecraft panels would necessarily result in an increase in panel temperatures similar to that experienced by the SSU panels. At the minimum solar incidence angle the lateral oscillation of solar energy is minimized reducing the range of temperature variations in these panels. Under this condition the analytical model results should be in good agreement with the flight data. When the angle is larger, the analytical model results should tend to be centered in the fluctuations of the flight data. As shown in Fig. 14, these conditions are met only if the Z-93 paint degradation for these four panels follows the spacecraft bay 8 panel absorptance curve. Hence, it is believed that only one of the six panels has been contaminated. The Z-93 paint on all other panels appears to be following the trends exhibited in laboratory tests for ultraviolet exposure.

In conducting this analysis it was found that a 0.01 change in the solar absorptance value would cause about 1.5°C change in the two main panel temperatures and about 2°C change in the other four panels. The uncertainity associated with the resultant absorptance curve (fig. 13) due to telemetry and model uncertainity is about ±0.02.

The 0.15 increase in the SSU bay 8 panel Z-93 paint has caused a panel temperature rise of about 20°C. The resulting internal temperature change, however, was only 5°C which is within the capability of the thermal control system. Therefore, even with this unexpected degradation of the paint on the SSU bay 8 panel, the Z-93 paint is functioning properly as the solar reflector coating for the SERT II satellite.

Concluding Remarks

This report has presented a flight performance evaluation of the Z-93 white paint used in the SERT II passive thermal control system. This evaluation is a determination of the change in the solar absorptance of the paint for a 6200 hour period. The evaluation is accomplished by a comparison of the flight data with the results of a parametric study using the thermal analytical model. The solar absorptance change of the main spacecraft panel Z-93 paint has degraded by the expected amount but at a slower rate than anticipated. This data correlates well with existing laboratory data for ultraviolet degradation of the paint. Of the increase of 0.04 in 6000 hours of exposure, half of this change occured in the first 1500 hours. After this period the rate of change decreased. The Z-93 paint on the four off-axis panels (bay 1 and 7 of the spacecraft and SSU) is believed to be following a similar degradation curve.

The degradation of the Z-93 paint on the main SSU panel, however, is far more rapid than that of the spacecraft paint. This increase is about 0.15 in 6000 hours. The rate of change, however, has been continually decreasing for the last 4000 hours of this period. It is believed that the degradation of the Z-93 paint on this panel is due to inadvertent contamination.

The SSU Z-93 paint degradation has caused a panel temperature rise of 20°C. However, the internal bulkhead temperature has increased by only 5°C and there is sufficient margin in the design to tolerate this rise. Hence, the Z-93 paint is functioning satisfactorily in keeping all component temperatures within their respective limits.

References

1. Stevens, N. J., and Smolak, G. R., "Design of the Passive Thermal Control System of the SERT II Satellite," Proceedings of the Symposium on Thermodynamics and Thermophysics of Space Flight, Western Periodical, North Hollywood, 1970, pp. 101-120.
2. Gaski, J. D. and Lewis, D. R., "IBM-7094-11-DCS Computer Program C09945, Chrysler Improved Numerical Differencing Analyzer," TN-AP-66-15, NASA CR-89100, Apr. 30, 1966, Chrysler Corp., New Orleans, La.
3. Toups, K. A., "A General Computer Program for the Determination of Radiant-Interchange Configuration and Form Factors: CONFAC II," SID-65-1043-2, NASA CR-65257, Oct. 1965, North American Aviation, Downey, Calif.
4. Streed, E. R. and Arvesen, J. C., "A Review of the Status of Spacecraft Thermal Control materials," Science of Advanced Materials and Process Engineering Series, Vol. II, Western Periodicals, North Hollywood, 1967, pp. 181-192.
5. Neel, C. B., "The Role of Flight Experiments in the Study of Thermal-Control Coatings for Spacecraft," Thermophysics of Spacecraft and Planetary Bodies: Radiation Properties of Solids and the Electromagnetic Radiation Environment in Space. Vol. 20 of Progress in Astronautics and Aeronautics, G. B. Heller, ed., Academic, New York, 1967, pp. 411-438.
6. Pearson, B., Jr., "Preliminary Results from the Ames Emissivity Experiment on OSO-II," Thermophysics and Temperature Control of Spacecraft and Entry Vehicles. Vol 18 of Progress in Astronautics and Aeronautics, G. B. Heller, ed., Academic, New York, 1966, pp. 459-472.
7. Millard, J. P., "Results of the Thermal Control Coating Experiment on OSO-III," Paper 68-794, June 1968, AIAA, New York, N. Y.
8. Stevens, N. J., "Application of SERT II Thermal Control Coatings," TM X-2155, Jan. 1971, NASA, Cleveland, Ohio.
9. Goetzel, C. G., Rittenhouse, J. B., and Singletary, J. B., eds., Space Materials Handbook, Addison-Wesley, Reading, 1965.
10. Cunnington, G. R., Grammer, J. R., and Smith, F. J., "Emissivity Coatings for Low-Temperature Space Radiators," CR-1420, 1969, NASA, Washington, D.C.
11. Streed, E. R., "The Influence of Temperature on the Stability of Low Solar Absorptance and Thermal Coatings," Proceedings of Conference on Spacecraft Coatings Development, TM X-56167, 1964, NASA, Washington, D.C.
12. Zerlaut, G. A., Carroll, W. F., and Gates, D. W., "Spacecraft Temperature-Control Coatings--Selection, Utilization and Problems Related to the Space Environment," SpaceCraft Systems, Vol. 1, Michal Lunc, ed., Gordon and Breach, New York, 1966, pp. 259-313.
13. Reichard, P. J. and Tiolo, J. J., "Preliminary ATS Thermal Coatings Experiment Flight Data," Proceedings of the Joint Air Force - NASA Thermal Control Working Group, AFML-TR-68-198, AD-841387, Aug. 1968, Air Force Systems Command, Wright-Patterson AFB, Ohio.
14. Lewis, D. W. and Thostesen, T. O., "Mariner-Mars Absorptance Experiment," Thermophysics and Temperature Control of Spacecraft and Entry Vehicles. Vol. 18 of Progress in Astronautics and Aeronautics, G. B. Heller, ed., Academic, New York, 1966, pp. 441-457.

