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Advances in Spacecraft Thermal Control

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Abstract

Spacecraft thermal management is critical for ensuring mission success, as it affects the performance and longevity of onboard systems. A comprehensive overview of the state of the art in spacecraft thermal control solutions, as well as a design methodology framework for efficient and effective thermal management, is provided. Various thermal control solutions, including coatings, insulation, heat pipes, phase-change materials, conductive materials, thermal devices, actively pumped fluid loops, and radiators, are discussed along with the primary sources of heat loading in space. The need for accurate modeling and analysis of the thermal environment to identify appropriate thermal control solutions and design pathways is highlighted. Future innovations in thermal management, such as new materials and technologies that have the potential to further improve the efficiency and effectiveness of thermal control solutions for spacecraft, are explored.

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Nomenclature

A area [m^2]
d distance [m]
E emitted radiation of Earth [W/m^2]
$\dot{E}$ energy transfer rate [W]
F view factor
h altitude [m]
J radiation flux [W/m^2]
P power of the sun [W]
$\dot{Q}$ heat transfer rate [W]
q heat flux [W/m^2]
R radius [m]
S total solar irradiance [W/m^2]
T temperature [K]

Greek symbols

α absorptivity
ϵ emissivity
φ angle between incident solar radiation and surface normal
ρ albedo, reflectivity
θ solar zenith angle

Subscripts

alb albedo
E Earth
env environment
gen generated
in in
IR infrared
orbit orbit
out out
plan planetary
rad radiating surface
sol solar
stored stored
TOA top of atmosphere

1. Introduction

Today, the exploration and exploitation of space is becoming increasingly common. Space has become a valuable resource for advancing technologies on Earth, both simple and complex. In addition, space offers unique opportunities, such as communication, scientific and militaristic observation, weather monitoring, navigation, remote sensing, surveillance, and data relay services [1]. Spacecraft (S/C) typically consists of a payload and a bus that provides the necessary infrastructure and may contain multiple subsystems, including the thermal control system. The thermal control system is not only important for the S/C, but it also plays a critical role in executing experiments.

All types of S/C use various thermal management solutions to counteract the effects of thermal loading, or lack thereof, from the harsh vacuum of space. Furthermore, internal components generate heat as they dissipate energy from the power utilized to operate the S/C. This power requirement has doubled every 5–6 years as the demand for satellite services has increased [2]. Therefore, the S/C requires additional methods to dissipate heat and maintain temperatures that will enable components to operate effectively.

The future of the satellite industry is witnessing an increasing demand for high-power and high-bandwidth components. At the Indian Space Research Organization (ISRO) Satellite Centre, there is a high demand for high-throughput I-6K communication satellites that can provide 10–15 kW of power and support high-resolution mapping and observation missions [3]. Additionally, there is a need for components such as 10 Gbps data transmission systems that require large bandwidth and processing capability. However, the increased power and bandwidth result in higher heat generation and loading, which highlights the need for more efficient thermal control. Even in the early days of spaceflight, engineers realized the need for more powerful, reliable, flexible, and efficient thermal control systems on S/C [4]. As missions continue to evolve and S/C lifetime requirements increase, the thermal design challenge faced by engineers becomes more complex. The industry must also consider factors such as weight minimization, as it is directly correlated to available fuel, with a S/C life of 1 month per kg of N_2H_4 [4].

The fastest-growing segment of the satellite industry is *small satellites* weighing less than 500 kg [5]. Table 1 shows satellite classifications by weight, including small satellites, which are further subdivided as mini, micro, and nano [5]. The number of small satellite launches has increased tenfold in recent years. The challenge of mitigating thermal loading on S/C through effective thermal management is exacerbated by numerous additional challenges such as microgravity, atmospheric drag, atomic oxygen degradation, vacuum environment, micrometeoroids, and charged

Table 1 Satellite classifications by weight.

Satellite classification	Mass [kg]
Large satellite	>1000
Medium satellite	500–1000
Mini satellite	100–500
Micro satellite (CubeSat)	10–100
Nano satellite	1–10

particles [2]. Emerging trends in S/C and instrument design continue to complicate the already challenging thermal control problem. The future of thermal management must consider high heat flux greater than 100 W/cm^2, temperature control within 1°C, extreme temperature exposure, mass minimization, power minimization, integration of thermal, mechanical, and optical systems, structural stability, and commonality of design for fleets of small S/C [6].

