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LETECKÝ ÚSTAV

THERMAL CONTROL SYSTEMS FOR MICROSATELLITES AND PROBES
SYSTÉMY TEPELNÉ REGULACE PRO MIKROSATELITY A SONDY

BACHELOR'S THESIS
BAKALÁŘSKÁ PRÁCE

AUTHOR
AUTOR PRÁCE
Marko Bermell

SUPERVISOR
VEDOUCÍ PRÁCE
Ing. Jakub Mašek

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Assignment Bachelor's Thesis

Institut:	Institute of Aerospace Engineering
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As provided for by the Act No. 111/98 Coll. on higher education institutions and the BUT Study and Examination Regulations, the director of the Institute hereby assigns the following topic of Bachelor's Thesis:

Thermal control systems for microsatellites and probes

Brief Description:

Space vehicles are exposed to harsh environment. Long diurnal cycles and related thermal fluxes play an essential role for the thermal comfort of microsatellites. It is important to ensure sufficient heat dissipation for a limited time. Systems incorporating a thermal switch may be one solution to ensure variability in thermal conductivity.

Bachelor's Thesis goals:

– Thermal systems with heat switch, description and classification of system elements
– Characterization of performances, properties and their comparison
– Qualified solutions for the space industry and specific applications

Recommended bibliography:

GILMORE, David G. Spacecraft Thermal Control Handbook, Volume I: Fundamental Technologies. El Segundo,California: The Aerospace Press, 2002, 836 s. 2nd ed. ISBN 978-1884989117.

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doc. Ing. Jaroslav Juračka, Ph.D. Director of the Institute	doc. Ing. Jiří Hlinka, Ph.D. FME dean

Abstract

This article presents an overview of thermal control system designs and trends for unmanned spacecraft during their cruise and operational phases. The thermal control systems play a crucial role in maintaining the electronic and mechanical components within their specified temperature ranges, considering the diverse and extreme environmental conditions encountered in space. The systems are categorized based on their distance from the Sun and their function, providing insights into their layouts and trends. The analysis covers spacecraft in various orbits and configurations, including microsatellites and space observatories. The article concludes with a summary of the main points and a comparative table highlighting the characteristics and power balance of different thermal control system categories.

Key Words

Thermal control, spacecraft, temperature, microsatellite, overview, heat balance

Rozšířený abstrakt

Vesmírné sondy a družice čelí extrémním teplotním výkyvům. Elektronické a mechanické součásti lodi však obvykle dokážou pracovat jen v určitém intervalu teplot. Obvykle je to v rozmezí 0–50 °C, v případě některých detektorů jsou to i kryogenní teploty v jednotkách nebo desítkách Kelvinů. Komponenty jako baterie, optické přístroje nebo palivové nádrže potřebují velmi stabilní teplotu. Naopak antény nebo solární panely snesou rozdíly teplot několika set °C. Úkolem tepelných regulačních systémů je udržet teplotu na příznivé úrovni pro danou komponentu při zachování co možná nejnižší hmotnosti.

Cílem této práce je poskytnout přehled různých konstrukcí a trendů v oblasti tepelné regulace. V odborné literatuře jsou obvykle uvedeny tepelné komponenty a vysvětlena jejich funkce. Tento dokument se však snaží ukázat jejich úlohu a umístění v rámci systému jako celku a dále v rámci různých systémů tepelné regulace v různých podmínkách. Na základě těchto poznatků je vytvořeno komplexní srovnání uspořádání a trendů. Systémy jsou rozděleny do kategorií především na základě jejich vzdálenosti od Slunce, neboť to je klíčovým zdrojem tepla v naší sluneční soustavě, a také na základě jejich funkce a celkového uspořádání. Tato práce je zaměřena výhradně na bezpilotní kosmické lodě s výjimkou fáze startu a případného atmosférického návratu. Výsledkem práce je tabulka porovnávající charakteristiky a tepelné bilance jednotlivých kategorií.

Kosmické lodě jsou vystaveny přímému záření ze Slunce. Dále mohou být vystaveny odraženému slunečnímu svitu z povrchu planety nebo měsíce a jejich vlastnímu tepelnému záření. V poslední řadě je teplo generováno vnitřní elektronikou. V rámci lodi se teplo přenáší vedením a zářením. Lodě ve vakuu se zbavují a získávají teplo pouze zářením do kosmu, jenž má teplotu kosmického pozadí (2,7 K).

Satelity na oběžných drahách kolem Země jsou vystaveny celoročně stálému slunečnímu záření o přibližně 1,5 kW/m^2. Na nízké oběžné dráze (LEO) jsou navíc vystaveny střídání dne a noci a záření ze Země, které přichází z jiného směru než to

sluneční. Většinou se jedná o komunikační družice, které produkují velké množství vnitřního tepla v řádu jednotek až několika desítek kilowattů.

Sondy ve vnitřní sluneční soustavě, od Venuše až po vnější pás asteroidů, mají výkon od 100 do 2000 W. Tyto sondy mají společné, že se na své dráze setkávají s měnící se intenzitou slunečního záření, od 2,5 kW/m^2 u Venuše po méně než 100 W/m^2 u vnějších asteroidů a komet. Kosmické sondy na oběžné dráze Měsíce jsou vystaveny střídavým cyklům dne a noci, a také tepelnému záření z povrchu Měsíce. Tepelné systémy sond u Venuše jsou primárně zaměřeny na prevenci přehřátí. Sondy na Marsu se zaměřují na uchování tepla. Sondy mířící k asteroidům a kometám upřednostňují uchování tepla s minimálním využitím energie pro vytápění. Louvery mají klíčovou funkci při řízení teploty během letu.

Kosmické lodě na oběžných drahách pod 0,5 au jsou vystaveny extrémnímu slunečnímu záření v řádu 10^4 až 10^6 W. Nejvyšší teploty zvládají sondy s tepelným štítem namířeným stále ke Slunci. Odlišnou filozofii představují sondy obalené ze všech stran n2kolika vrstvami MLI, jež spoléhají na silnou izolaci a stíněné radiátory. V případě sondy MPO toto umožňovalo zvládnout záření ze Slunce i Merkuru na jeho orbitě. Kosmické lodě stabilizované rotací jsou pokryty odrazivými prvky, které odrážejí většinu tepla. Všechny kosmické lodě pod 0,5 au využívají další opatření, jako jsou antény a detektory s vysokou tepelnou odolností a solární panely se střídavými odrazivými pásy, které se dají naklápět.

Ve vnější sluneční soustavě je sluneční záření téměř zanedbatelné, proto je hlavní úlohou systému tepelné regulace především izolace a dohřívání v řádu 10^2 W. Sondy mohou využít radioizotopové termoelektrické generátory (RTG) k výrobě energie, což s sebou nese řadu potíží s tepelnou regulací. Malou část odpadního tepla je možné použít k ohřevu kosmické lodi, především palivových nádrží. Většina energie generované RTG se však kvůli malé účinnosti vyzáří do kosmu. Užitečná energie se používá k napájení elektroniky a ohřívačů. V současnosti se místo RTG stále častěji používají solární panely, což má za následek odlišné návrhy tepelných systémů.
Miniaturní satelity mají mají vysoký poměr výkonu ku povrchu, což vyžaduje účinné metody odvodu tepla a značně mění konstrukční filozofii. Je třeba věnovat zvláštní pozornost přenosu tepla v rámci družice, jelikož drobné součástky mají v malém prostoru mnohem větší vliv. V současné době bylo jen málo mikrosatelitů posláno od větší vzdálenosti od Země, jelikož prvky jako antény nebo tepelné štíty je výhodnější použít u větších sond.

Vesmírné observatoře vyžadují pro optimální fungování kryogenní teploty. Většina tepla tak musí být odražena nebo vyzářena. Jedná se o výkony v řádech 102 až 103 W. Samotná kosmická loď musí udržovat stálou teplotu, aby se zamezilo teplotní deformaci, což by znemožnilo přesná vědecká měření. Teleskopy na heliocentrických oběžných drahách a SSO jsou vybaveny sluneční clonou, která chrání dalekohled před slunečním zářením. Dalekohledy na nízké oběžné dráze kolem Země využívají k řízení zamezení vstupu tepla reflexní a izolační materiály ze všech stran. Tělo lodě je rozděleno na oddíly podle teploty. Teleskop je od těla tepelně izolován. Detektory jsou napojeny na kryogenní chladicí systém. Provozní životnost observatoří zpravidla závisí na rychlosti vyčerpání chladicího média. Některé využívají také mechanické chlazení pro prodloužení životnosti.

Komerční a vědecké kosmické lodě používají při vývoji systémů tepelné regulace obvykle osvědčené postupy a prvky. Výjimky představují především vědecké mise do stále extrémnějších prostředí. Orientace kosmické sondy a její dráha představují potíže i příležitosti pro tepelnou regulaci. Různé kategorie kosmických lodí, podle jejich vzdálenosti od Slunce a konfiguraci, mají odlišné požadavky na tepelnou regulaci. Je zřejmé, že se celkově upřednostňují pasivní systémy tepelné regulace, avšak všechny satelity jsou také vybaveny aktivními topnými tělesy. V případě potřeby se využívají i aktivní chladiče, především u komunikačních satelitů a observatoří. Dalším společným prvkem je potřeba izolace a ochrana před výkyvy teplot, ačkoliv se jednotlivé kategorie liší v její míře i provedení. Výstupem práce je tabulka srovnávající vlastnosti a energetickou bilanci různých kategorií kosmických lodí spolu se shrnutím jejich hlavních charakteristik.

Bibliography

BERMELL, M. *Thermal Control Systems for Microsatellites and Probes*. Brno, 2023. Bachelor thesis. Brno University of Technology, Faculty of Mechanical Engineering, Institute of Aerospace Engineering. Supervised by Ing. Jakub Mašek.

Declaration of Authenticity

I declare that I have worked on this bachelor's thesis independently, using technical literature and other sources which are all properly quoted in the thesis and detailed in the list of bibliography at the end.

Marko Bermell

26. 5. 2023

Brno

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Table of Contents

1 Introduction

Electronic and mechanical components of a spacecraft usually operate correctly within a specific temperature range. This is normally around room temperature, as they were derived from components for terrestrial use. On the other hand, some specialized detectors require very low temperatures for correct operation. The role of a Thermal Control System (TCS) is to ensure that the equipment stays within its operating range, and thus to combat the extreme temperature swings of a space environment. [1]

Space environments in which the probes operate vary significantly: A probe flying close to the Sun will be exposed to very different conditions compared to a probe in low Earth orbit, or one flying far beyond the orbit of Jupiter. Moreover, two spacecraft flying in the same environment can have different dimensions, power sources, or guidance and positioning requirements. These factors influence their layouts, and consequently the thermal control systems. New spacecraft tend to be based on their predecessors; they take parts, ideas, and improve them, which shapes thermal control system designs. [2]

The goal of this thesis is to provide an overview of different thermal control system designs and trends. Scientific literature usually lists the thermal components and explains their function. However, this document shows their role and placement within the system as a whole, and furthermore within various thermal control systems in different conditions. With these findings, a comprehensive comparison between layouts and trends is created. The systems are primarily categorized based on their distance from the Sun, as it is the key heat source in our Solar System, and also based on their function and general layout. This thesis is focused entirely on unmanned spacecraft during their cruise and operational phase, that is, excluding launch and possible aerobraking.