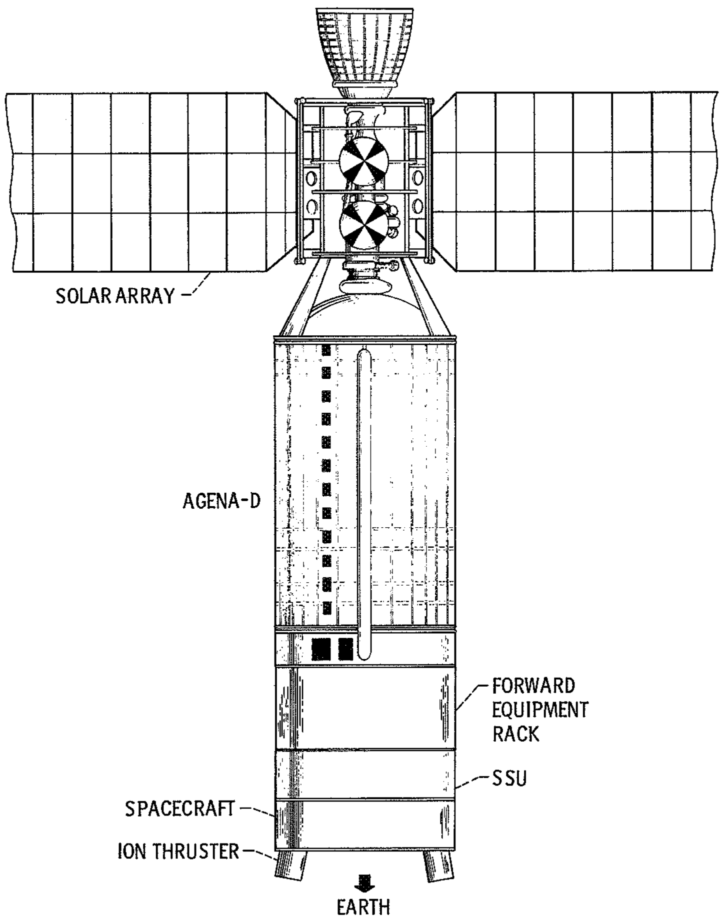


Figure 1. - SERT II satellite.

