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THERMAL CONSIDERATIONS OF SWIFT XRT RADIATOR AT-35°C OR COLDER IN LOW EARTH ORBIT

Michael K. Choi
NASA Goddard Space Flight Center
Greenbelt, MD 20771

ABSTRACT

The X-Ray Telescope (XRT) is an instrument on the National Aeronautics and Space Administration (NASA) SWIFT spacecraft. The thermoelectric cooler (TEC) for the charge coupled device (CCD) of the XRT requires a radiator temperature of –35°C or colder, and a goal of -55°C to minimize the damage by radiation. The waste heat rejected from the TEC to the radiator is in the 8 W to 20 W range. In the Phase A baseline design, the XRT radiator is mounted to the rear end of the XRT telescope tube and is very close to the bottom closeout of the spacecraft bus. The bottom closeout is multi-layer insulation (MLI) blankets. At sun angles between 90° and 180°, there is direct solar impingement on the bottom closeout. When the rolls ±5°, the XRT radiator is exposed to direct solar radiation. The radiator also has a view factor to the solar arrays. The results of thermal analysis showed that the flight temperature prediction of the radiator exceeds the temperature requirement of –35°C substantially at sun angles from 110° to 180°. A new location on the anti-sun side of the spacecraft is proposed for the radiator. It requires a heat pipe to couple the TEC and the radiator thermally. The results of thermal analysis show that the flight temperature prediction of the proposed radiator meets the temperature requirement at all sun angles.

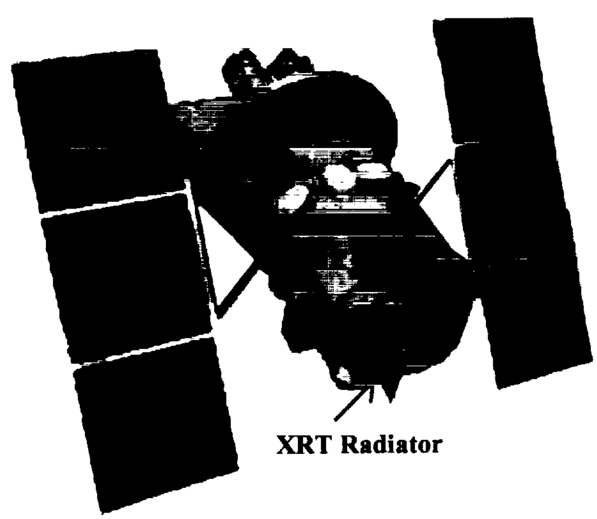
INTRODUCTION

XRT is one of the three telescopes on the SWIFT spacecraft. The SWIFT mission is part of the NASA Medium-Size Explorer (MIDEX) Program, and is managed by Goddard Space Flight Center (GSFC). It is scheduled to launch in 2003. The Swift mission is a first of its kind of multi-wavelength transient observatory for gamma ray burst astronomy. Its mission life is 3 years. SWIFT has a low Earth orbit. The altitude is 600 km, and the inclination is 22°. The XRT instrument is developed jointly by Penn State University and the University of Leicester. The CCD of the XRT is cooled by a TEC, which rejects heat to a radiator. The waste heat is in the 8 W to 20 W range. In the Phase A baseline design, the XRT radiator is mounted to the rear end of the telescope at the bottom of the spacecraft. Both sides of the radiator radiate heat to space. The total area of the two-side radiator is about 6,394 cm^2. Since the radiator is close to the bottom closeout of the spacecraft, it has a good view factor to it. The radiator is made of aluminum, and its mass is 4 kg. Figure 1 shows the Phase A baseline location of the XRT radiator on the SWIFT spacecraft.

THERMAL REQUIREMENT

Presently, the temperature requirement of the XRT CCD is –105°C, and the goal is –115°C to provide sufficient margins to minimize the risk of radiation damage to the CCD. It translates to a temperature requirement of –35°C or colder, and a goal of –55°C for the TEC radiator.

Figure 1. XRT Radiator Location in Phase A.



*Associate Fellow

THERMAL PROBLEM OF XRT RADIATOR IN PHASE A

The spacecraft bottom closeout is MLI blankets. The absorptance and emittance of the spacecraft bottom closeout of the spacecraft bus are 0.39 and 0.62, respectively, in the spacecraft thermal model. An instrument and spacecraft integrated thermal model was developed by the author in the summer of 1999, and a thermal analysis of the XRT radiator was performed. Based on the initial results of thermal analysis, the concerns on the XRT radiator location were that the flight temperature predictions of the XRT radiator are too warm at large sun angles.

The sun angle is the angle between the solar vector and the optical axis of SWIFT. Presently, the sun angle of the SWIFT mission is 45° minimum and 180° maximum. When the sun angle is 180°, the solar vector is perpendicular to the spacecraft bottom closeout. When the sun angle is between 90° and 180°, there is direct solar radiation incident on the spacecraft bottom closeout. Some of the solar flux is reflected from the spacecraft bottom closeout to the XRT radiator. Direct solar radiation absorbed by the spacecraft bottom closeout increases its temperature, and therefore increases the heat radiation to the XRT radiator. Also, the SWIFT spacecraft has a ±5° roll about the X-axis, which is the optical axis, exposing the XRT radiator to direct solar radiation. In addition to the variation of sun angle, the thermal environment varies from sunlight to eclipse. Therefore, the temperature of the XRT radiator varies transiently.

A more detailed thermal analysis was performed in Phase B, and the thermal concerns on the XRT radiator location remain. Table 1 presents the worst hot case flight temperature predictions of the XRT radiator, which has silver teflon as the coating, versus candidate thermal coatings on the spacecraft bottom closeout. The worst case parameters include winter solstice environment, end-of-life thermo-optical properties, and a sun angle of 180°. The temperature is maximum at orbit noon and is minimum at the end of the eclipse. Table 2 presents the results of using Z-93P white paint as the coating on the XRT radiator. Table 3 presents the worst hot case flight temperature predictions of the spacecraft bottom closeout versus its thermal coating. The flight temperature predictions of the XRT radiator are too warm in the sunlight, even if the coating on the radiator is silver teflon or Z-93P white paint. The design values of solar constant, albedo and Earth emitted infrared (IR) radiation have a significant impact on the flight temperature predictions the XRT radiator. The GSFC design values[1] in Table 4 are used in the thermal analysis. Tables 5 and 6 compare the GSFC albedo and Earth IR values to those used at other NASA centers as reported by F. A. Costello in 1995[2], and to those recommended by F. A. Costello based on his analysis of the Earth Radiation Balance Experiment (ERBE) flight data.[2]

Figure 2 shows the effect of the heat rejected from the TEC to the XRT radiator when the sun angle is 180°. Silver teflon is the thermal coating on both the XRT radiator and spacecraft bottom closeout. The effect is not very significant because the heat rejection from the TEC is much less than the environmental heat flux absorbed by the radiator and the heat radiation from the spacecraft bottom closeout to the radiator.