Even the largest space-faring entities, such as the National Aeronautics and Space Administration (NASA), face difficulties in addressing the various components of this challenge. For example, programs like the International Space Station (ISS) and Artemis had to rely on best-guess thermal environment evaluations as the resources were not available to accurately evaluate thermal impacts to these vehicles. Design iterations could be required after flying the profile as data are gathered that would further validate or invalidate previous analysis.

Starting from the conservation of energy equation as it pertains to S/C thermal control analysis, this chapter provides a state-of-the-art review of recent thermal control methodologies to support a design framework for development of efficient and effective S/C thermal control systems. A combination of active and passive thermal control solutions that best assist onboard components in maintaining operable temperature ranges in extreme and varied environments is presented, followed by thermal control guidelines, a design framework, and a discussion of future innovations.

2. Heat transfer analysis of spacecraft

The general form of the conservation of energy is

$$\dot{E}_{\text{in}} - \dot{E}_{\text{out}} + \dot{E}_{\text{gen}} = \dot{E}_{\text{stored}} \tag{1}$$

where $\dot{E}_{in}$ is the rate of energy into a control volume, $\dot{E}_{out}$ is the rate of energy out of the control volume, $\dot{E}_{gen}$ is the rate of energy generated, and $\dot{E}_{stored}$ is the rate of energy stored within the control volume. Since the rate of energy into the control volume is influenced by the S/C environment, it is discussed in detail here.

The energy input, $\dot{E}_{in}$, comprises energy incident upon a S/C from the thermal environment. The radiation from the surroundings can be summarized as

$$\dot{Q}_{env} = \dot{Q}_{sol} + \dot{Q}_{alb} + \dot{Q}_{plan} \tag{2}$$

where $\dot{Q}_{env}$ is the total heat transfer rate from the space environment, $\dot{Q}_{sol}$ is the direct solar radiation, $\dot{Q}_{alb}$ is the total albedo, and $\dot{Q}_{plan}$ is the outgoing long-wave radiation (OLR), also referred to as planetary radiation, emitted from the nearest planetary body which in this case is Earth. A diagram of the incident radiation on a S/C is depicted in Fig. 1. The incident radiation is dependent on the orbital period of the S/C. When the S/C is in the sun, the radiation received includes solar, albedo, and planetary radiation, while in the shadow of the body it orbits, only OLR is received. The direct solar flux is the greatest source of heating for most S/C [7].

Both albedo and OLR must also be considered in the thermal analysis of a S/C. Using Earth as the reference, albedo represents the fraction of incident solar energy that is reflected off Earth and back into space, while OLR represents radiation that has been previously absorbed by Earth and then emitted back out to space. Albedo is highly variable and depends on factors such as cloud cover and the distribution of reflective properties of the surface [7]. Variations in OLR are less severe, with the primary influences being the surface temperature and the amount of cloud cover.

2.1 Solar radiation

The total solar irradiance, denoted by *S*, also commonly referred to as the solar constant, is the amount of solar radiation per unit area received at a surface one astronomical unit away from the sun, perpendicular to the rays. In the mid-1980s, a baseline set of thermal range parameters was used to guide the design of components and thermal systems. However, during

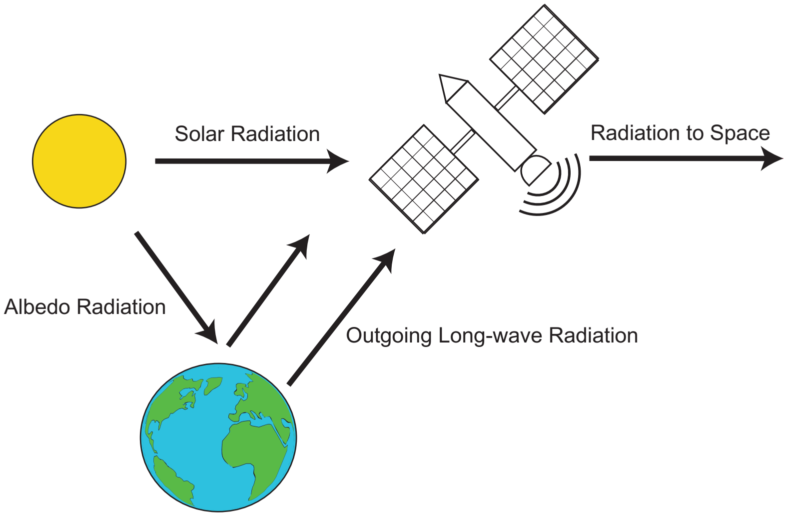

Fig. 1 Summary of radiation incident upon a S/C.