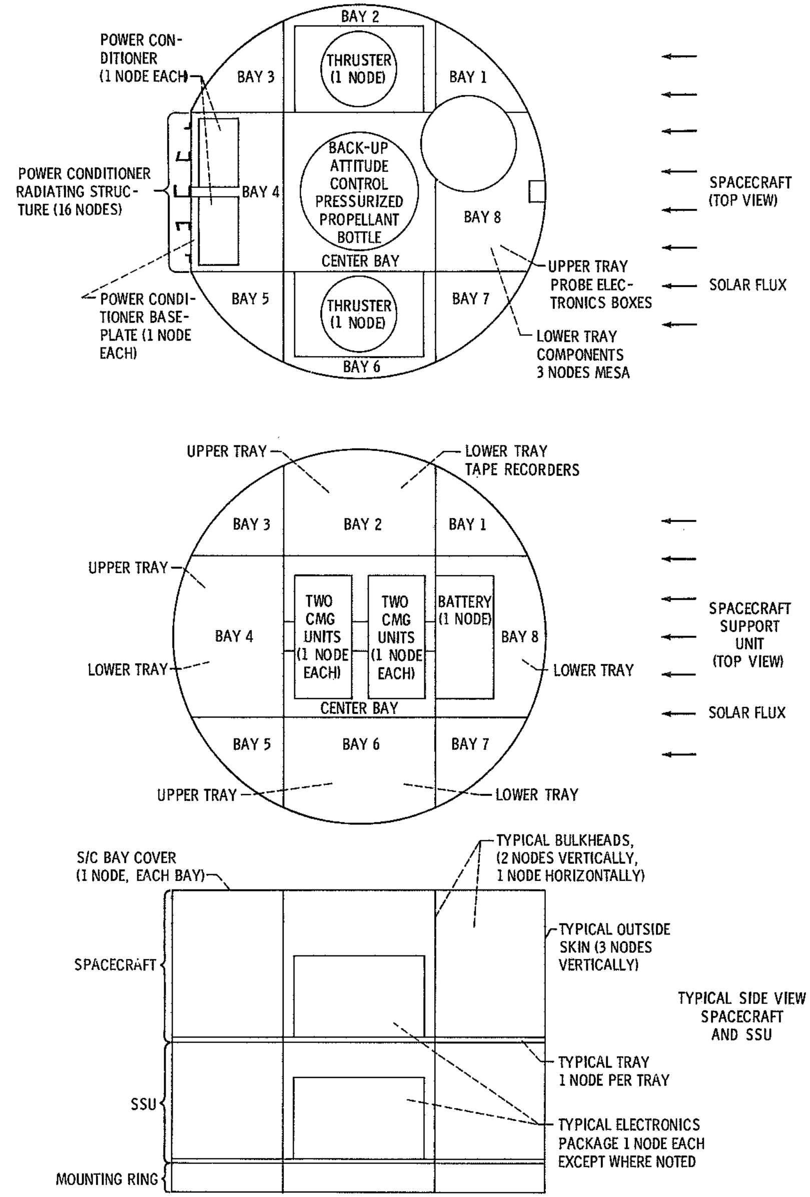
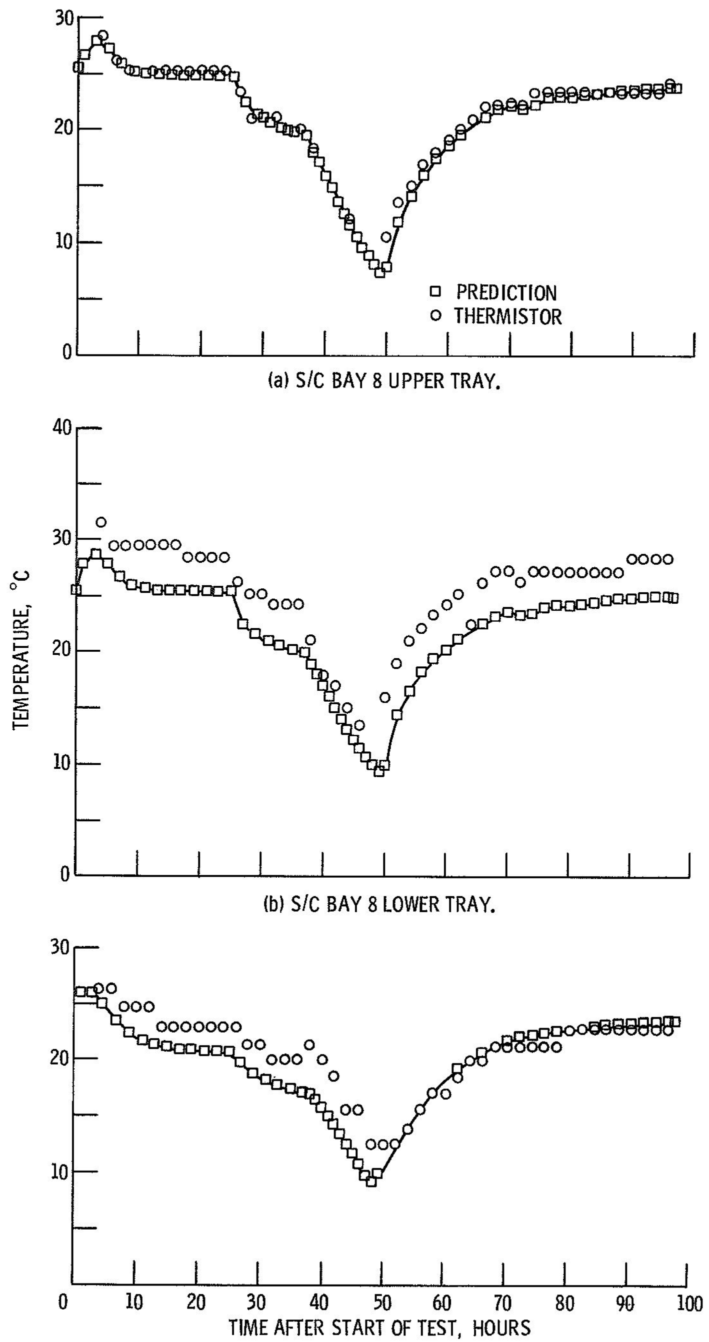


Figure 2. - Nodal layout of SERT II Spacecraft and SSU.