Figure 3 presents the hot case flight temperature predictions of the XRT radiator versus the sun angle. The temperature is maximum at orbit noon and is minimum at the end of the eclipse. The radiator temperature is too warm at large sun angles.

Table 1. Hot Case Flight Temperature Prediction of Phase A XRT Radiator with Silver Teflon As Coating at 180° Sun Angle.

α/ε of Spacecraft Bottom	Orbit Noon (°C)	End of Eclipse (°C)
.12/.026 (VDA)	-19	-40
.20/.21 (Ag-Al_2O_3 Overcoating)	-12	-37
.20/.76 (Silver Teflon)	-13	-38
.39/.62	-1	-33
.87/.77 (Black Kapton)	+29	-26

Table 2. Hot Case Flight Temperature Prediction of Phase A XRT Radiator with Z-93P White Paint at 180° Sun Angle.

α/ε of Spacecraft Bottom	Orbit Noon (°C)	End of Eclipse (°C)
.12/.026 (VDA)	-15	-44
.20/.21 (Ag-Al_2O_3 Overcoating)	-4	-37
.20/.76 (Silver Teflon)	-5	-37
.39/.62	+6	-35
.87/.77 (Black Kapton)	+31	-29

Table 3. Hot Case Flight Temperature Prediction of Spacecraft Bottom Closeout at 180° Sun Angle with Silver Teflon or Z-93P as XRT Radiator Coating.

α/ε of Spacecraft Bottom	Orbit Noon (°C)	End of Eclipse (°C)
0.12/0.026 (VDA)	316	-61
.20/.21 (Ag-Al_2O_3 Overcoating)	139	-61
.20/.76 (Silver Teflon)	33	-61
.39/.62*	95	-61
.87/.77 (Black Kapton)	146	-61

Table 4. GSFC Design Values of Environmental Constants.

	Minimum	Maximum
Solar Constant (W/m^2)	1287	1419
Albedo	0.25	0.35
Earth Emitted Infrared Radiation (W/m^2)	208	265

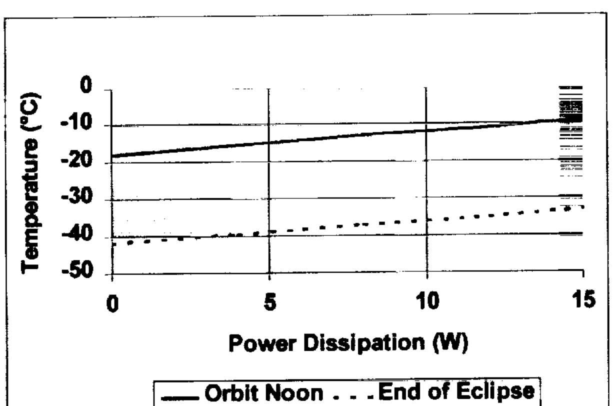
Table 5. Design Values of Albedo (Reported by F. A. Costello, 1995[2]).

	Averaging Time (Hr)	Set	Min.	Max.
NASA-MSFC	1.5	Max/Min	0.15	0.39
NASA-MSFC	1.5	1-99%	0.18	0.34
NASA-JSC	4.0	Max/Min	0.20	0.31
NASA-GSFC	Long	Max/Min	0.25	0.35
F. A. Costello	Long	3-σ	0.27	0.39
F. A. Costello	Long	Max/Min	0.30	0.37

Table 6. Design Values of Earth Emitted Infrared Radiation (W/m^2) (Reported by F. A. Costello, 1995[2]).

	Averaging Time (Hr)	Set	Min.	Max.
NASA-MSFC	1.5	Max/Min	198	276
NASA-MSFC	1.5	1-99%	202	257
NASA-JSC	4.0	Max/Min	232	280
NASA-GSFC	Long	Max/Min	208	265
F. A. Costello	Long	3-σ	197	241
F. A. Costello	Long	Max/Min	210	232

Figure 2. Effect of Heat Rejected by TEC to XRT Radiator in Phase A Location.



EFFECT OF SOLAR ARRAYS ON BASELINE XRT RADIATOR

In the Phase A thermal analysis of the XRT radiator, the solar arrays were neglected. Also, the heat radiation from the spacecraft interior through the spacecraft bottom closeout MLI was neglected.

The Swift spacecraft has two solar array wings. Each wing has a solar array drive and three articulating panels. The sun angle of the Swift mission is 45° to 180°. Figure 4, which is in the Thermal Synthesizer System (TSS) format, shows the position of the solar arrays when the sun angle is 45° and the baseline location of the XRT radiator in the thermal model. The following assumptions are made in the thermal analysis. The heat rejection from the XRT TEC to the radiator is 8 W. Both sides of the baseline radiator radiate heat to space. The total area of the two-side radiator is 6,394 cm^2. The mass of the radiator is 4 kg. The coating on both the radiator and spacecraft bottom closeout MLI is 0.254 mm (10-mil) thick silver teflon. The temperature of the spacecraft interior is assumed to be 30°C, and the MLI effective emittance is assumed to be 0.03.

Figure 3. Hot Case Flight Temperature Prediction of XRT Radiator (Silver Teflon) Versus Sun Angle with Solar Arrays Neglected.