2 Thermal Management of Spacecraft

This chapter discusses key concepts of thermal management.

2.1 Heat Transfer

The balance between heat gained and lost will determine the spacecraft temperatures. As seen in fig. 2-1, heat is gained from solar heating, either directly or reflected by the planet. Heat is also received from proper IR heating from the planet. Heat is generated within the spacecraft by its components, and it is emitted via thermal radiation. [3]

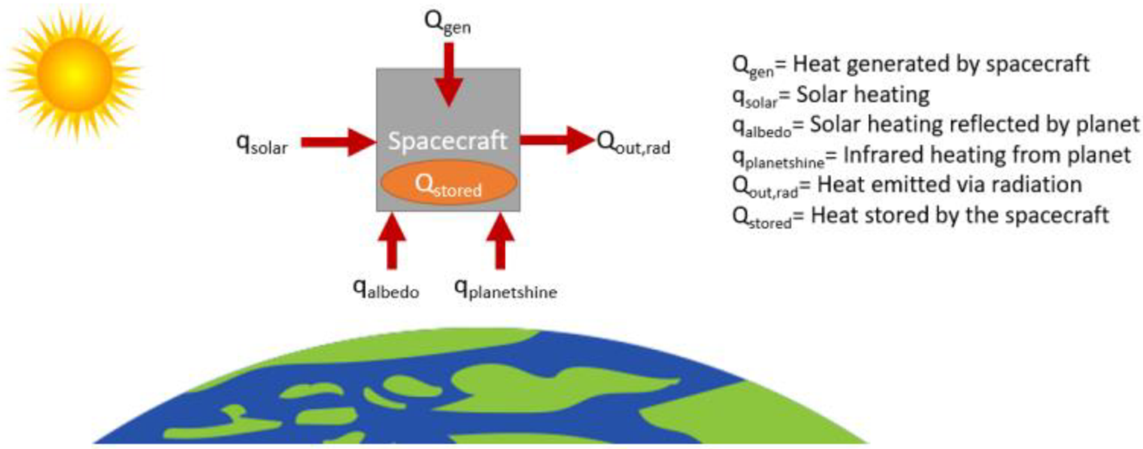


Figure 2-1 Sources and flow of heat on a spacecraft. [3]

Solar Radiation

Heat from the space environment is largely the result of direct solar radiation. 99 % of this energy comes from the 150 nm to 10 µm range (UV to IR), with a peak in the yellow part of the visible spectrum. The intensity follows the **inverse-square law** and thus rapidly *increases* as the spacecraft gets *closer* to the Sun. The amount of heat absorbed this way depends on the absorptance α of the surface finish, which can range from 0 to 100 %, area A of the exposed parts, and the respective heat flux q. It can be seen in the equation below. Note that *albedo* (‘whiteness’) is sometimes used instead of absorptance. [4]

$$P = qA\alpha \text{ [6]}$$

The Sun’s radius in the sky is about 0.5° at 1 au, so sunlight can be regarded as a parallel beam from a point source. This is not the case for spacecraft that get very close to the Sun, which must take into account the angle of the converging beam.

A spacecraft on its orbit may spend periods of time in the planet’s shadow, where it gets no sunlight. Figure 2-2 shows an example of the temperature variation of a solar panel as it passes through Earth’s shadow on a geostationary orbit, therefore experiencing an eclipse. [5]

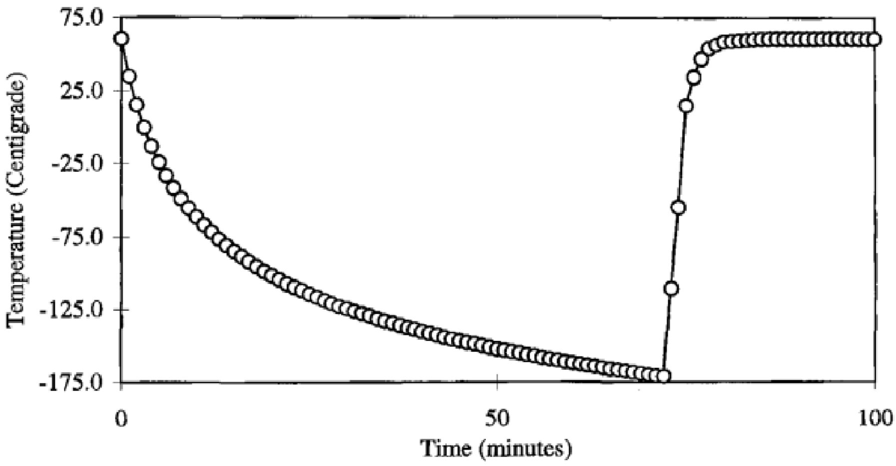


Figure 2-2 Temperature variation of a solar panel during geostationary orbit eclipse [5]

Planetary Radiation

Reflection of solar radiation from a planet or moon (albedo radiation), as well as its proper infrared radiation (thermal radiation) contribute to the heating of a spacecraft. The intensity depends on the spacecraft's distance from the planet/moon, more specifically on its apparent radius, therefore the importance of these factors will depend on the orbital path. [1]

Spacecraft-generated Heat

Heat is generated within the spacecraft by rocket motors, electronic devices, batteries, and radioisotope thermoelectric generators (if a spacecraft is equipped by one). It is transferred within the spacecraft by conduction through joints, as well as radiation between surfaces. Heat is also lost from the spacecraft by radiation. The rate of radiative heat loss depends on the emissivity ε, temperature T (in Kelvin) and area A of the exposed parts, and Stefan-Boltzmann constant, as seen in the equation below.

$$P_{rad} = \sigma \varepsilon A T^4 \text{ [6]}$$

Atmospheric Heat Exchange

On bodies with significant atmospheres, such as Venus, Mars or Titan, heat can also be transferred to and from the environment via conduction. The rate depends on the thermal resistance R between the exposed parts and the atmosphere, as well as their area A and temperature difference ΔT. [6]

$$P = \frac{\mathrm{A} \cdot \Delta T}{R}$$

2.2 Thermal Components

Thermal components form the backbone of the TCS. Passive thermal control maintains component temperatures without using powered equipment. Passive systems are generally cheaper, lighter and by having little to no moving parts, they have a lower risk of malfunction. They are advantageous to small spacecraft, such as SmallSats and CubeStats. Active components require an external power source to work. They can adapt to the changing demands of the TCS and can handle extreme thermal loads. [3]

Passive Control

Radiators are panels with high emissivity. They are placed in direct view of outer space and connected to other components which they cool down. Radiators can be body-mounted or deployable. A well-placed radiator can cool down to cryogenic temperatures of outer space. [3]

Thermal Louvers are usually operated by bimetallic strips. They open and expose the radiator when the spacecraft is too hot, and close to prevent outward radiation when the spacecraft is too cold. Some louvers are static and only serve as sunshields to prevent sunlight from hitting the radiator from specific angles. [3]

Paints and coatings on the outer layer can absorb or reflect heat from external sources. They are also implemented inside the spacecraft to equalize the temperature and eliminate gradients, or to insulate components. Diffuse paints prevent reflections from hitting other spacecraft components. [3]

Optical Solar Reflectors (OSRs) reflect sunlight, and prevent it from entering the spacecraft. They often serve as radiators, so in some literature they are used interchangeably. [3]

Multi-Layer Insulation (MLI) can insulate the spacecraft or individual components from the environment or from each other. It also serves as micro-meteorite protection, or solar wind protection. The outer layer can be coated or painted. [3]

Sunshields can shield the whole spacecraft or specific components from radiative heat form the Sun, planets, or high-dissipation components, such as a Radioisotope Thermoelectric Generator. [3]

Tapes and other conductive paths are used to increase conductivity between parts, and conduct heat either from or towards components. [3]

Heat pipes have a similar purpose to tapes, but they use capillary effects to conduct heat. They are often used to deliver heat and spread to the radiators. [3]

Phase Change Materials are used on some spacecraft to store heat when demand is low and release it when demand is high. [3]

Active Control

Heaters work by using electrical resistance to produce heat directly. [3]

Cryocoolers are used on sensitive instruments, such as infrared imagers. As the name suggests, they cool it down to cryogenic temperatures. [3]

Fluid Loops are mechanically pumped. They can transport more heat over longer distances than heat pipes or thermal straps. [3]

2.3 Spacecraft Configuration

Spacecraft can position themselves with respect to the Sun and planet as a way of controlling the radiation and thus the heat flow. The effects of this can be a very important part of designing the TCS. Three-axis-stabilized probes, which can point in and stay in any direction, may have very different possibilities than spin-stabilized probes. The spacecraft's size, and especially its component and power density, has a critical influence on the TCS design.

The thermal exposure of a spacecraft can be altered by the choice of orbit. For instance, a highly elliptical orbit takes it far away from the planet for long periods of time. In a Sun-synchronous orbit the Sun never sets. The travel path can also determine and limit the TCS design, as some spacecraft may want to take advantage of gravity assists in different parts of the Solar System.

Heat balance is considered for two extreme cases (hot and cold) to design a spacecraft that meets the temperature requirements on both upper and lower limits. The power profile for the hot case corresponds to the case in which all components are at the highest level of heat dissipation, while the orbit is such that the spacecraft is exposed to maximum solar, albedo and Earth IR loads. All margins should be included into the input data to produce the maximum possible temperature. Similarly, the input data for the cold case should be selected to result in the lowest possible temperature. However, the assumptions made for both cases should be realistic and correspond to the considered mission. For example, solar flux cannot be perpendicular to both the radiator and solar array, if they are perpendicular to each other.

A preliminary analysis usually begins with consideration of the hot case. In the hot case a radiator area is estimated for a given external heat load, a given internal heat generation level, and a given radiator temperature. The estimated radiator area should be compared with an available external surface of the spacecraft for heat rejection into space. In the cold case, radiator temperature is determined for a given radiator area (determined earlier in the hot case study) and the lowest possible heat load including external and internal heat loads. [7]

3 Earth Orbit

The environment around Earth is characterized by a constant solar flux of 1361 W/m^2. The value varies through the year by ± 4 % because of Earth's orbital eccentricity. Depending on their orbit, spacecraft may often find themselves in Earth's shadow. There are two most common orbits: Low Earth Orbit and Geostationary Earth Orbit. Other orbits have environments that are on a spectrum between those two. Most Earth satellites are high-power communication satellites, which have internal heat dissipations often a magnitude higher than the scientific missions beyond Earth.

3.1 Low Earth Orbit

Low Earth Orbit (LEO) is the region of space surrounding Earth where satellites can maintain a stable orbit with an altitude of approximately 200 to 2000 kilometers. LEO is the economically easiest to reach. The proximity to Earth's surface makes it appealing for Earth-observation satellites. It also results in a short information travel time, which is good for communication satellites. On the other hand, the proximity limits their field of view, which forces them to operate in large constellations. This drives the satellites' simplification and miniaturization. This phenomenon is often called NewSpace. [8] [9]

From a TCS perspective, LEO spacecraft can be divided into three categories. First, small sats, where the main limitations are size and weight. They are discussed in detail in the chapter 7. Second, Earth-pointing satellites, mostly communication. Low price and dimensions are desirable, but they also need to accommodate for the demand for more powerful electronics. The last category are various military satellites and technology demonstrators. Their TCS vary significantly due to their nature and require very specific thermal design solutions.