(a) S/C BAY 8 UPPER TRAY.

(b) S/C BAY 8 LOWER TRAY.

(c) SSU BAY 8 BATTERY.

Figure 3. - Comparison of experimental and analytical model temperatures-calibration test.

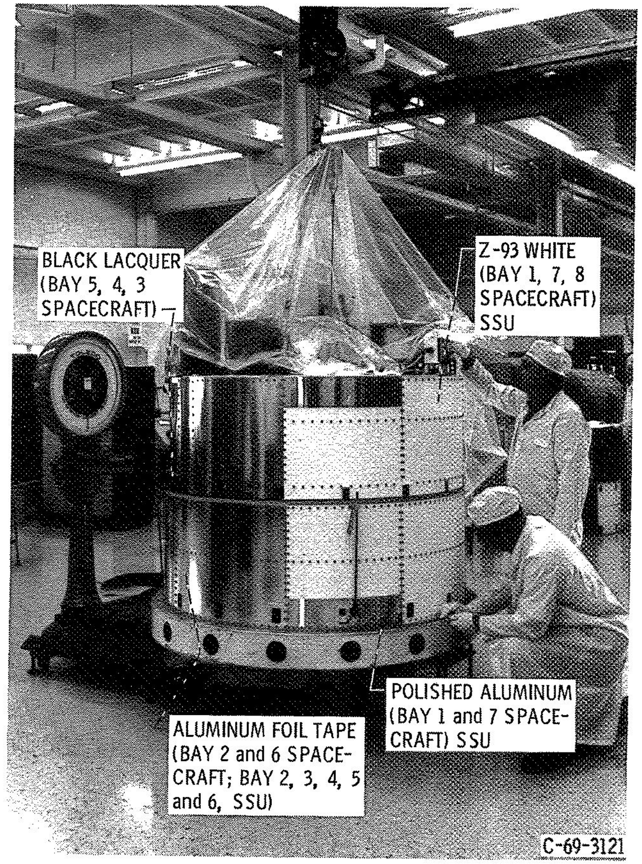


Figure 4. - Side view of SERT II thermal control pattern. Pattern symmetric.

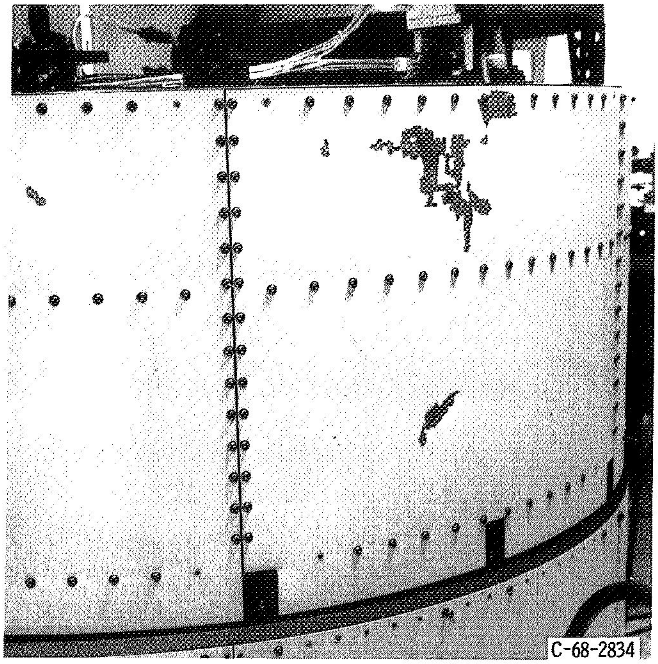


Figure 5. - Self lifting of Z-93 paint from magnesium.

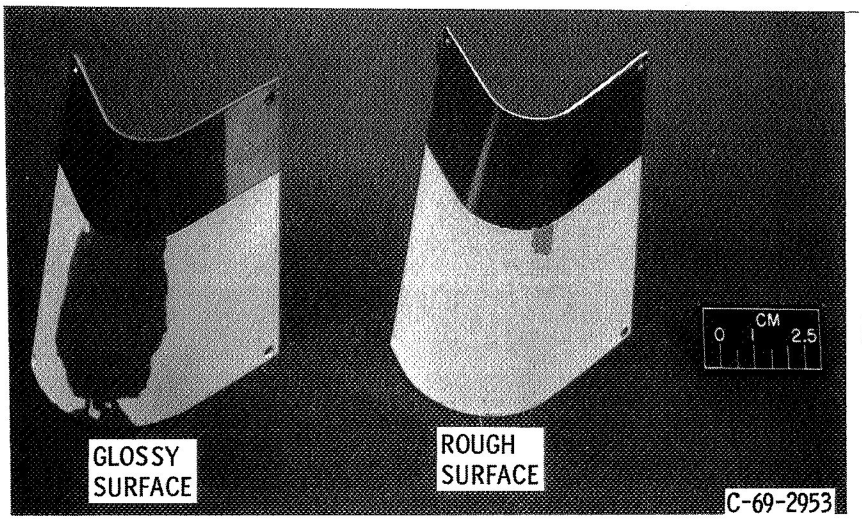


Figure 6. - Bend adhesion test results: effect of surface texture.