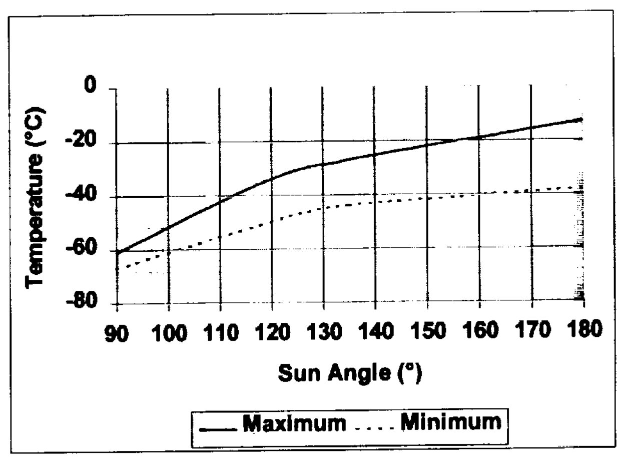
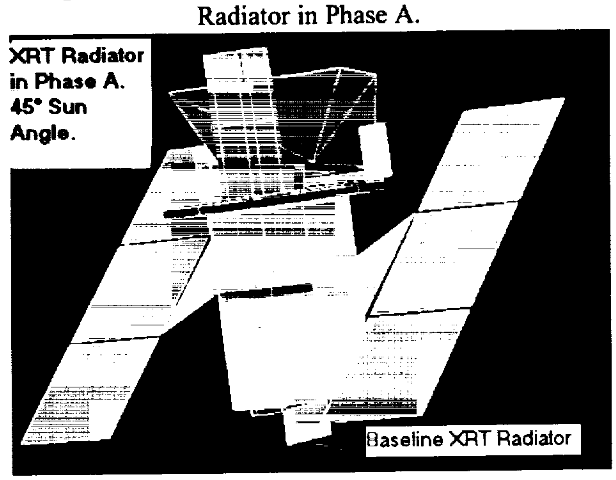


Figure 4. Solar Arrays at 45° Sun Angle and XRT Radiator in Phase A.



Figures 5 through 10 present the results of the thermal analysis in the hot case. Winter solstice environment and end-of-life (EOL) thermo-optical properties are used in the hot case thermal analysis. The EOL absorptance and emittance of 0.254 mm (10-mil) silver teflon are assumed to be 0.25 and 0.827, respectively. Figure 5 presents the flight temperature prediction of the XRT radiator versus the sun angle. The thermal effect of the solar arrays and heat radiation through the spacecraft bottom closeout MLI on the XRT radiator temperature can also be seen in Figure 5. The flight temperature prediction of the radiator at orbit noon exceeds the temperature requirement of –35°C at sun angles between 110° and 180°. To maintain the XRT radiator temperature below –35°C, a time constraint on the sun angles is required in flight.

Figure 6 presents the radiation coupling between the XRT radiator and solar arrays versus the sun angle. The radiation coupling is largest when the sun angle is 45° and 135°, and smallest when the sun angle is 180°. Figure 7 presents the peak heat radiation, which occurs at orbit noon, from the solar arrays to the XRT radiator versus the sun angle. Figure 8 presents the orbital average environmental heat flux absorbed by the XRT radiator versus the sun angle. It includes solar, albedo and Earth IR radiation. When the sun angle is between 45° and 90°, there is no direct solar flux incident on the radiator. When the sun angle is 90° or larger, the direct solar flux incident on the radiator increases as the sun angle increases. Figure 9 presents the peak heat radiation from the spacecraft bottom closeout MLI to the XRT radiator versus the sun angle. When the sun angle exceeds 90°, the spacecraft bottom closeout MLI temperature increases as the sun angle increases, and therefore the heat radiation from the MLI to the radiator increases. When the sun angle is less than 90°, albedo and Earth infrared radiation are the only environmental heat fluxes incident on the bottom closeout. Figure 10 presents the flight temperature prediction of the solar arrays.

Figure 5. Hot Case Flight Temperature Predictions of XRT Radiator in Phase A.

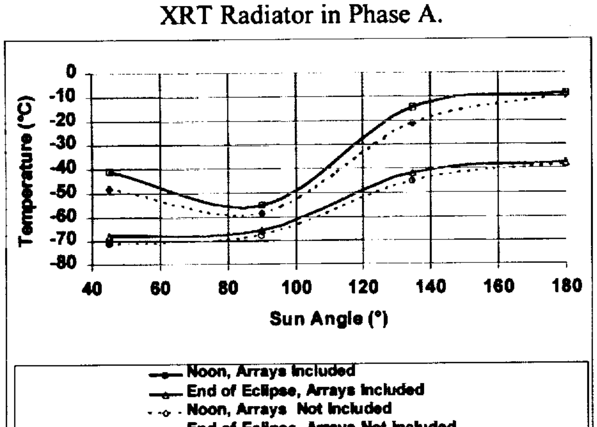


Figure 6. Radiation Couplings between Solar Arrays and XRT Radiator in Phase A.

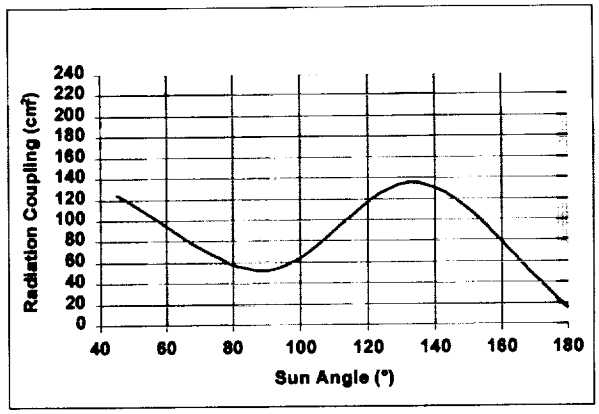


Figure 7. Peak Radiation from Solar Arrays to XRT Radiator in Phase A.