TCS of spacecraft in LEO must tackle two main issues: **the day/night cycle and radiation coming from two directions.**

As mentioned, the *solar flux* is 1361 W/m^2 at 1 au. More importantly, as spacecraft in LEO have orbital periods of only up to 128 min, they often find themselves in Earth's shadow. This frequently changing aspect of the environment must be tackled by the TCS. [8]

Earth reflects solar radiation, which is known as the *albedo flux*. It has a value around 30% of the solar flux when emitted from Earth's surface. The spacecraft only receives this from the illuminated side of Earth, so this is another frequently changing factor that the TCS must tackle. [8]

Some of the solar flux reaching Earth is absorbed and then reemitted as longwave *IR radiation* due to the temperature of Earth. Earth can be considered as a black body, and its effective blackbody temperature is assumed to be 255 K. Earth's relatively fast spin and presence of an atmosphere minimizes the temperature differences between the day and night sides; therefore, the IR radiation can be considered constant throughout the orbit. Typically, the infrared radiation received by a low Earth orbit satellite from Earth is about 230-250 W/m^2, which is constant regardless of latitude and longitude. [8] [10]

Satellites in LEO operate high power electronics 10^2 to 10^3 W in a compact bus. This means that the TCS will be mostly oriented towards dissipating heat. The bus is usually rectangular or a flat panel-type. The latter allows for a large **body-mounted radiator**, or for the body itself to be used as a radiator. MLI and Optical solar reflectors (OSRs) are used to prevent the overheating of equipment and the transfer of generated heat into space during the mission. Smaller radiators may be placed near specific demanding equipment. [11] [12]

Equipment is densely packed, so various types of **thermal coupling** is used to prevent overheating, such as thermal interface fillers. These are added between the panels and the electronic equipment. Heat pipes and doublers are applied to the radiator for effective heat dissipation. [9] [13]

Payload dissipation can make around 70% of the power. Despite that, **electric heaters** are used to prevent the spacecraft from freezing during eclipses on the night side, or when not much of the equipment is in use. Especially sensitive are the batteries and hydrazine tanks. Usually there are heaters operated by bimetallic thermostats near critical equipment, so that it can operate even in an event of software failure. [11] [12]

Besides the mentioned heaters, satellites in this category tackle the day/night cycles by having high-enough thermal inertia. In fact, they often take advantage of the shadow phase to cool down the radiators. However, there is one type of LEO which does not experience such day/night cycle, called *Sun-Synchronous Orbit (SSO)*. The orbital plane of a probe in SSO turns slowly around the north/south axis (about one degree per day or one revolution per year). This rotation is synchronized with Earth's rotation around the Sun in such a way that the Sun is fixed relative to the orbit. [spot4]

Spacecraft in SSO do not differ significantly from other LEO spacecraft, besides having larger radiators for constant heat dissipation. They are still equipped with heaters, as mentioned, for phases where some of the equipment is turned off.

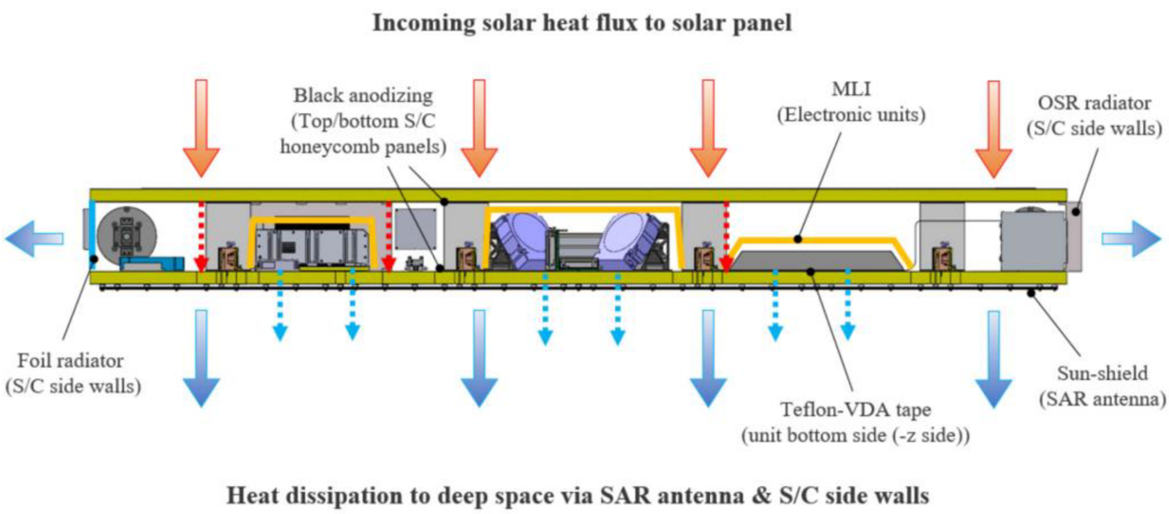


Figure 3-1 The heat balance of a flat LEO satellite. [9]

3.2 Geostationary Orbit

A *Geostationary Orbit* (GEO) is a circular orbit at around 36,000 km above Earth's equator. The spacecraft appears to be motionless to a ground observer since the orbital period is equal to one sideral day. Similarly, a *Geosynchronous Orbit* (GSO) has the same period but is not necessarily placed above the equator. GEO is very popular for communication, weather, and navigation satellites.

Satellites on GEO and GSO have a block-shaped bus. The propulsion and fuel tanks are placed in the middle in a cylindrical compartment. The electronics (payload) are placed around it along the spacecraft's walls. One side of the spacecraft always points at Earth, and so it leaves the other sides pointing toward the other cardinal directions. The solar arrays stem from the north and south sides, roughly perpendicular to the orbital plane. This is in line with Earth's equator; therefore, the sun never illuminates the north and south sides at an angle higher than 23.5°. Most satellites have **body-fixed radiator panels** located here. The transponders, which are the main power consumers on board of the satellite, are usually directly mounted on the backside of the radiator panels (i.e., the payload walls). [15]

Solar panels on commercial satellite platforms provide a large amount of electrical power. Around 40% of this power is transmitted as data by the transponders on board of the satellite. The remainder of the power is turned into heat which must be dissipated into space by the radiators. Electric power demand is already high, often reaching up to 25 kW. [15]

Using only north and south body-fixed radiators restricts the power consumption of satellites to around 11 kW. The LS1300 platform uses additional east and west radiators, but these experience large variations in solar illumination, which makes this option less attractive. **Deployable radiators** are thus the main option for increasing the cooling area. Deployable radiators are used even in less powerful satellites when the bus needs to be smaller to fit multiple on a single launch vehicle, therefore limiting the body-fixed radiator size. [15]

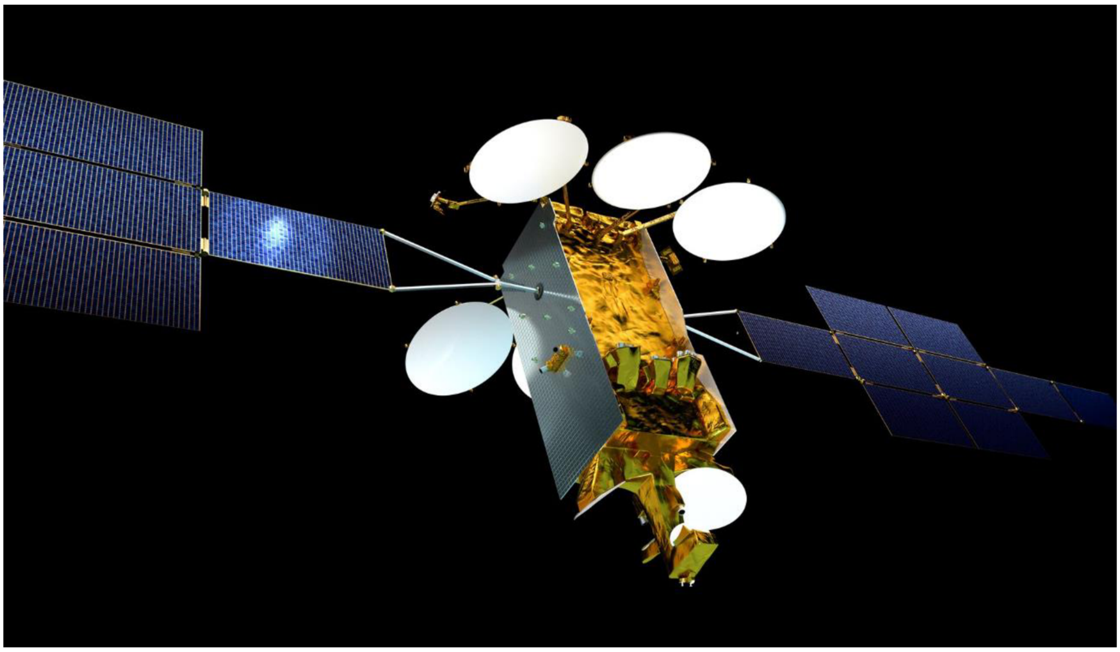


Figure 3-2 Eurosat Neo. The antenna dishes are facing Earth. Notice the radiators (silver) on the north and south sides, with the solar panels. The orbital plane, which copies Earth's equatorial plane, is perpendicular to the panels' axis. [15]

Heat pipes are currently the workhorse for thermal control of spacecraft and are used in practically all modern-day satellites. Heat pipes are used to distribute the localized heat input of the transponder over the total radiating area of the panel. A drawback of using heat pipes is the problem related to ground testing due capillary behavior in the gravitational field. Another challenge is the increased complexity and mass of the heat pipes in systems with high power demand. On top of that, the distance between the radiators and cooled components increases as the radiators become larger. The BSS-702 platform uses a separate cooling loop for the bus and for the payload area, which increases its thermal performance and thus payload power capacity over older versions (see 3-3). Other alternative systems have been investigated in recent years. [15]

Mechanically Pumped Fluid Loops (MPFL) have the advantage over heat pipes that they are very flexible in their design and can be stretched over a long distance thanks to the pump. In contrast with heat pipes MPFLs force a constant mass flow via an active pumping system. Multiple heat exchangers can be placed in series, i.e., thermal control of a large number of units can be achieved with a single loop. For small heat transport lengths, the mass of the accumulator has a huge negative impact on the total mass of a MPFL, compared to a heat pipe that has no accumulator. MPFLs are easily integrated into older satellite buses, reducing cost derived from the integration. [15] [17]

A MPFL consists of the following components: *accumulator* – pressurizes the loop and buffers the fluid volume variations; *pump* – considered as the weakest component and the most likely to fail during operation because of moving parts; *evaporator* – to transfer large amounts of heat from a heat-dissipating element to a fluid; *condenser* – heat exchanger. They add complexity to the system but are easier to mount and do not use toxic liquids, which further simplifies their handling (heat pipes use ammonia). MPFL are to be used on systems with heat dissipation >10 kW. [15]

Most of the bus is covered in **MLI** to equalize large temperature spikes and differences. The surface coating is reflective, which usually gives it a golden look. [15]

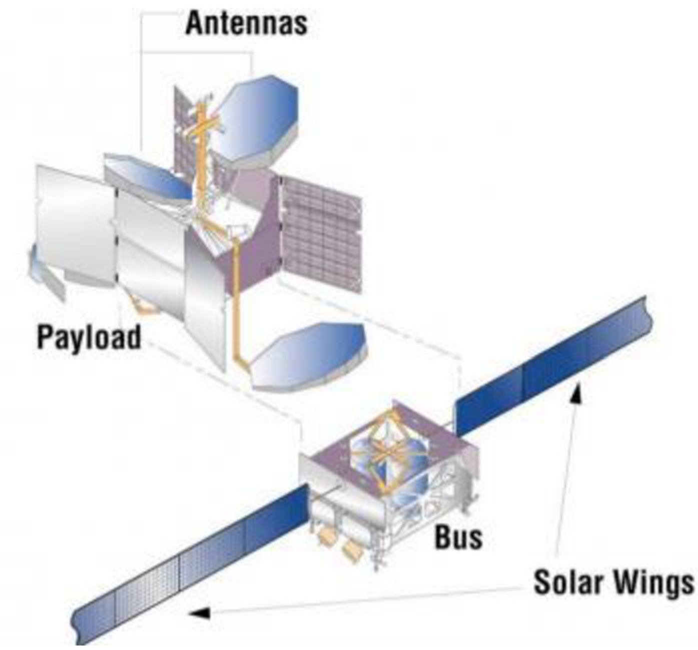


Figure 3-3 Separated payload and bus on the BSS-702. Notice the deplozable radiators. [18]

4 Inner Solar System

Probes flying beyond Earth orbit are generally scientific and have powers of 100 to 2000 W. They experience varying Sun illumination along the way, from about twice Earths value at Venus (0.7 au) to 4 % at the 5.2 au. If the target is a planet or the Moon, the TCS must face the thermal challenges caused by the body.