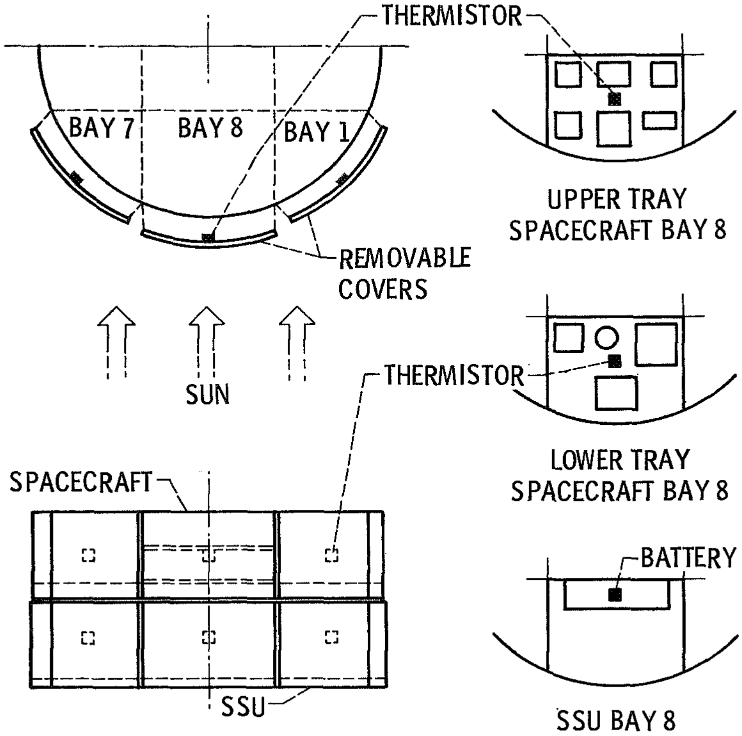


Figure 7. - Flight thermistor locations.

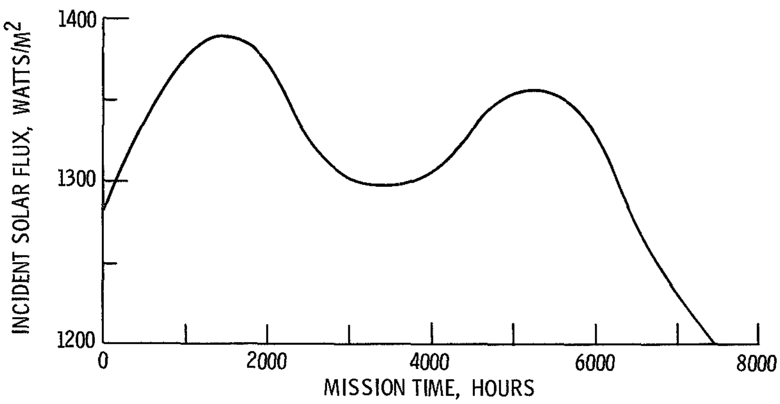


Figure 8. - Solar flux incident on SERT II orbital plane.

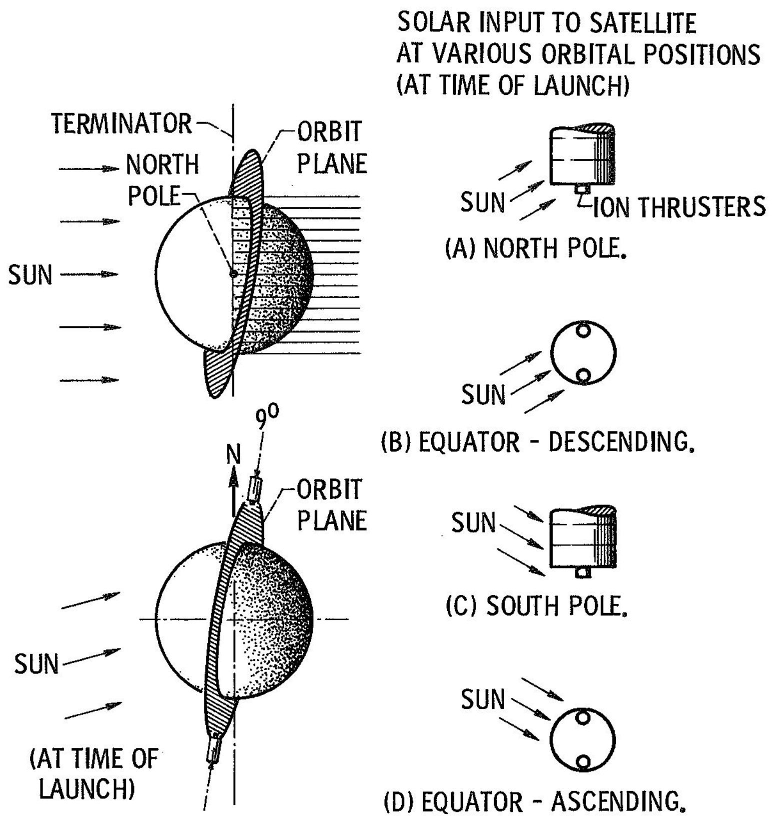


Figure 9. - Variation in solar heating due to orbital position of satellite.