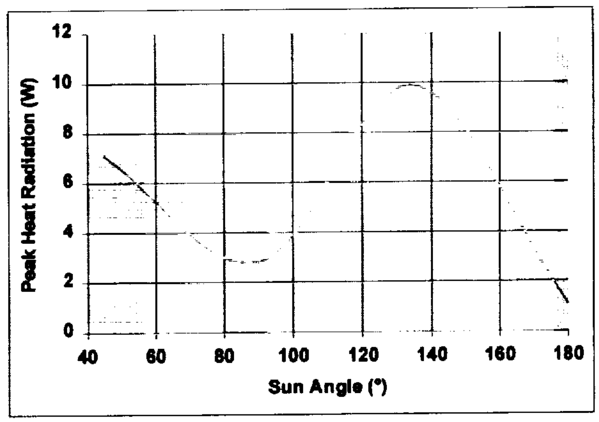


Figure 8. Orbital Average Environmental Heat Flux Absorbed by XRT Radiator in Phase A Location.

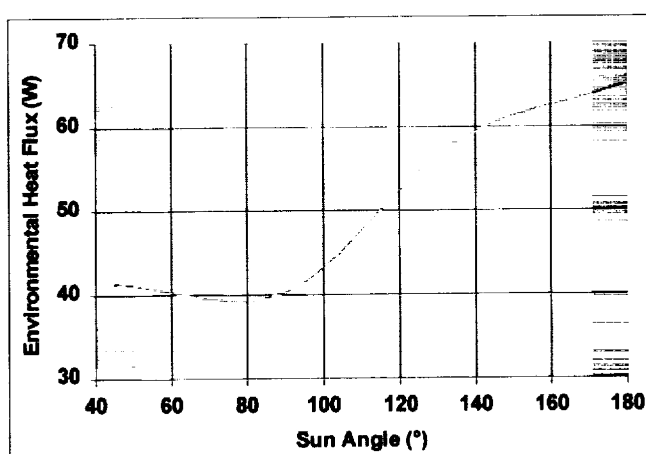


Figure 9. Peak Heat Radiation From Spacecraft Bottom Closeout To XRT Radiator in Phase A.

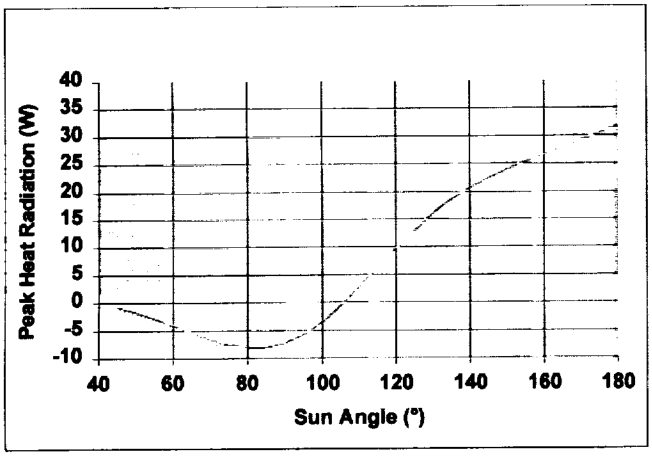
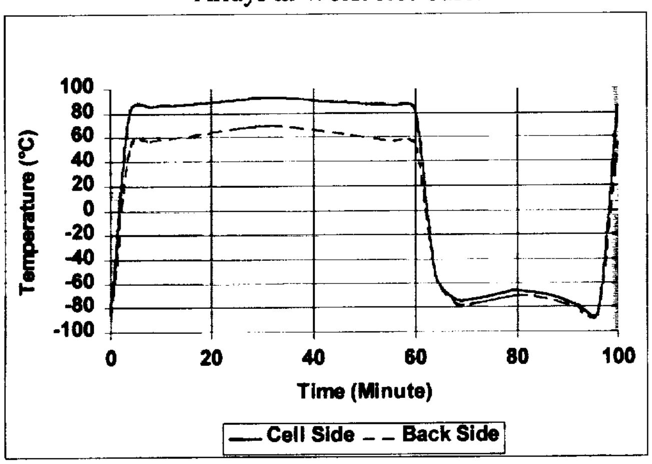


Figure 10. Flight Temperature Prediction of Solar Arrays in Worst Hot Case.



PROPOSED XRT RADIATOR LOCATION

Presently in the Phase B study, the thermal concept of relocating the XRT radiator to the anti-sun of the spacecraft bus is proposed. In this design concept, a constant conductance heat heat pipe (CCHP) transports the waste heat from the TEC to the radiator. Also, separate CCHPs minimize the temperature gradient on the radiator. The results of the preliminary thermal analysis show that the flight temperature predictions of the radiator in the new location are much colder than that of the Phase A baseline location at high sun angles. Figure 11 shows the new radiator location and the heat pipe that thermally couples the TEC and radiator.

EFFECT OF SOLAR ARRAYS ON PROPOSED XRT RADIATOR LOCATION

Figure 12 shows the position of the solar arrays when the sun angle is 45° and the proposed location of the XRT radiator on the anti-sun side of the spacecraft in the thermal model. The coating on the radiator is also 0.254 mm (10-mil) thick silver teflon. The radiator is assumed to be isothermal. CCHPs on the radiator minimize the temperature gradient. The radiator is mounted to the spacecraft bus by titanium flexures and G-10 washers, so that it is thermally isolated from the spacecraft. Presently, the conduction coupling between the radiator and spacecraft is assumed to be 0.05 W/°C. The spacecraft interface temperature is assumed to be 20°C in the hot case. The backside of the radiator is insulated with MLI blankets. The heat pipe from the TEC to the radiator is approximately 2 m long. By maximizing the thermal interface conductance between the heat pipe and TEC, and between the heat pipe and radiator, the temperature gradient between the TEC and radiator is very small.

Figure 11. Proposed XRT Radiator Location.