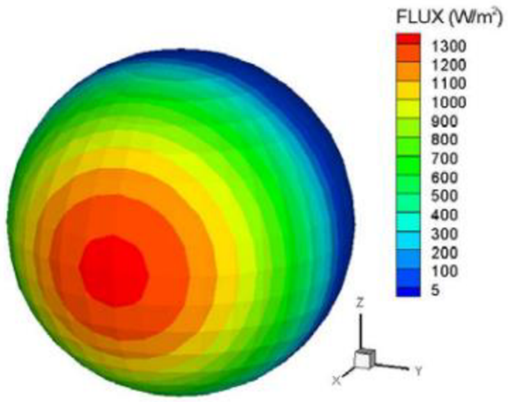
The Venus Express and Mars Express probes give us an unique opportunity to compare TCS within the Inner Solar System. On top of that, they are based on the Rosetta spacecraft design, a probe that visited a comet. They show that while each is optimized for their respective conditions, they are still similar enough to be mounted on a common bus. The main differences are surface coatings (Mars Express is black and Venus Express is gold), use of louvers (Rosetta uses louvers to stop radiating heat in the outer phases), and the size and design of solar panels. The focus of the TCS changes from rejecting heat to conserving heat during their paths. It must be capable of handling both conditions. [19]

4.1 The Moon

Spacecraft in lunar orbit stay at roughly the same distance from the Sun all year long. The solar flux is 1361 W/m^2 ± 4 %. It varies as the Moon orbits Earth, and because the orbit of Earth around the Sun is elliptical. The orbital path takes the spacecraft into the Moon's shadows for up to several hours. In addition, the spacecraft might find itself in Earth's shadow during a lunar eclipse. This will not only block the solar radiation from heating the spacecraft, but as a result, no power will be generated by the solar array. This drives up the battery size to provide sufficient power for the heaters. The spacecraft is often *preheated* in preparation for the eclipse. For instance, Danuri begins this process on the 10th orbit before the eclipse. [10] [20]

As a result of the lack of an atmosphere, the length of the lunar day, the low thermal conductivity, and low albedo (0.073 on average), the lunar surface gets very hot. The infrared radiation from the Moon varies greatly with latitude and longitude, ranging from 5 to 1335 W/m² in hot conditions and 5 to 1114 W/m² in cold conditions (see fig. 4-1). The IR loading must be modelled carefully for useful results. Albedo radiation is relatively low and can be seen in fig. 4-1. [10] [20] [21] [22]

Probes operating around the Moon mostly point towards its surface (the nadir). The thermal power output is in the range of 10^2 W. [10] [20]



	Lunar Orbiter		Low Earth Orbiter	
	Hot	Cold	Hot	Cold
Solar Flux (W/m²)	1420	1280	1420	1287
Albedo	0.13	0.06	0.35	0.3
Planet IR Radiation (W/m²)	5-1335 (Variation)	5-1114 (Variation)	249	227

Figure 4-1 Lunar IR environment (left) and heat values for lunar orbit and LEO. [10]

The thermal designs are combinations of active and passive components. As the thermal environment of the Moon changes significantly during the orbit, the passive thermal control approach tries to insulate the spacecraft as much as possible. MLI is applied to most of the orbiter's exterior to completely block radiant heat exchange with space, and OSRs are applied to radiator panels to dissipate heat generated internally to space. Radiators are not placed on the nadir-facing side to avoid IR radiation. On Danuri, white paint was applied to the back of the solar panels and HGA to prevent excessive temperature rise. The materials used on the bus are not very thermally conductive. Louvers can be used to change the emissivity of the spacecraft for thermal management during the eclipse, however, most spacecraft manage without them, using preheating and larger batteries instead. [10] [22]

In the interior, highly thermally conductive pads are used on the heat-generating electronics to increase the heat conduction of the equipment and panels. On the other hand, batteries are highly affected by temperature, so internal MLI and thermal isolators are applied to the battery mounting panel to block heat exchange with other electronics. The Lunar Reconnaissance Orbiter (LRO) uses 25 *small* heaters together with aluminum tape were used to keep the temperature stable while preventing uneven heating. On Danuri, black paint was applied to all equipment and panels inside the multi-orbit vehicle to facilitate internal heat flow, and internal MLI was applied to the propellant tank and pressurization tank to block heat exchange with other equipment. [10] [22]

Danuri, being relatively small at 550 kg, managed conduct heat efficiently without the use of heat pipes. It is a result of an iterative analysis, and it managed to reduce the weight. LRO, on the other hand, makes full use of heat pipes. Its avionics module is connected to a single radiator using a network of heat pipes, which allows it to be larger in size. In addition, it makes it flexible in design and allows for change of equipment without major changes in the TCS. [10] [22]

For the active thermal control method, heaters are used. On Danuri, the equipment that requires precise temperature control (battery, star tracker, reaction wheel, etc.) is divided into a processor control heater group that can change the operating temperature of the heater. The remaining equipment (gyro, S-band transponder, etc.) is designed as a thermostat control heater group that cannot change the operating temperature during the mission. In addition, all heaters used during the mission were composed of a main heater and auxiliary heater to prevent single fault failure of the heater. [10]

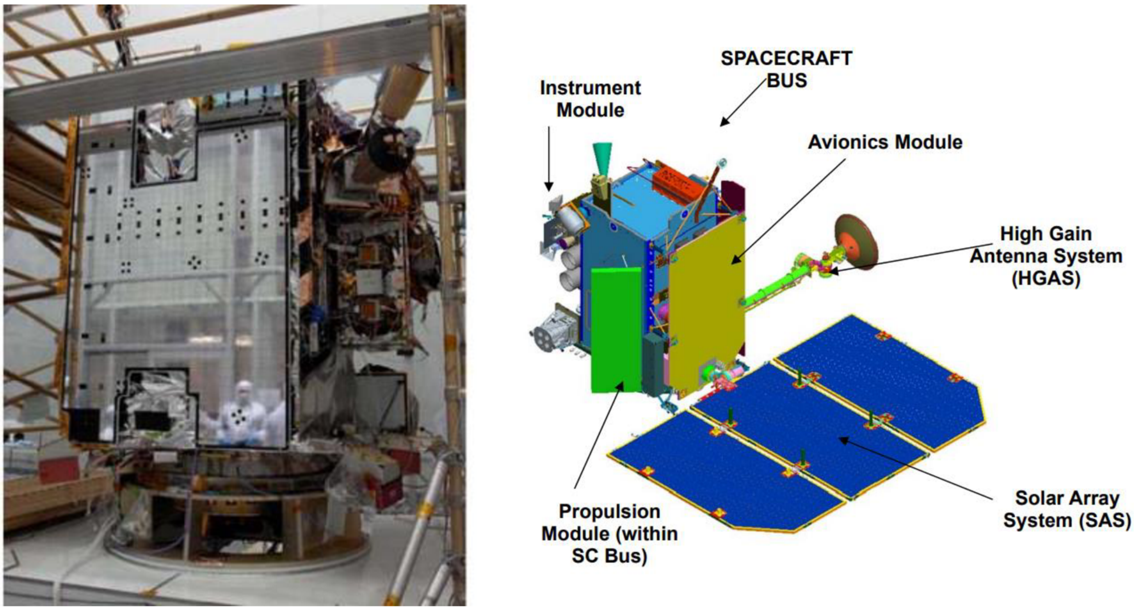


Figure 4-2 LRO radiator covered with OSRs (left) [23] and the modular design of LRO (right). [20]

The LRO uses a modular design with 5 different modules in total (see fig. 4-2). 2 modules, the High Gain Antenna System and the Solar Array System, are separate because the operate on dual axis gimbals and have a high dissipation. The High Gain Antenna System has an output of 40 W, 50 % is converted into heat. It is mounted on a long boom. It has blankets, radiators, and heaters around the gimbal actuators. The other 3 modules are in the bus. First, the propulsion module design is integrally coupled to the base of the spacecraft. It consists of an upper and lower tank, both thermally coupled to the central cylinder. Large heaters were used on the cylinder to provide a benign temperature for the tank interfaces and lines on and inside the cylinder. The titanium tanks were covered in low density heaters and overtaped with aluminum tape. The propulsion system was then radiatively isolated from the spacecraft with blanketing to keep the propulsion system from freezing. Second, the instrument module is located separately and itself has 4 thermally isolated instruments, since they operate at different temperatures. They use dedicated instrument radiators, some of which are angled or shielded by a sunshade. Third, the avionics module. All avionics are connected to one radiator and survival heater circuit via a network of heat pipes. This was done in order to reduce costs of testing and allow for more flexibility in the avionics. It must be noted that a paper on the thermal analysis of LRO advises to thermally couple TCS for redundancy, simplification of hardware and ease of testing. [22]

4.2 Venus

Venus is 40 million kilometers closer to the Sun than Earth. Heat input from the solar radiation at a Venus orbit is approximately 2.6 kW/m^2; twice larger than that at an earth orbit and 4 times that of Mars. Venus is entirely covered by clouds and therefore has a very high albedo of around 0.8, roughly 5 times Earth's value. The IR thermal radiation is less significant. The probes face challenges to design a more efficient method to *release* heat from the spacecraft into space and to control the heat generated by onboard equipment. The spacecraft are cold-biased, meaning that their TCS should limit the maximum temperature around Venus. The spacecraft might therefore get too cold during some phases, such as the initial flight, or when less instruments are operational, and the TCS must tackle it. [24] [25] [26]

Probes operating around Venus use 10^2 to lower 10^3 W of power. As an example, the Venus Climate Orbiter was supplied 1200 W from the solar arrays. It had a 300 W heater capability. In Venus orbit, it around 140 W/m^2 got inside the spacecraft, with approximately 1000 W of additional heat leaking through the optical cameras, and 500 W produced by the electronics. The radiator area was 5 m^2. [27]