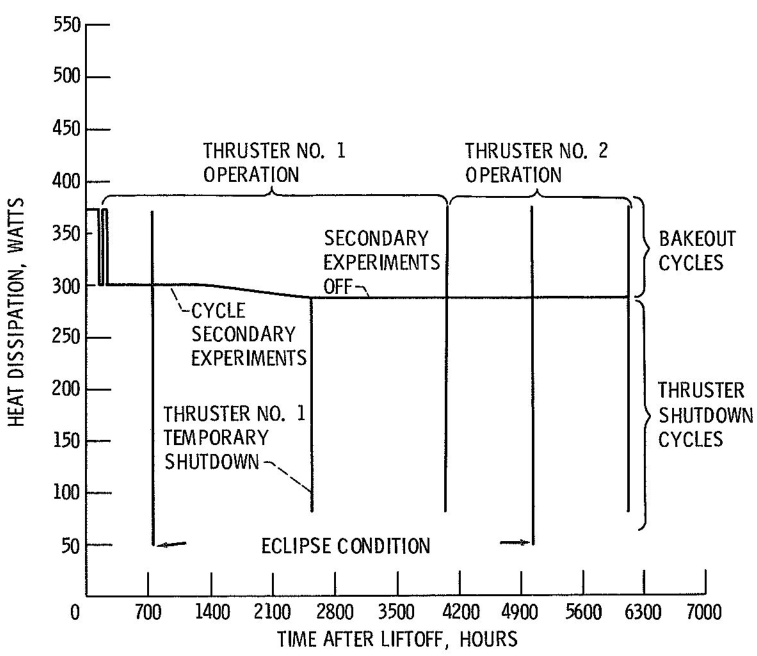


Figure 10. - SERT II internal heat dissipation.

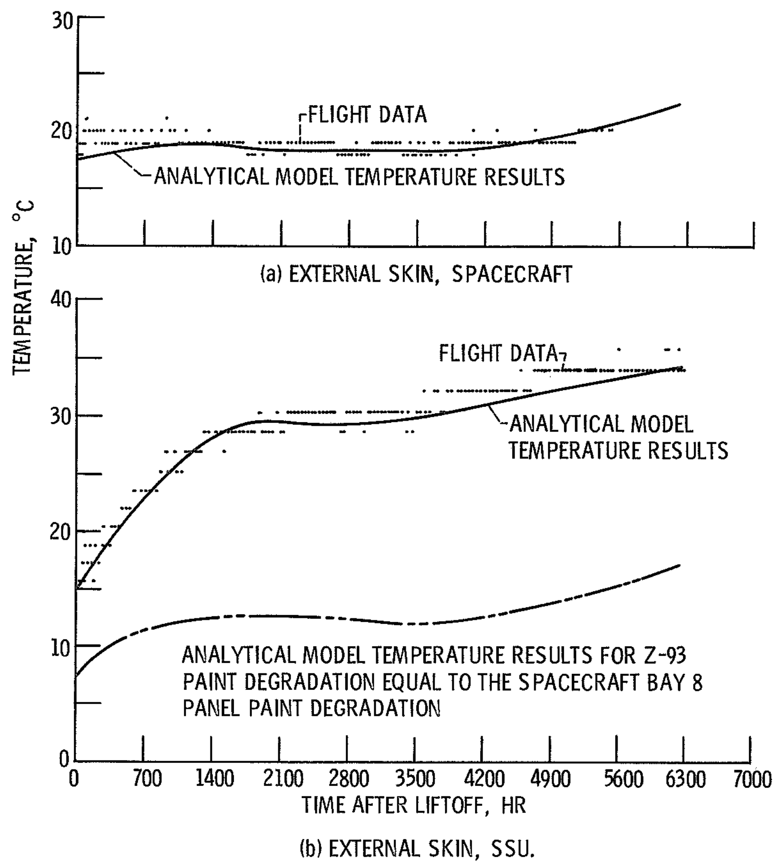


Figure 11. - Comparison of flight temperatures with analytical model results - Bay 8 panel temperatures.