Table 2 Total solar irradiance values from Ref. [7].

Case	**S [W/m^2]**
Hot	1414
Cold	1322
Mean	1367

the development of the ISS in 1992, these parameters came under review as a lack of data made them overly conservative, leading to design difficulties. With the success of the Earth Radiation Budget Experiment (ERBE), which provided extensive data on environmental parameters, NASA created the "Guidelines for the Selection of Near-Earth Thermal Environment Parameters for Spacecraft Design" in 2001 [7].

While the solar constant (values listed in Table 2), the average solar flux at the average Earth–Sun distance, is considered constant in calculations, it actually varies by approximately 3.4% over an orbit. This value trends higher leading up to and following the winter solstice and shows an inverse trend leading up to and following the summer solstice due to the slightly elliptical orbit of Earth. An additional ± 5 W/m^2 could be added to the values in

Table 2 to account for solar cycle variations and uncertainties in the measurements taken from 1969 to 1980, on which these values are based [7].

The Alpha Magnetic Spectrometer (AMS) requires the dissipation of approximately 2000 W of heat through thermal radiation to prevent off-nominal conditions while in orbit. Xie et al. [8] conducted a study to evaluate the thermal performance of the main radiators of the AMS on the ISS. The study aimed to examine the impact of ISS components reflecting solar illumination and shadowing on the AMS radiators. The results of the thermal modeling showed that both radiators receive a solar flux of 1367 W/m^2 when fully illuminated. During the Earth shadow, or eclipse, the input solar flux on the radiator drops to 0 W/m^2. The temperature of the AMS main radiators was found to be influenced by orbit parameters in the low Earth orbit (LEO), ISS components, and operations. The reflected solar illumination from ISS components increases the solar flux by about 80–200 W/m^2, while the shadow from ISS components reduces the solar flux by about 100 W/m^2.

2.2 Albedo and planetary radiation

Albedo values, which represent the reflectivity of a surface, are often expressed as a fraction or percentage and can vary greatly based on factors such as the distribution of reflective properties of the surface and the type and amount of cloud cover. Planetary radiation, also known as OLR, is a combination of radiation from the atmosphere and surface, but it is partially absorbed by the atmosphere. In thermal analysis for S/C, it is typically sufficient to assume a graybody spectrum with a temperature of 250 to 300 K for Earth [7].

Planetary radiation is influenced by the emittance of the planet and the distance of the S/C from the emitting surface. For more specific information on albedo and Earth-emitted infrared (IR) radiation values, refer to Gilmore [1]. The albedo and Earth radiation values presented by Gilmore [1] are based on a NASA/Marshall Space Flight Center study that analyzed 28 datasets of 16-second-resolution sensor data collected monthly from the ERBE. The study was conducted using sensors flown on the ERBE at a low inclination orbit at 610 km altitude and the National Oceanographic and Atmospheric Administration (NOAA) 9 and 10 satellites at high inclination orbits, at altitudes of 849 km and 815 km, respectively.

The investigators performed a statistical analysis to identify maximum and minimum albedo and Earth IR heating rates for time periods ranging from 16 s to 24 h and found that the values did not change significantly for periods greater than 24 h.