The challenges for thermal control on Venus Express centered around its limited capacity to reject and store heat between pericenter science observations. This limitation was a result of the inherited design from Mars Express. Venus Express operated in a 24 hour highly eccentric orbit, which was a result of the thermal requirements. Under some scenarios, a cool-down of 22 hours was used in the design, with 2 hours for the next pericenter pass, which maximized science return during the pass. [19]

Surfaces in the shadow for longer periods of time cool down to cold temperatures, which might be a thermal control challenge sometimes, but it also presents an opportunity. Spacecraft tackle this by maneuvering and using heaters, since its TCS was well optimized for the Venus environment. The more restrained Venus Express and used positioning to its advantage. Its -X side was pointed away from the Sun most of the time, which created a cold side. The -Z side, where the main engine nozzle and the launch vehicle adapter interface reside, also must avoid illumination by the sun since these will absorb the solar heat very quickly. As a result, the +X side was illuminated most of the time. The spacecraft was kept in such position that the Earth-Sun plane was aligned with the X-Z plane. Thus, ±Y sides were in partial to full shadow much of the time too. The solar panels stemmed from here, so they were illuminated all the time. The Venus Climate Orbiter uses a similar approach. [26] [28] [29] [30]

The solar panels are made of a combination of solar cells and OSRs, giving it a striped appearance. The cells optimized for operation at higher temperature (e. g., using gallium arsenide instead of silicon. The cells must also withstand the temperature drop during eclipses. The rear side of the panels are completely covered with OSRs. [26] [29] [30]

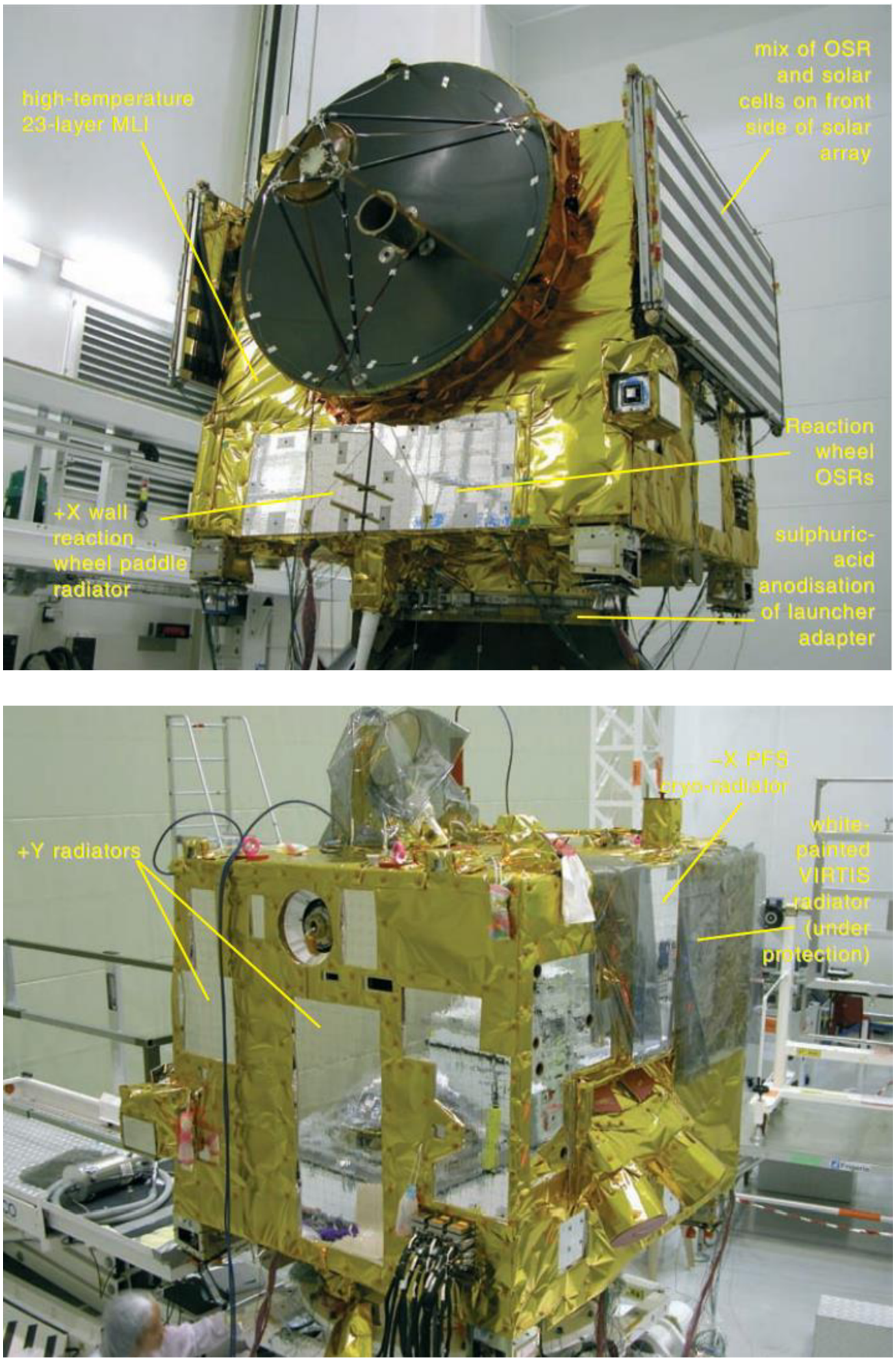


Figure 4-3 External components of the TCS of Venus Express. [28]

Radiators are placed on the shadowed parts of the shadowed parts of the spacecraft. In Venus Express, the ±Y panels were used for internal units. High-dissipation units are mounted directly behind the radiators in order to provide good conductive paths. Where dissipation was insufficient, the qualification temperature of some units was raised to cope with the hotter environment. Heat pipes were added under the Power Conditioning Unit and Power Distribution Unit to spread the high PCU thermal dissipation evenly. The colder –X was used for the instruments, including the cryo-radiators for the VIRTIS and PFS. Thermal straps connect the reaction wheels and those payload units with dedicated radiators. [28]

The HGA is fixed on the probe. When the spacecraft maneuvers to keep the radiator sides from being exposed to the Sun, it may temporarily loose contact with Earth. The Magellan spacecraft doesn't have cold sides. Instead, it has 10 electronics bays with louvered radiators. It also helps tackle the variations in the output of electrical instruments. The louvers are operated by bimetallic actuators. The louvers are covered with OSRs. The ±Y have shunt radiators, also covered with OSRs. [26]

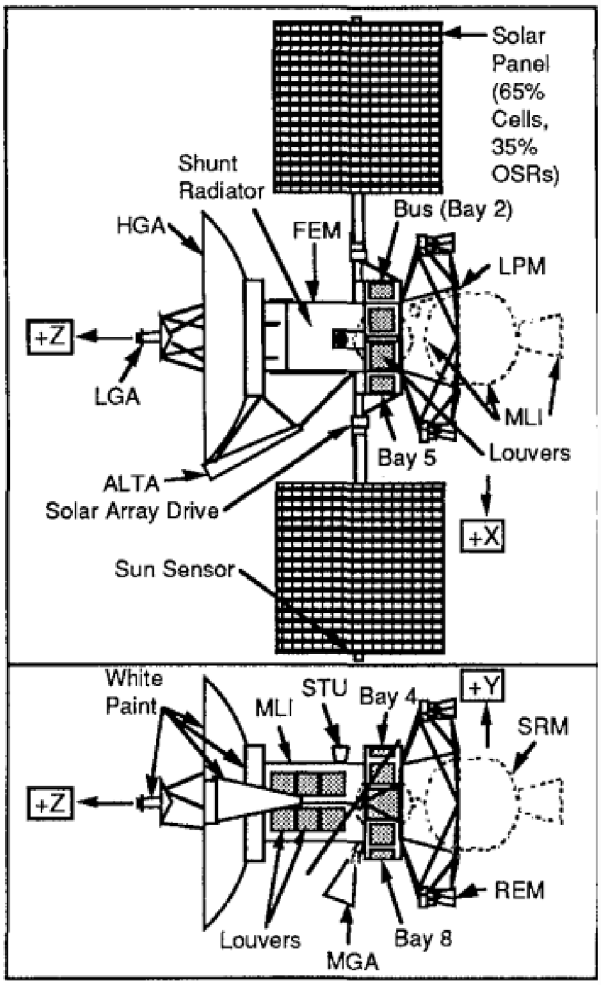


Figure 4-4 Magellan spacecraft configuration. [26]

The parts of the spacecraft which are not covered with radiators are insulated with **MLI** which uses as low as possible solar radiation absorption and low ageing coatings. To avoid multiple reflections and to diffuse the direct sunlight, most of the blankets have an embossed external layer. On Venus Express, the MLI has up to 23 layers, with no spacer material on the external layers for better resistance at higher temperatures. Care was taken with the fitting overlaps to avoid damage through Sun-trapping. [19] [28]

The outer layer of all external blankets on Magellan is a material called astroquartz. It is similar to glass-fiber cloth but is better able to withstand intense solar 0radiation. In fact, chemical binders normally used in astroquartz to control flaking had to be baked out when tests showed that the light intensity at Venus could discolor them and eventually cause a buildup of heat. [26]

Internal MLI blankets are used to insulate more temperature sensitive equipment, like batteries, or high dissipation equipment, like the HGA. Black paints are used in the interior to minimize thermal gradients. [26]

The net effect of the radiators, MLI and OSRs makes the spacecraft tend toward cold temperatures rather than hot. To assure that some cold-sensitive components do not become too cold, flexible electrical heaters have been installed inside housings or wrapped around such fixtures as the solar-panel articulation bearings. Heaters are used to prevent excessive cooling of units, structures and propellant during the cold phases and eclipses, payload non-operation (partial or total) and safe mode. They are also used when the payload is fully operational in order to comply with the units' minimum temperature limits. [19] [26]

The heating system on Venus Express consists of 16 nominal heater lines and extra 16 identical redundant lines. Almost all the heater lines are controlled using bimetallic thermostats with fixed temperature set points in order to allow a large number of controlled areas. Two thermostats are used in series to avoid closed circuit failure. [19]

The Magellan spacecraft heaters were designed for one AU distance from the Sun with closed louvers. There are laminate patch heaters, the largest being the Solid Rocket Motor, the radar and the battery heaters. The heaters are redundant using separate patches except for the propellant lines which share a common patch but have a redundant tape overwrap. The purpose of the separate patch or redundant overwrap approach was to prevent a single bondline failure from causing the loss of both heaters. Most of the heaters are controlled using quad redundant (two thermostats in series in each of two parallel circuits) mechanical thermostats with arc suppression. The exceptions are the radar and the batteries which use temperature sensors, relays, and software to open and close the relays. Any bank of heaters can be disabled or commanded off if necessary. [26]

4.3 Mars

Mars is the last planet in our Solar System near which a spacecraft will experience significant environmental heating. The average solar flux is 589 W/m 2, or about 42% of what is experienced by an Earth-orbiting spacecraft, and about a quarter that of Venus. As a result of the eccentricity of Mars' orbit, however, the solar flux at Mars varies by ± 19% over the Martian year, which is considerably more than the ± 4 % variation at Earth. [31]