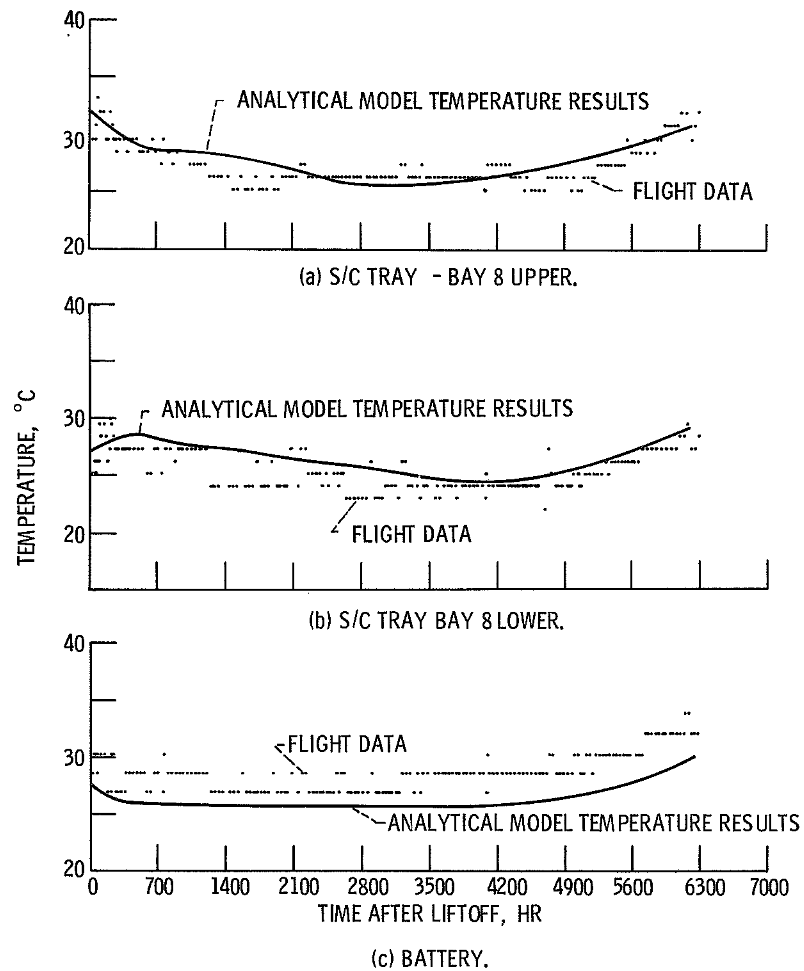


Figure 12. - Comparison of flight temperatures with analytical model results - Bay 8 internal temperatures.

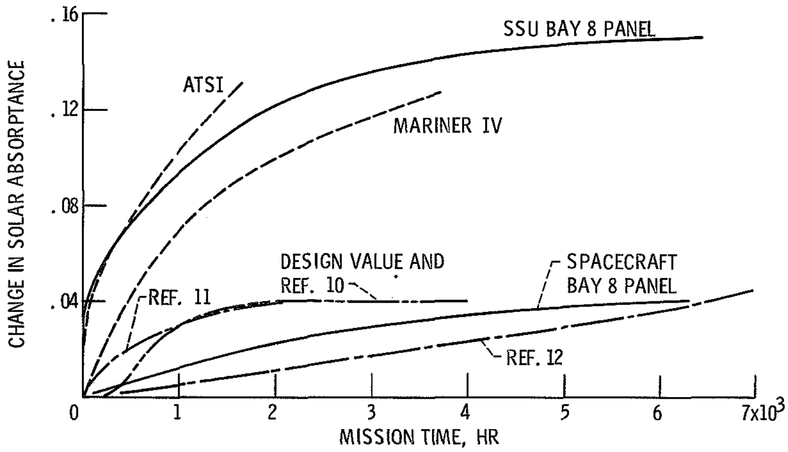
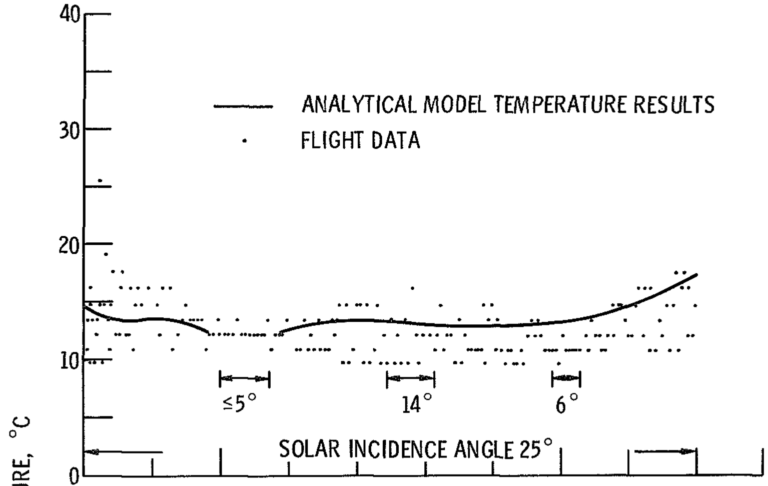
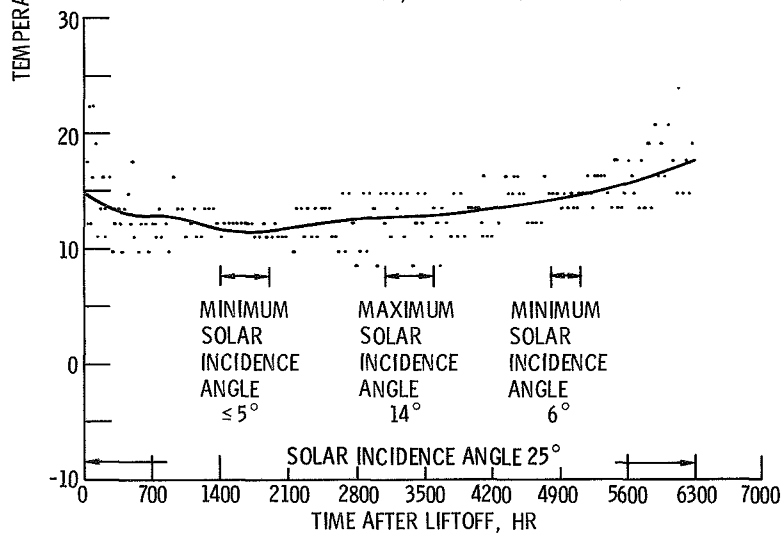