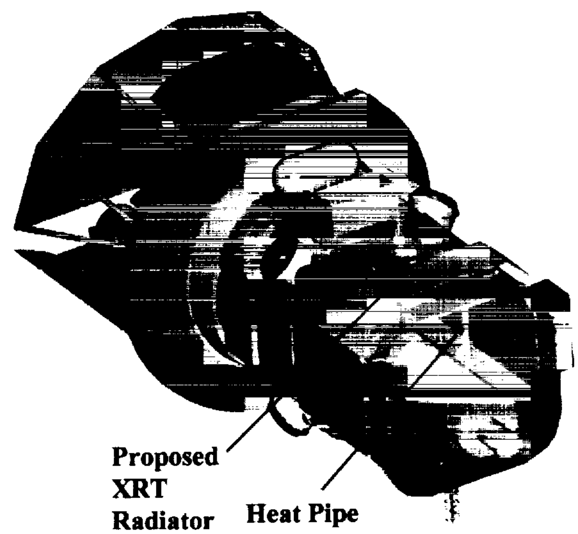
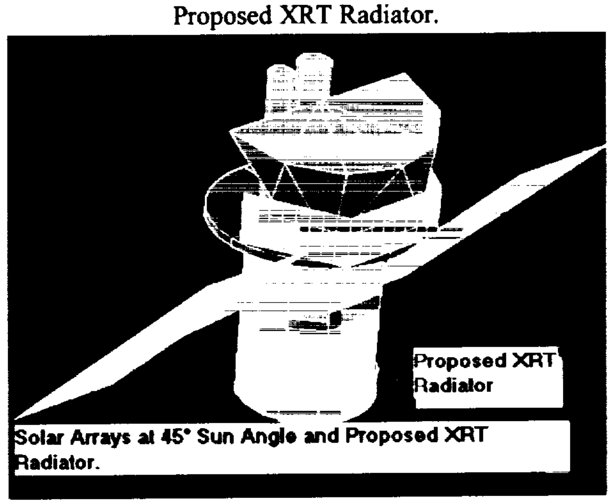


Figure 12. Solar Arrays at 45° Sun Angle and Proposed XRT Radiator.



Figures 13 through 16 present the results of the thermal analysis in the hot case. In the current thermal analysis, the radiator forms a 132° sector around the spacecraft, and the height is 33 cm. The radiator area is 6,185 cm^2. The mass of the radiator is 4.4 kg. Figure 13 presents the flight temperature prediction of the XRT radiator versus the sun angle. The thermal effect of the solar arrays can also be seen in Figure 13. The flight temperature prediction of the radiator at orbit noon satisfies the temperature requirement of –35°C at all sun angles. A colder radiator temperature can be achieved by increasing the radiator area. For example, increasing the radiator height from 33 cm to 45 cm reduces the temperature at orbit noon from –35°C to –37°C at a 90° sun angle. Also, the temperature at orbit noon can be decreased by increasing the thermal mass of the radiator. The mass of the radiator includes the CCHPs for isothermalization of the radiator, and the heat pipe for transport heat from the TEC to the radiator. For example, increasing the mass of the radiator from 4.4 kg to 8.8 kg decreases the radiator temperature at orbit noon from –35°C to –42°C for a 90° sun angle. The proposed XRT radiator location on the anti-sun side is thermally more favorable than the Phase A baseline location.

Figure 14 presents the radiation coupling between the XRT radiator and solar arrays versus the sun angle. The radiation coupling is largest when the sun angle is 180°, and smallest when the sun angle is 90°. Figure 15 presents the peak heat radiation, which occurs at noon, from the solar arrays to the XRT radiator versus the sun angle. Figure 16 presents the orbital average environmental heat flux absorbed by the XRT radiator versus the sun angle. It includes albedo and Earth IR radiation. There is no direct solar flux incident on the radiator for all sun angles. Also, there is no heat radiation between the radiator the spacecraft bottom closeout.

Figure 13. Hot Case Flight Temperature Prediction of Proposed XRT Radiator versus Sun Angle.

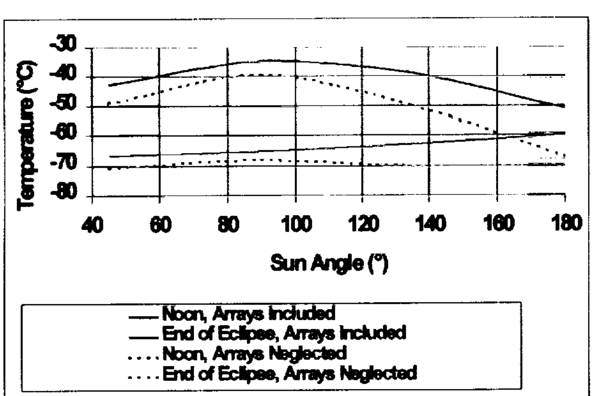


Figure 14. Radiation Couplings between Solar Arrays and Proposed XRT Radiator.

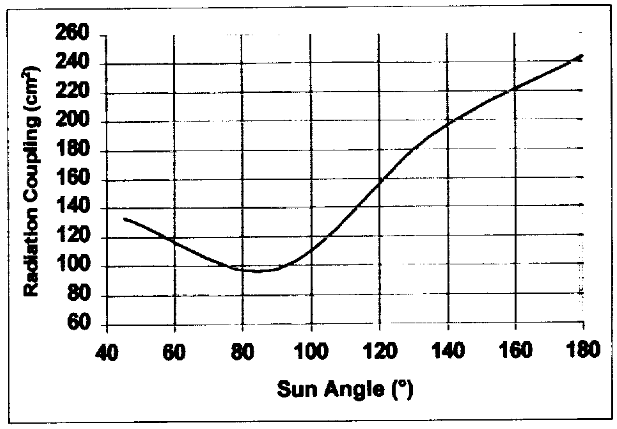


Figure 15. Peak Radiation from Solar Arrays to Proposed XRT Radiator.

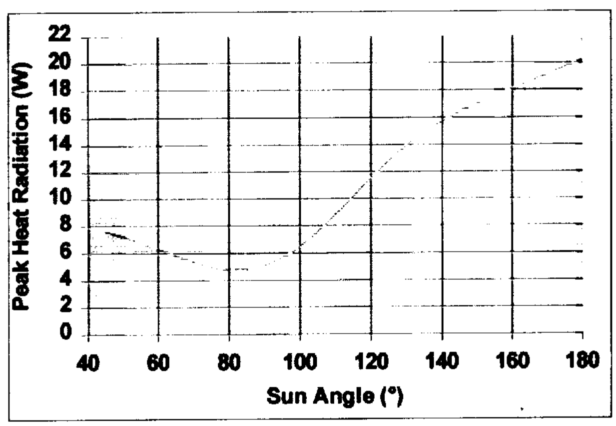
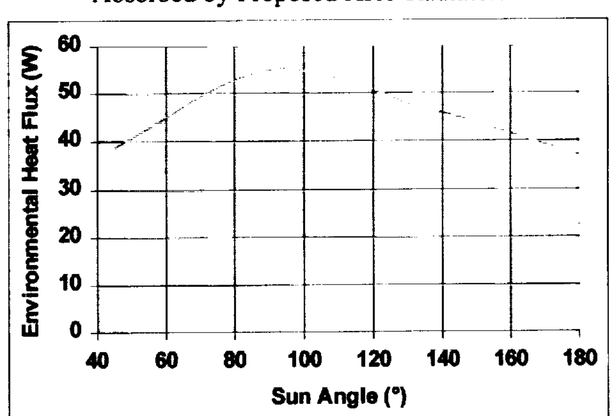


Figure 16. Orbital Average Environmental Heat Flux Absorbed by Proposed XRT Radiator.