Mars' thin, relatively cloudless atmosphere is highly transmissive to IR. This condition contributes to the cold nighttime surface temperatures and causes nightside planetary IR to be much lower than that on the dayside. Mars presents a relatively stable environment without extreme temperature differences or spikes. [31]

Most of the spacecraft units are collectively controlled within a thermal enclosure. For more demanding units like payload sensors, special precautions are taken by individually insulating them and by providing them with dedicated radiators to ensure the correct temperature levels. External units (antennas, mounts) are insulated from the spacecraft, as they have to withstand greater temperature ranges than the others. Power ranges at Mars are in the order of 10^2 W, up to $1x10^3$ W. As much as 50 % is used to run the heaters on different parts of the spacecraft. During the cruise phase, a similar amount of heat is *rejected* in the beginning, since the probe receives significantly more heat from the Sun. [19] [32] [33]

The design approach is essentially passive, relying on surface coatings, conductive or insulated mounting schemes for units, radiative surface properties, dissipative doubler mounting plates and multi-layer insulation where appropriate. This is supplemented by electrical heaters split into various types of circuits and control schemes, and with heat-pipes on some high-dissipation equipment. The most demanding payloads with very tight temperature requirements, such as optics and cryogenic cooled detectors, are provided with dedicated cryo-temperature (<90 K) radiators connected to the instrument units with dedicated cold fingers. [19] [34]

Electrical heaters are required to attain minimum required operating temperatures, start-up temperatures, thermal gradients (e.g., within propellant tanks), or to compensate for major dissipating equipments when not in use. These are regulated and controlled by a combination of hardware thermostats and on-board software-controlled switches. [19]

The heater system on Mars Express comprises 14 redundant heater lines. The prime heater power installed is about 400 W for the bus and 200W for the payloads (note that the maximum power generated by the solar panels is 650 W at Mars). The average worst-case heater demand is about 270 W for Safe Mode worst cold-case environment. Platform electrical heaters are designed at spacecraft level to provide the required heating for platform units when not in operation, to compensate for lost dissipation of equipment, or to compensate for changes in the thermal environment. Some spacecraft heaters are also installed to control or meet specific payload instrument internal unit temperatures, including the camera (tight tolerance of 10°C). [19]

Externally mounted equipments including antennas, booms, some payload hardware are thermally decoupled from the spacecraft and insulated, with dedicated radiator areas where needed. [19]

4.4 Asteroids and Comets

Probes visiting asteroids must handle the diminishing solar radiation as they travel far from the Sun. The asteroid belt is located between 2.2 to 3.2 au from the Sun, but some spacecraft, such as Rosetta, which rendezvoused with a comet, reached as far as 5.25 au. At these distances, spacecraft TCS are optimized to preserve heat. However, they often take multiple orbits before reaching their destinations, which brings them closer to the Sun multiple times.

Probes approach the asteroids or comets from distances where the target does not cover a significant portion of the view. The albedo and thermal radiation, and the obstruction of the radiators' view to deep space, is not a concern unless the spacecraft plans to obtain a surface sample. That was the case of OSIRIS-Rex, but even during this event, the spacecraft's components remained within allowable temperature ranges without additional changes to the TCS. [2]

For these missions, the design driver is mostly the cold case. In order to minimize the heater power, leakage to space and radiator areas are reduced as much as possible. Therefore, the radiator thermal louvres, which close during the cold phase, are pivotal to the design. These TCS designs often blur the line between the Inner Solar System, which are optimized for hot and cold cases, and the Outer Solar System, which is mostly concerned with preserving heat. [35]

When the Rosetta spacecraft reaches 4.5 au from the sun, the deep space hibernation mode is entered. During this mode, all but the essential equipment is turned off, giving rise to the minimum dissipation case. The aphelion point of the Rosetta mission occurs at the maximum solar distance of 5.25 au, where the value of the incident solar flux on the spacecraft is only 3.7% of the value at 1.0 au. This is the cold design case, for which the heater power required by the TCS will be a mission maximum. [35]

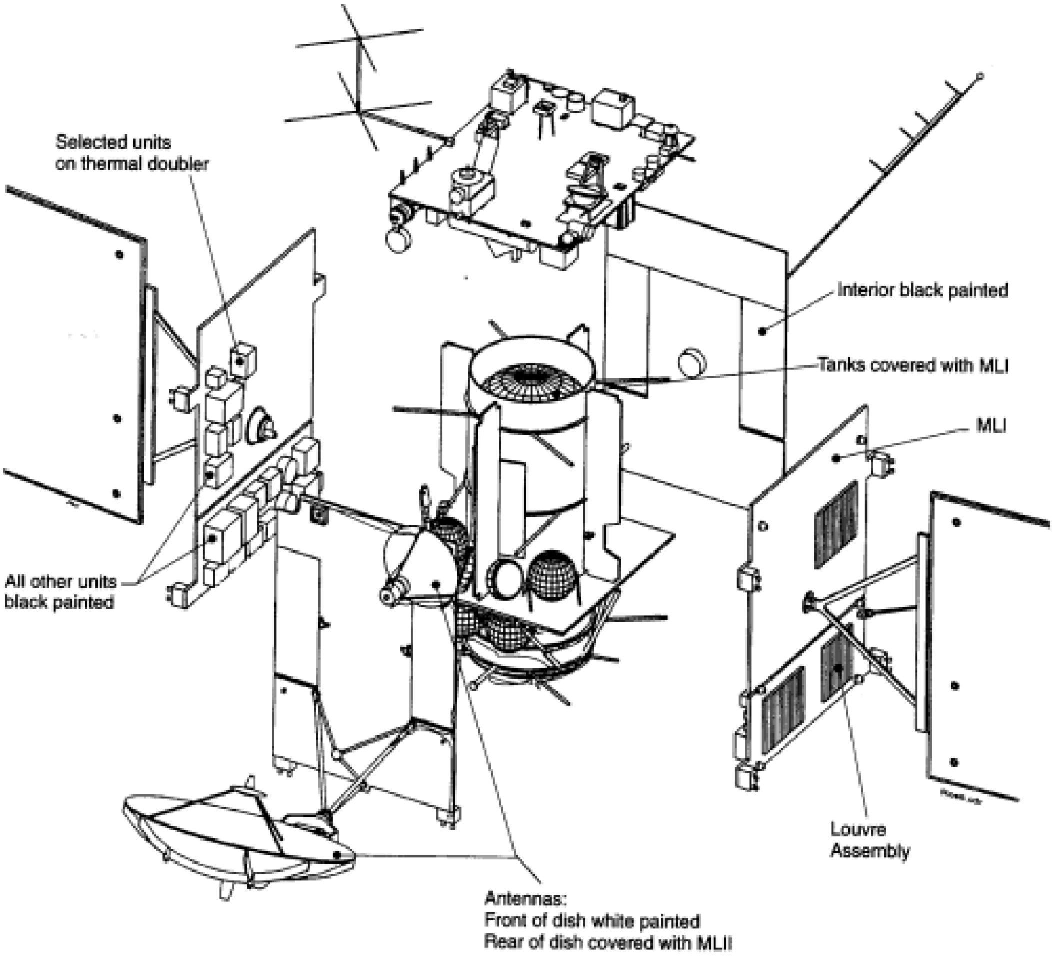


Figure 4-5 Rosetta spacecraft configuration. [35]

The Psyche spacecraft is a stripped-down version of the SSL 1300 geostationary satellite bus from SSL with the large telecommunications payload removed and with some thermal control enhancements to support the range of solar distances (1.0 to 3.3 AU). Psyche's thermal design is greatly simplified compared to SSL's typical high power GEO communications spacecraft. The elimination of approximately 8 kW of typical thermal dissipation results in the simpler and lighter spacecraft. As a result, Psyche's thermal loads are comparatively modest, generated by relatively low power bus units, which are easily accommodated on a simplified version of SSL's smallest radiator panel through the removal of thermal control hardware and OSRs. Psyche's spacecraft structure is a rectangular panel-box construction, supported by a core central cylinder with a xenon tank placed inside it. This assembly provides an optimal configuration for the thermal control of the xenon propellant, providing a unified thermal mass with common thermal blanketing that surrounds all of the tanks and the central cylinder. [36]

In a similar fashion to Psyche, and most Inner Solar System probes for that matter, large spacecraft are box shaped with a central cylinder and radiator panels on the sides. High dissipation equipment is placed on those. Psyche has a dedicated thermal cold zone separated from the rest of the heat pipe matrix for demanding instruments. Dawn has the components along the walls (Y axis), which are made with conductive aluminum covered with MLI, which radiates heat into space. These wall compartments are filled with heat pipes, including a few redundant. [37] [38]

Psyche's thermal design also uses thermal louvers to keep the spacecraft from becoming too cold when far away from the Sun. Replacing the electric heaters typically used on GEO spacecraft, thermal louvers are passive mechanical shutters placed over OSRs that open to provide a clear view to space when warm and close to limit radiative cooling when cold. [36]

On Rosetta and Dawn, thermal louvres are mounted over OSR radiator areas on the spacecraft ±Y walls. These provide the main spacecraft heat rejection capability during hot mission phases and reduce the heater power demand in cold environments. The temperature differences caused by the ion thruster firing or not is regulated by a louver too. Louvres are also used to minimise the heat loss in deep space hibernation but will open to ensure adequate heat rejection during cometary operations. [35] [37]

The heater power was approximately 40% of the total during the critical deep space hibernation phase on Rosetta, which reaches 5.25 au. In comparison, on Dawn at 3 au, the active Thermal Control System demands around 200 Watts of power, about 15% of the spacecraft's total power supply. [37] [39]

The component with the tightest thermal requirements is the spacecraft battery which uses conductive and radiative isolation along with heaters that are actuated based on temperature readings. A large number of heaters and temperature sensors are used. The tanks and propellant lines are lined with heaters and wrapped in Multilayer Insulation to keep propellants from freezing. [37]

The active thermal control elements consist of two groups: thermostatically controlled heaters for autonomous mission modes and software-controlled heaters for operational mission modes. The heater systems use a redundant architecture to maintain minimum temperatures on components. Most heaters are software controlled based on thermostat readings from the associated heater zones. Thermostatic backup heaters are used to maintain minimum temperatures in the event of a switchover of the Central Electronics Unit as a result of Fault Protection. For spacecraft hibernation modes, it is necessary that the heater circuits operate without any external intervention. This approach uses thermostats for the redundant heater circuit to automatically switch on or off as required. [35] [37] [39]

External non-radiator, non-equipment surfaces are covered in MLI blankets. On Psyche, MLI consists of 15 layers plus a Beta Cloth that provides the primary protection against micrometeoroids. [36]

The thermal testing of MLI and its edge effects should not be underestimated. Small simplifications in the model lead to large inaccuracies. It was found that key thermal hardware performed better when tested at component-level than when fully integrated within the TCS on the spacecraft. Hence it is important to maintain margins through the program. Heat losses through couplings is often underestimated. An interface filler is used between electrical units and the structural substrate to ensure adequate and predictable conductive heat flow. Sigraflex graphite foil was used for this purpose on Rosetta. [35] [39]

5 Orbits Below 0.5 au

As stated in chapter 1.1, the intensity of the solar heat flux follows the inverse-square law, thus increases four times when the distance halves. Three different mission phases must be addressed: outer cruise, inner cruise, and Mercury orbit (if applicable). The highly varying thermal environments of these orbits are a key engineering challenge and play a prime role in the spacecraft design, in contrast to most designs in other chapters, where thermal control is only one of many factors. Probes diving below 0.5 au are equipped with significant heat shielding. [1]

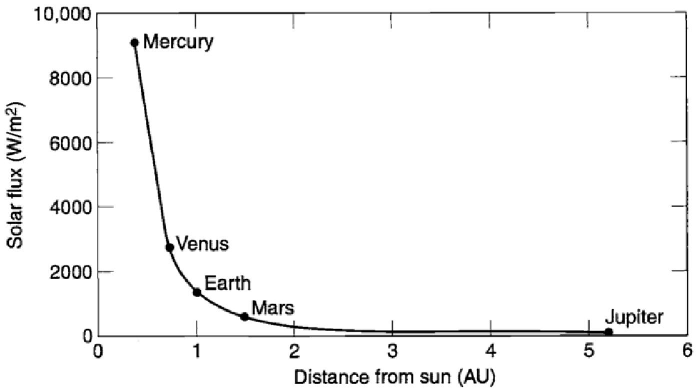


Figure 5-1 Solar flux as a function of distance from the Sun. [4]

Every spacecraft begins its mission at 1 au, where Earth is located. As getting close to the Sun requires a lot of energy, they typically use several Venus and Earth gravity assists. The parts of the mission when the spacecraft experiences relatively colder environments are known as the outer cruise.