Figure 13. - Change in solar absorptance of SERT II Z-93 paint.

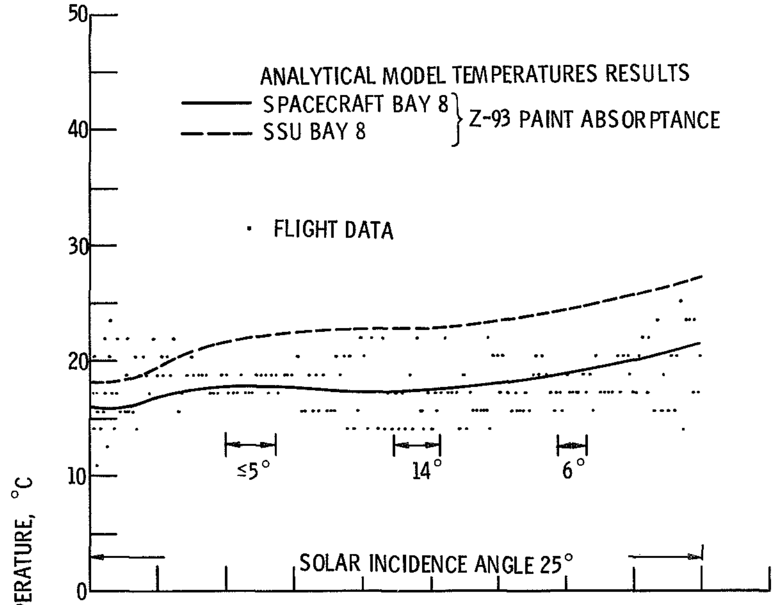


(a) EXTERNAL SKIN, SPACECRAFT - BAY 1.

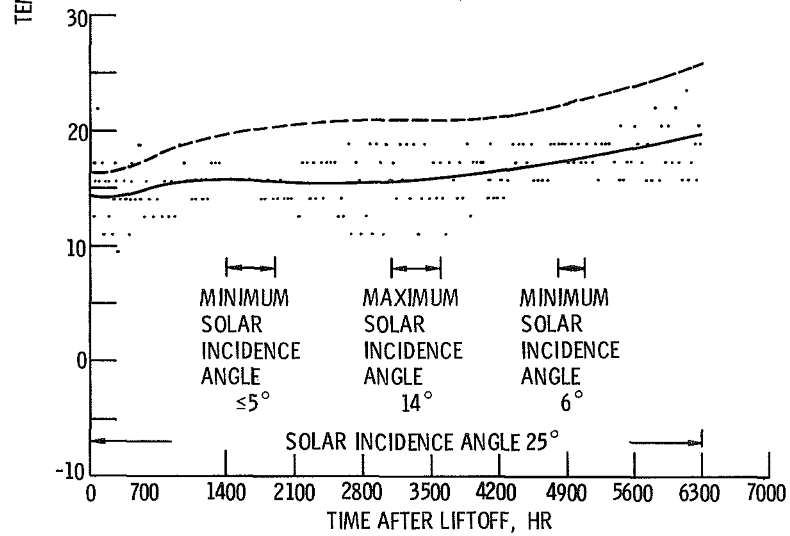


(b) EXTERNAL SKIN, SPACECRAFT - BAY 7.

Figure 14. - Comparison of flight temperatures with analytical model results.



(c) EXTERNAL SKIN, SSU - BAY 1.



(d) EXTERNAL SKIN, SSU - BAY 7.

Figure 14. - Comparison of flight temperatures with analytical model results.