OPTIMIZATION OF PROPOSED XRT RADIATOR

Albedo flux and Earth emitted IR radiation are the only environment heat fluxes that affect the temperature of the XRT radiator at the proposed location on the anti-sun side of the spacecraft. These environmental heat fluxes vary from day to night, from summer solstice to winter solstice, and from one sun angle to another. Also, degradation of the thermal coating on the radiator has an impact on the radiator temperature.

The thermal energy balance on the XRT radiator is

$$\varepsilon A F(t)\sigma\theta^4(t) + mc_p(\theta)\, d\theta(t)/dt = Q_{TEC} + Q_{Albedo}(A,t) + Q_{EIR}(A,t) + Q_{SC}(\theta) + Q_{SA}(t) \quad (1)$$

where ε = hemispherical emittance of thermal coating on radiator,

A = radiator area,

$F(t)$ = view factor of radiator to space, which is a function of time because position of solar arrays changes with sun angle,

σ = Stefan-Boltzmann constant,

$\theta(t)$ = radiator temperature which is a function of time,

m = mass of radiator and CCHPs,

$c_p(\theta)$ = specific heat, which is a function of temperature,

Q_{TEC} = power dissipation of TEC,

$Q_{Albedo}(A,t)$ = Albedo flux absorbed, which is a function of radiator area and time,

$Q_{EIR}(A,t)$ = Earth emitted IR radiation absorbed, which is a function of radiator area and time,

$Q_{sc}(\theta)$ = heat conduction from spacecraft mounting interface to radiator, which is a function of the radiator temperature, and

$Q_{SA}(t)$ = heat radiation from solar arrays to radiator, which is a function of time because position of solar arrays changes with sun angle.

From the above equation, for a given Q_{TEC}, the radiator temperature is dependent on A and m. Therefore, they must be optimized. Of course, m needs to be within the mass budget.

Degradation of the thermal coating on the radiator has a significant impact on the radiator temperature. When a coating degrades, its solar absorptance increases, and the albedo flux absorbed increases. One way to minimize the effect of albedo flux is to select a thermal coating that has a very low solar absorptance at the beginning of life (BOL), and a small degradation over a 3-year mission life for the radiator. AZ-Tek's AZW/LA-II white paint is likely the best candidate thermal coating for the XRT radiator, based on its thermo-optical properties. The Thermal Coatings Committee at GSFC recommends the following thermo-optical properties for this coating in the Swift mission, when it is not exposed to direct solar radiation. The solar absorptance is 0.08 and hemispherical emittance is 0.91 at BOL. The solar absorptance is 0.12 and hemispherical emittance is 0.90 at the end of life (EOL).[3] Its high emittance provides a high radiation coupling to space. It also has a low degradation in the solar absorptance. The flight data of the AZW/LA-II white paint shows that initially the solar absorptance was 0.088, and after 48 days in space flight, it increased to 0.095, and after 197 days, it increased to 0.103.[4] It shows that the degradation of this paint in space is very small. Preliminary results of an ongoing ultraviolet exposure test at GSFC show that the paint is stable. Also, the AZW/LA-II white paint will be flying on the GSFC calorimeter on the EO-1 spacecraft, which is scheduled to launch later this year. EO-1 is an Earth orbiting spacecraft, and has an altitude of 705 km. More testing of this paint will be performed at GSFC, including measurement of the emittance at temperatures ranging from room temperature to 40 K (-233°C).

The XRT radiator and CCHPs are made of aluminum. Figure 17 presents the relationship between the specific heat of pure aluminum and temperature in the –94°C to 27°C range.[5] When the temperature decreases, the specific heat of aluminum decreases. At –70°C, the specific heat is 89% of that at room temperature. Because the thermal capacitance of the XRT radiator is the product of its mass and specific heat, and its temperature is significantly colder than room temperature, the data in Figure 17 is included in the thermal mathematical model. Note that the XRT radiator and CCHPs are made of an aluminum alloy, and not pure aluminum. The specific heat is adjusted in Figure 17 accordingly.

Figure 18 presents the worst hot case flight temperature predictions of the proposed XRT radiator versus radiator area. The effect of the mass of the radiator is also shown. The mass includes the CCHPs. The parameters stacked in the worst hot case are 90° sun angle, winter solstice environment, and EOL thermo-optical properties. The mass selected must be within the mass budget. For a radiator mass of 8.8 kg to 13.2 kg, the radiator area should be approximately 6,185 cm^2. If the mass is 8.8 kg and the area is 6,185 cm^2, the radiator temperature at orbit noon is –46°C. If the mass is 13.2 kg and the area is 6,185 cm^2, the radiator temperature at orbit noon is –49°C.

Figure 19 presents the hot case flight temperature prediction of the XRT radiator versus the sun angle. The thermal effect of the solar arrays is included. Note that in the cold case, the flight temperature prediction of the XRT radiator is significantly colder. It is important to select a proper working fluid for the CCHPs to ensure adequate heat transport capacity and no freezing.

Figure 17. Specific Heat of Pure Aluminum versus Temperature.