During the inner cruise, when the spacecraft gets close to the Sun, it will experience a sun flux in the order of 10^5 W/m^2. This is the most distinctive and influential part for the design. This is also where most of the mission objectives are performed. [40]

Spacecraft orbiting Mercury must take into consideration not only the exceptionally wide swings in temperature, but also that it can be exposed to heat from several different directions. Mercury has a high excentricity, with a perihelion of 0.31 au and aphelion of 0.47 au. The high surface temperatures are driven by its proximity to the sun and generally low albedo. The rotation is so slow that the surface temperature of the side of the planet facing the sun reaches up to 423 °C, while the dark side is quite cold at -173 °C. A spacecraft orbiting at 0.1 planet radii on the day side is exposed to as much as 14.5 kW/m^2 of solar radiation and 12.7 kW/m^2 of planetary IR, while on the dark side it receives little to no radiation. It can be addressed either by adapting the design or the choice of orbit (e.g., elliptical, polar). [41] [42]

Spacecraft	Type	Configuration	Perihelion (au)	Launch Year
Mariner 10	Mercury flyby	3-axis stabilized	0.46	1973
Helios	Solar probe	Spin-stabilized	0.29	1976
Messenger	Mercury orbiter	3-axis stabilized	0.30	2004
Parker Solar Probe	Solar probe	3-axis stabilized	0.046	2018
BepiColombo	Mercury orbiter	---	0.31	2018
Solar Orbiter	Solar probe	3-axis stabilized	0.23	2020

Figure 5-2 Examples of spacecraft with orbits reaching below 0.5 au. Note that BepiColombo is comprised of MPO, MMO and MOSIF. [43][44][45][46][56][59]

5.1 Spacecraft with Sunshades

During the inner cruise (and in Mercury orbit), a three-axis stabilized probe can take advantage of a **sunshade** to protect the vehicle from the intense solar environment. It shields most spacecraft components from direct solar exposure and allows them to operate at conditions typical of other interplanetary spacecraft without much temperature variation. As an example, the sunshade on Solar Orbiter is subjected to around 10^5 W of solar radiation, which is reflected or reemitted, so only 8 W of those reach the spacecraft. The spacecraft attitude control points the sunshade toward the Sun at all times when the spacecraft-Sun distance is less than approximately 0.95 au. To reduce the heater-power requirements during the outer cruise, the spacecraft is turned around such that the body is pointed toward the Sun, thus increasing the solar flux. Nevertheless, all spacecraft are equipped with heaters. [40]

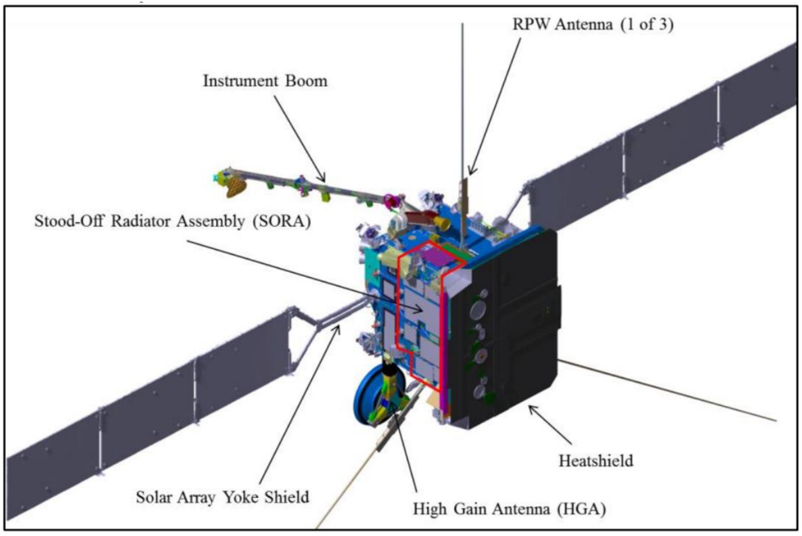


Figure 5-3 Overview of the Solar Orbiter Spacecraft, a typical example of a probe in this category. [source]

Since the sun covers several degrees in the sky and the rays are not parallel, the sunshade extends beyond the body of the spacecraft to properly cover it. The heat shield may contain openings for the instruments. It is separated from the body by a gap for additional insulation. The design of the sunshade varies from mission to mission as a result of carefully balancing the cost, complexity and new materials and technologies. The degradation of the material is an important factor in the selection, as it will be constantly impacted by intense heat, UV and other high-energy radiation and particles of solar wind. Generally, a multi-layer sunshade with low thermal conductivity and high emissivity is used, which also channels thermal IR radiation outwards. [40] [43]


Figure 5-4 Comparison of various sunshades. From left to right: Mariner 10 [44], MESSENGER [43], Parker Solar Probe [45], Solar Orbiter [46]

The solar panels, which are outside the thermal shield, are designed to survive Sun incidence. Solar panels thermal control is performed by tilting the panels with increasing solar flux. In the outer phase, the solar array is orientated perpendicular to the radiation flux. As the spacecraft approaches the Sun, the panels are tilted at an angle to limit their effective area and increase reflectivity. While this limit in exposure would decrease the power output, it is compensated by the higher flux, and the lack of need for heating power. The size of the solar array depends mostly on the distance from the sun and power requirements during the outer phase. [40] [43]

The solar panels consist of alternating rows of solar cells placed between mirror-like Optical Solar Reflectors (OSRs). The OSRs are placed in such way that it reflects light away from the spacecraft body and its instruments. The point of contact between the body and the solar panels is hidden behind the sunshade. Just like the heat shield, the materials of the solar cells are subject to significant degradation. [43] [47]

On the Parker Solar Probe, which got closer to the Sun than any other vessel before, the solar panels tilt backwards behind the sunshade. They are equipped with a secondary section, which is tilted 10° outwards, and stay in the penumbra of the heat shield. Additionally, they contain plumbing which pumps heat from the solar panels to the radiators. [48]

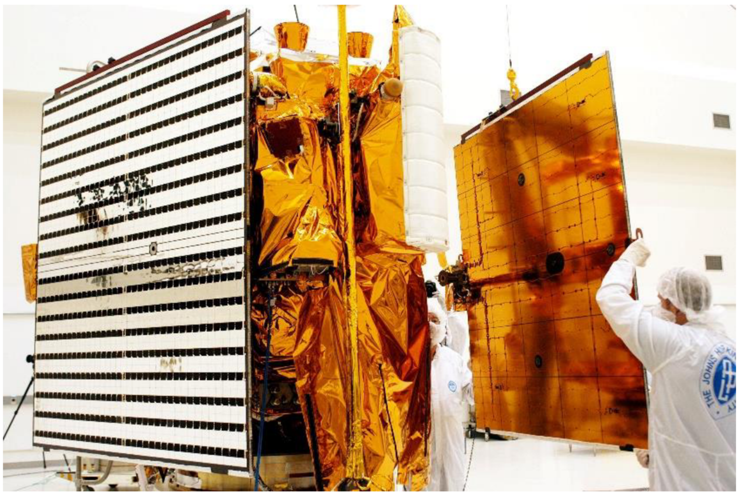


Figure 5-5 Messenger's solar panel installation. Note the alternating rows of solar cells (dark) and OSRs (metallic white). [49]

Radiator panels tackle the heat generated by the instruments as well as the heat leaking through the sunshade. They are always hidden from direct sunshine during the inner cruise, but otherwise they are placed in various spots, depending on the mission needs. For instance, Messenger has a dedicated radiator for every instrument with a power of 20 W or more. [50]

On Solar Orbiter, several small radiators are located next to the instruments, because they get direct sunlight through the feedthroughs in the heat shield, and each has its operating temperature. [50]

When Messenger passes between the Sun and Mercury, which takes approximately 30 min, the sunshade protects the spacecraft from direct solar radiation. Sensitive instruments, such as the battery and star trackers, are hidden from the planetary radiation behind the radiators and other less sensitive parts. When the radiators get too hot, diode heat pipes stop conducting, and restore normal thermal control when the radiators cool down. Careful orbit geometry also helps with thermal control in these phases. [50]

Parker Solar Probe has four large radiators located just behind the heat shield. Water is pumped via **two-speed pumps** through the probe to the solar arrays, which are the first to utilize liquid-cooling. This extraordinary system is used because other passive methods stated above were not enough in this case. The cooling system and radiators must tackle 5900 W of heat during perihelion, coming from the solar panels alone, as the rest of the spacecraft is protected by the sunshade (which is subjected to 10^6 W). The system is equipped with a heated accumulator to prevent water from freezing in the first stages of the mission. In addition, there are several louvers and heaters for the instruments. [48] [51] [52]

Appendages, i.e., instruments outside of the sunshade, are Sun-exposed components that required non-standard thermal design and specialized construction. They are either covered by special MLI, or made of heat-resistant materials, such as niobium alloy. They have to be painted and positioned in such way that they minimize the reflected and thermal heat towards the spacecraft. [53] [54]

A conventional **Multi-Layer Insulation** (MLI) can be used for most of the spacecraft surfaces, as the sunshade protects most of the spacecraft and the reflected solar flux from the solar array is minimal. Very close to the thruster nozzles, a high temperature MLI is required. The rest of the body is only subjected to high temperatures during positioning for maneuvers. The MLI must have an electrically conductive outer skin to prevent charge build up. [47] [54]

5.2 Wrapped Spacecraft

The Mercury Planetary Orbiter (MPO), part of the BepiColombo mission, is quite different from other from the other designs for three-axis stabilized probes this close to the Sun. It does not have a primary sunshade. Instead, its body is as compact as possible, **entirely covered with MLI** in with 3 to 4 layers instead of the usual 1 (however note that MLI, as the name suggests, is itself comprised of multiple layers). This protects the MPO from heat coming from the Sun and Mercury in different directions, however, it also prevents internally generated heat from escaping. To protect the instruments from this as well as parasitic heat piercing through the MLI, the spacecraft is equipped with 97 heat pipes. A downside of this design is that 80 % of the equipment had to be developed from scratch. Another reason for this unusual configuration was that the MPO had to be stacked together with the other parts of BepiColombo. [55]