2.3 Estimations of energy input from the environment

An equation for the total heat transfer to a S/C, accounting for solar, albedo, and planetary radiation, has been estimated by several researchers. Hengeveld et al. [2] presented the following equation estimating the incident flux on a S/C surface facing Earth from the surrounding space environment.

$$q_{\mathrm{env}} \approx \alpha_{\mathrm{sol}} S + \alpha_{\mathrm{sol}} \left(\frac{R_{\mathrm{TOA}}}{R_{\mathrm{TOA}} + h} \right)^2 \rho_{\mathrm{alb}} S \cos(\theta) + \epsilon E \left(\frac{R_{\mathrm{TOA}}}{R_{\mathrm{TOA}} + h} \right)^2 \quad (3)$$

In this equation, α_{sol} is the solar absorptivity of the surface of the S/C, S is the total solar irradiance, R_{TOA} is the radius of the Earth (TOA meaning top of atmosphere), h is the S/C altitude, θ is the solar zenith angle (see Fig. 2), ϵ is the longwave emissivity of the surface of the S/C, and E is the radiation emitted by Earth, which is assumed to be a blackbody. The albedo, ρ_{alb}, is defined as the fraction of reflected incident solar radiation reflected by a planetary body into space.

Bonnici et al. [9] modeled the thermal environment similar to Hengeveld et al. [2]. The incident solar radiation is defined as

$$q_{\mathrm{sol}} = \alpha_{\mathrm{sol}} S \cos \varphi \quad (4)$$

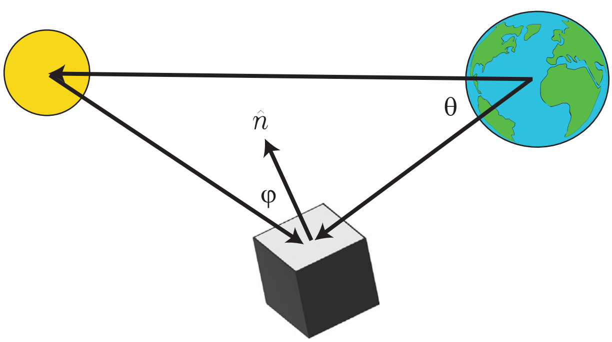

Fig. 2 Angles used in equations for radiation incident upon a S/C.

Here, φ is the incident angle of the sun on the S/C and is defined in Fig. 2. The albedo heat flux is defined as

$$q_{\text{alb}} = \begin{cases} F_E \alpha_{\text{sol}} S \rho_{\text{alb}} \cos\theta & \text{if } \theta < \frac{\pi}{2} \\ 0 & \text{if } \theta \geq \frac{\pi}{2} \end{cases} \tag{5}$$

In this equation, F_E is the view factor relative to the surface of the Earth. The OLR is defined as

$$q_{\text{plan}} = F_E \alpha_{\text{IR}} E \tag{6}$$

Alternatively, Alcayde et al. [10] present the following equation, which is similar to the analysis in the reference book by Miao et al. [11].

$$\dot{Q}_{\text{env}} \approx \alpha A_{\text{sol}} J_{\text{sol}} + \alpha A_{\text{alb}} J_{\text{alb}} + \epsilon A_{\text{plan}} J_{\text{plan}} \tag{7}$$

Here, A_{sol}, A_{alb}, and A_{plan} are the areas receiving solar, albedo, and planetary radiation, respectively. J_{sol}, J_{alb}, and J_{plan} are the heating components of solar, albedo, and planetary radiation, respectively. Furthermore, α is the absorptivity and ϵ is the emissivity of the S/C surface. The solar radiation, at a given distance, is defined in this case as

$$J_{\text{sol}} = \frac{P}{4\pi d^2} \tag{8}$$

where P is the power of the Sun, 3.846×10^{26} W, and d is the distance from the Sun to the S/C. The intensity of the albedo radiation impinging on the S/C is

$$J_{\text{alb}} = J_{\text{sol}} \rho_{\text{alb}} F \tag{9}$$

In this equation, ρ_{alb} is the albedo parameter, taken as 0.33 in Ref. [10] and F is the view factor. The value F is a function of the S/C altitude and the angle between the orbit plane and the Earth–Sun vector. The OLR is defined as

$$J_{\text{plan}} = 237 \left(\frac{R_{\text{rad}}}{d_{\text{orbit}}} \right)^2 \tag{10}$$

In this equation, the radius of the effective radiating surface of the planet is R_{rad} and d_{orbit} is the distance of the S/C from the center of the Earth. Typically, for Earth, R_{rad} is assumed to be the radius of the surface of the

Earth; however, this would not be the case for planets without an atmosphere. The leading constant, 237, is specific to Earth only [10].