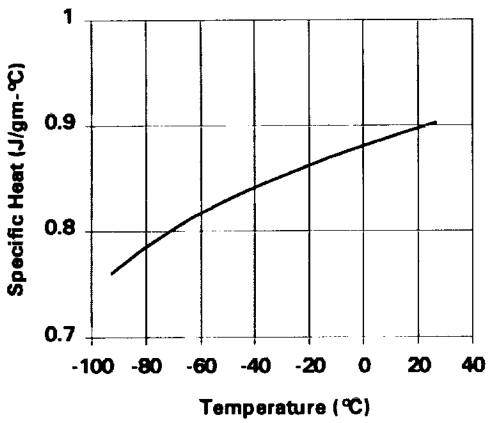


Figure 18. Hot Case Flight Temperature Prediction of Proposed XRT Radiator vs. Radiator Area at 90° Sun Angle.

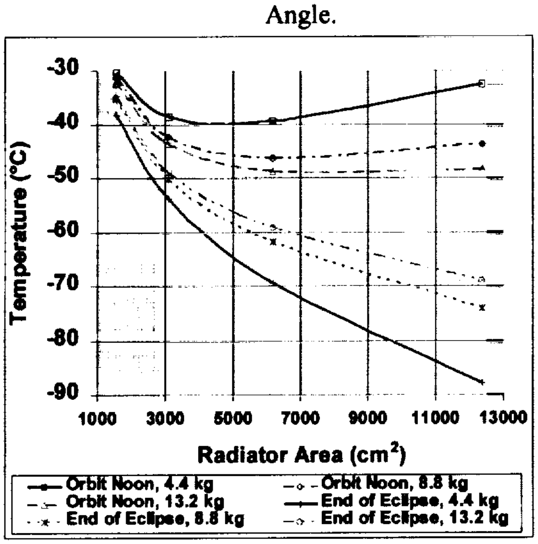
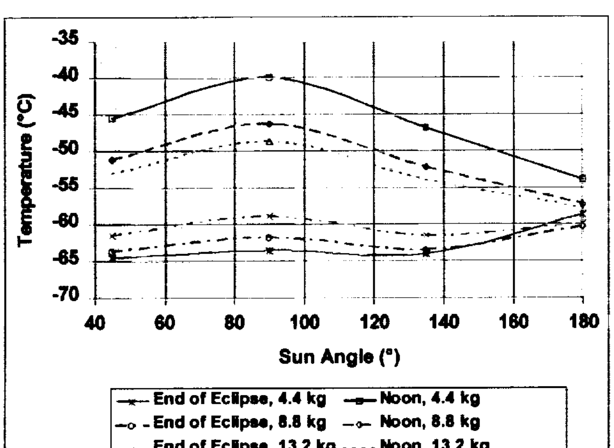


Figure 19. Hot Case Flight Temperature Prediction of Proposed XRT Radiator versus Sun Angle.



EFFECT OF TEC POWER DISSIPATION ON XRT RADIATOR OPTIMIZATION

Worst Case of 20 W

The power dissipation of the TEC was assumed to be 8 W in the thermal analysis above. But, the worst case power dissipation of the TEC could be as much as 20 W.[6] Figure 20 presents the worst hot case flight temperature predictions of the proposed XRT radiator versus the radiator area. The power dissipation of the XRT TEC is taken as 20 W. The effect of the mass of the radiator is also shown in Figure 20. The thermal effect of the solar arrays is included. From Figure 20, the optimum radiator areas for radiator masses of 4.4 kg, 8.8 kg, 13.2 kg, and 17.6 kg are 7,000 cm², 9,000 cm², 11,000 cm² and 13,000 cm², respectively. As mentioned earlier, the radiator mass includes all the CCHPs. For these optimum radiator areas, the worst hot case flight temperature predictions of the radiator at orbit noon are –31.5°C, -39.7°C, -43.8°C, and –46.2°C, respectively. Figure 21 presents the hot case flight temperature prediction of the XRT radiator versus the sun angle, using the optimum radiator areas for the four radiator masses. Note that the mass budget is 10 kg.

Sensitivity Study

Figure 22 presents the optimum radiator area versus the TEC power dissipation for the four radiator masses. The optimum area increases as the TEC power dissipation increases. Figure 23 presents the worst hot case flight temperature predictions of the XRT radiator at orbit noon versus the TEC power dissipation, when the radiator area is optimized.

SURVIVAL TEMPERATURE

In the safehold mode, the instruments are turned off, and there is no heat dissipation at the TEC. The worst cold case flight temperature prediction of the XRT radiator and CCHPs in the safehold mode is -90°C. Ammonia is not acceptable for the working fluid of the CCHPs because its freezing point is -78°C. The heat dissipation of the TEC is 8 W to 20 W, and is low. Propylene, which has a freezing point of -185°C, is possibly the best candidate working fluid. Ethane, which has a freezing point of -183°C, is another candidate.

Figure 20. Hot Case Flight Temperature Prediction of Proposed XRT Radiator vs. Radiator Area at 90° Sun Angle with 20 W TEC Power Dissipation.

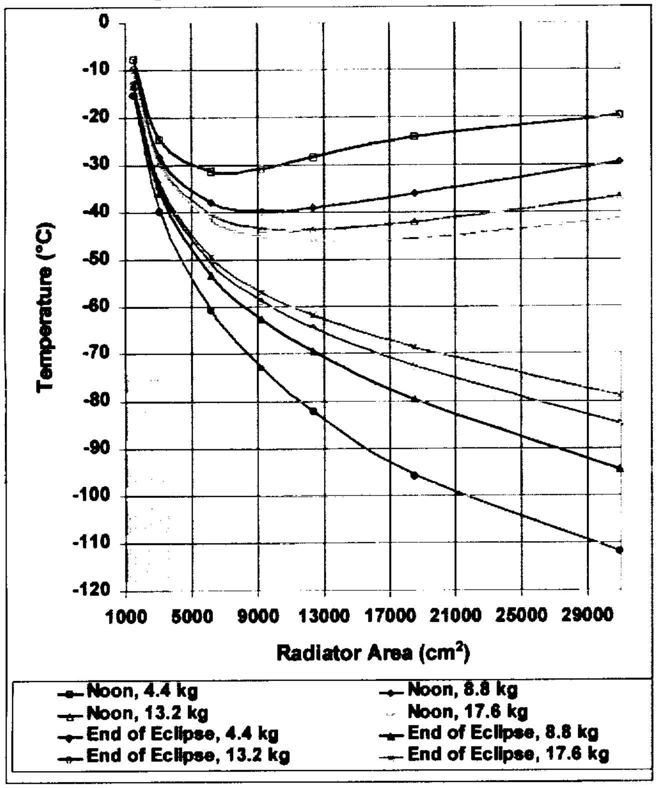


Figure 21. Hot Case Flight Temperature Prediction of Proposed XRT Radiator with Optimum Area versus Sun Angle.

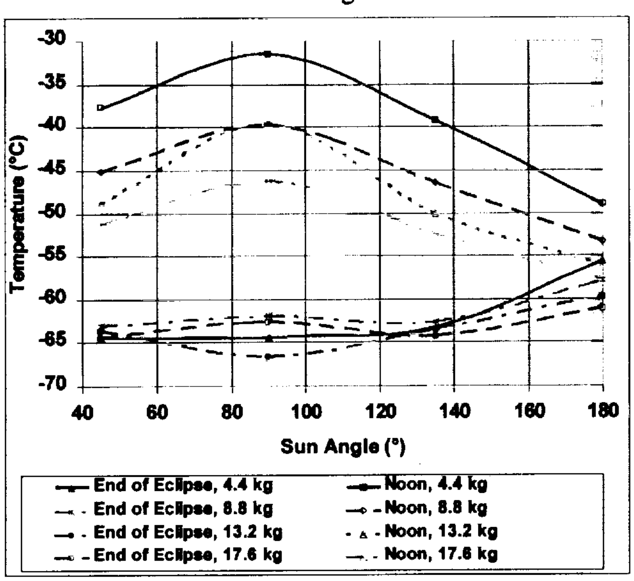


Figure 22. Optimum Area of Proposed XRT Radiator versus TEC Power Dissipation.

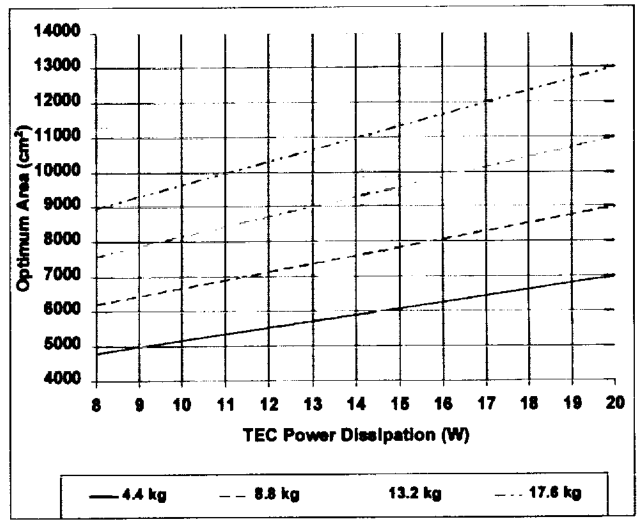
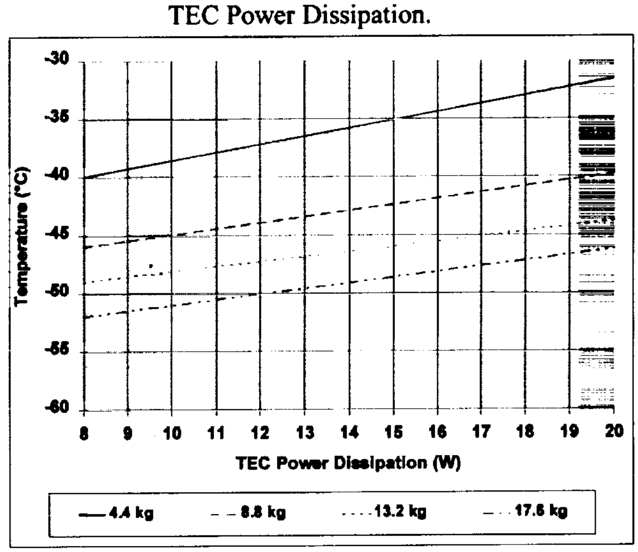


Figure 23. Hot Case Flight Temperature Prediction of Proposed XRT Radiator with Optimum Area versus TEC Power Dissipation.



SUMMARY AND CONCLUSIONS

•The thermal effect of the solar arrays and heat radiation through the spacecraft bottom closeout MLI on the flight temperature prediction of the Phase A baseline XRT radiator at orbit noon is rather significant. It increases the hot case flight temperature prediction of the radiator by 0.5°C at a 180° sun angle and by 5.7°C at a 45° sun angle.

•If SWIFT operates at a sun angle between 110° and 180° for an extended period, the hot case flight temperature prediction of the Phase A baseline XRT radiator exceeds the temperature requirement of –35°C substantially.

•To maintain the XRT radiator temperature in the Phase A baseline location below –35°C, a time constraint on the sun angles is required in flight.

•A new location, on the anti-sun side of the spacecraft, is proposed for the XRT radiator. It requires a CCHP to transport the waste heat from the TEC to the radiator, and the radiator is thermally isolated from the spacecraft.

•The thermal effect of the solar arrays on the temperature of the proposed XRT radiator on the anti-sun side at orbit noon is significant. It increases the hot case flight temperature prediction of the radiator by 5°C at a 90° sun angle and by 17°C at a 180° sun angle. However, there is no direct solar impingement in the new location, and the view factor to space is nearly 1.0.

•The area and mass of the XRT radiator, and the TEC power dissipation have significant impacts on the XRT radiator temperature, and must be optimized.

•With a 9,000 cm^2 and 8.8 kg radiator in the proposed location on the anti-sun side, the hot case flight temperature prediction of the XRT radiator satisfies the temperature requirement of –35°C at all sun angles, for a TEC power dissipation of 20 W or less. The margins are at least 5°C. The mass includes all the CCHPs.

•Ammonia is not acceptable for the working fluid of the CCHPs because its freezing point is -78°C, which is close to the XRT radiator flight temperature prediction in the cold case in the eclipse.

•In the safehold mode, the XRT radiator flight temperature prediction is –90°C.

•Propylene, which has a freezing point of -185°C, is possibly the best candidate working fluid for the CCHPs.

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