MPO’s **large radiator** is always pointed away from the Sun towards outer space, although receives intense albedo and infrared from Mercury itself. To minimize the influence of this parasitic heat flux, highly reflective fins have been mounted to it at an appropriate angle, while at the same time allowing radiation towards deep space. The radiator is divided into separate segments, allowing the generation of different interface temperatures and temperature stabilities. As stated above, the spacecraft is equipped with 97 heat pipes connecting the radiator to all other instruments. [42]

The radiator handles 1200 W of internally dissipated heat and 300 W of parasitic heat leaking through the MLI. Heaters are used in the coldest phases of the Mercury year. The solar panels are designed in the same manner as in chapter 5.1. [42]

5.3 Spin-Stabilized Probes

Both the Mercury Magnetospheric Orbiter (MMO, or Mio) and Helios are controlled by a **passive thermal design** combined with an actively controlled heater system for the coldest phases of the mission. The nominal spin rate of Mio is 15 rpm (spin period of 4 s) due to the scientific requirements. The spin axis is pointed nearly perpendicular to the Mercury orbital plane. The complete spacecraft is kept inclined at 9 degrees for the reduction of the sunlight reflection to the lower deck. During the cruise phase, it is placed within the MOSIF heat shield, because Mio is not rotating yet. Helios spins at 60 rpm with its axis pointed perpendicularly to the orbital plane. [42] [58]

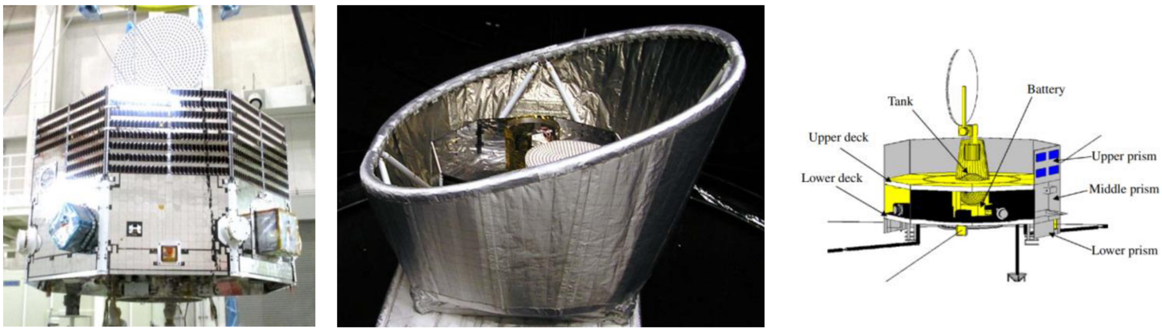


Figure 5-6 Mio (left and right). MOSIF with Mio installed (center). [56]

Both spacecraft bodies are divided into three sections. The central section contains most components and is insulated from the shell by MLI and thermal standoffs. Most of the outer surface is **covered with OSRs**, or a combination of solar cells and OSRs. The upper and lower sections form walls which protect the upper and lower decks from sunlight. Helios’ walls are covered with fins to prevent reflection towards the decks, and the walls themselves are tilted for better reflectivity, so that the solar cells do not overheat. The OSRs reflect 90 percent of the heat. [42] [57] [58]

The internal surfaces of Mio’s upper and lower deck have high emissivity surfaces (black paint) to equalize internal temperature. The batteries and nitrogen tank are controlled independently with the aid of radiators, additional MLI and heaters. The lower deck acts as a radiator. Helios is equipped with radiators and bimetallic-controlled louvers to further regulate the internal temperature, located on the upper and lower deck. [42] [57] [58]

Appendages, such as Mio’s de-spun antenna, are coated or painted to reflect as much sunlight as possible, however, thermal control on these less regular, complex shapes is more difficult. Therefore, they are built to withstand temperatures of several hundred °C. [42] [57] [58]

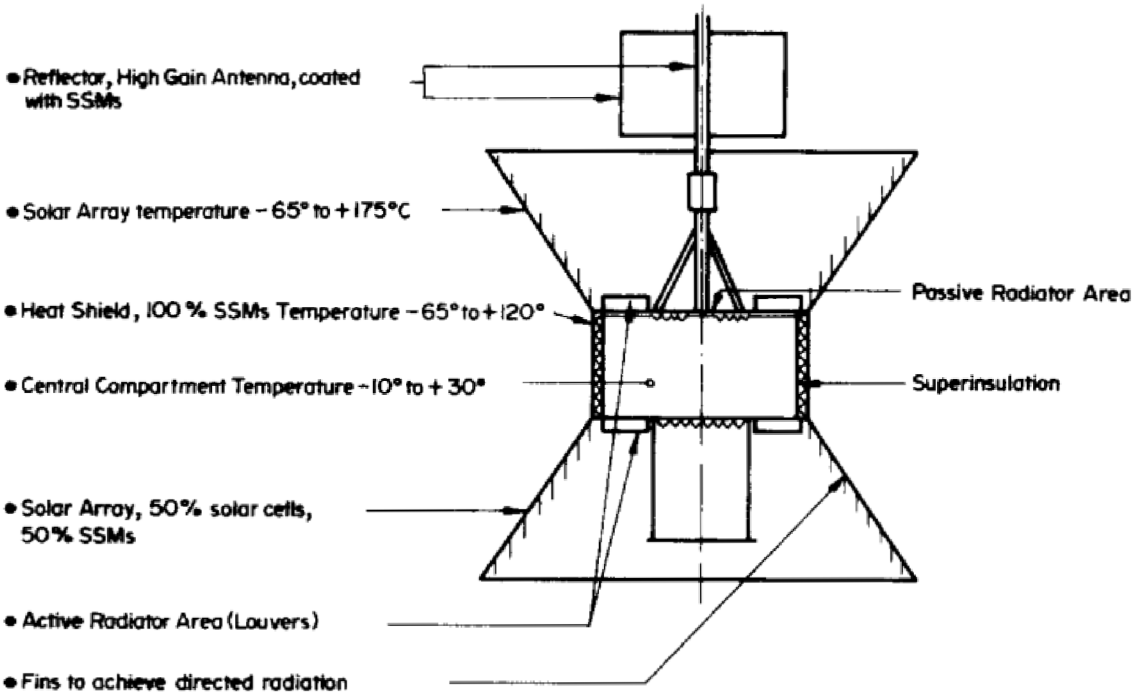


Figure 5-7 The TCS of Helios. [59]

6 Outer Solar System

In a similar fashion to chapter 5, the inverse square law and solar irradiation play a significant role in the thermal design of spacecraft designed to fly beyond the Asteroid Belt (spanning 2.2-3.2 au) into the Outer Solar System. In this case, the lack of solar heat flux creates a need for sufficient thermal insulation, but this is not the main driver behind the spacecraft design. Most importantly it changes the way of obtaining power.

For the most part in history, the distance from the Sun meant that solar panels could not be used, and thus a Radioisotope Thermoelectric Generator (RTG) was used instead. As this produces an excess amount of heat, in inherently changes the design of thermal control system. In recent years, spacecraft flying to the orbit of Jupiter (be that Jupiter itself or the Trojan asteroids) have started using new generations of solar panels, which together with new types of instruments, lead to a new trend in thermal control systems layouts. [60]

Another thing that influences design is the initial flight trajectory and the use of gravity assists in the Inner Solar System. While this is usually solved by calculated positioning of the spacecraft, additional insulation must be used if the spacecraft utilizes assist from Venus. These spacecraft are optimized for cold, so dives closer to the Sun raise the maximum expected temperature. [60]

Planetary radiation at these distances does not cause any significant strain that could not be tackled with clever positioning of the spacecraft, although it must be considered in the mission design for sensitive instruments, like cryogenic radiators. Jupiter in particular has a strong magnetic field which puts constrains on electronics but doesn’t influence the thermal properties and management. Overall, the cold case is much more important in the design than the hot case, and spacecraft are mostly concerned with keeping things warm. [60]

Planet / body	Solar distance (au)		Mean value (W/m^2)
Jupiter	5.2	Direct solar	51
		Albedo	0.343
		Planetary IR	13.6
Saturn	9.5	Direct solar	15.1
		Albedo	0.342
		Planetary IR	4.6
Uranus	19.8	Direct solar	3.71
		Albedo	0.343
		Planetary IR	0.63
Neptune	30.0	Direct solar	1.51
		Albedo	0.282
		Planetary IR	0.52
Pluto	39	Direct solar	0.88
		Albedo	0.47
		Planetary IR	0.5

Figure 6-1 Heat intensity values for Outer Solar System bodies. [61]

6.1 Spacecraft With RTGs

In spacecraft with **Radioisotope Thermoelectric Generators,** the thermal control system is closely linked with the power system. RTGs are a logical solution to the lack of solar irradiance in the Outer Solar System, as they are independent from it. Moreover, they are passive, steady state operating, and have an inherently high-radiation tolerance. RTGs have been used for decades and thus has lower risks, because it is a proven technology. Their disadvantage is heavy weight, having a typical electrical power output of 2.8 W/kg. [62] [63]

Conveniently, RTGs produce excess heat, which is used in the spacecraft's thermal control, and therefore removes the need for additional heaters. However, the efficiency of RTGs used for these missions is around 6 %, so they produce too much heat, which must be dissipated into open space, and the RTGs must be insulated from the rest of the spacecraft. The power output declines with time; the Pu-238 used has a half-life of 87.7 years, which results in a loss of about 15 % for an 8-year mission. [62] [63] [64]

Part of the electrical power is used for powering the heaters anyway, because of the constrains of the thermal control system. When there is low demand for electrical power, it is dissipated into space as heat by resistors. [62] [65]

The RTG is supported by a low thermal conductivity titanium support structure. Light heat shields keep infrared radiation from hitting the spacecraft's body. The temperature of the RTG is around 250 °C at the beginning of the mission and remains for the whole duration. An additional benefit of the RTG placement is that the heat from the RTG can be utilized to keep the propellant from freezing. [64] [65]

The proposed Europa orbiter uses a cooling loop with pumps to distribute the RTG heat and increase efficiency, however, this is unusual. Alternative radioisotope generator designs with higher efficiencies have a significant design impact on the thermal control systems. An example would be the Stirling Radioisotope Generator, which operates at 50 °C with and efficiency of 22 % and eliminates the need for shielding the spacecraft from it. It would use a cooling loop with heat pipes to further distribute heat. Unfortunately, it has a lower radiation and vibration tolerance. The additional shielding and structural changes lead to a higher overall mass. It has not been used in Outer Solar System missions so far. [62] [66]

The whole body of the spacecraft is **covered in MLI**, which creates a stable environment that is less susceptible to outside heat sources and can be treated as a whole system, instead of individual components. It also protects against micro-meteorites. Internal joints enhance thermal conductivity, while external joints impede conductivity. The hydrazine tanks are thermally coupled with the body since they operate within the same temperatures, and additionally protect electronics from excess heat. The compact design with few appendages is especially noticeable in the New Horizons spacecraft, which flew to Pluto. It tries to minimize heat loss in the cold environment and the required power, and thus the mass of the RTG. [67] [68]

produce excess heat and require **radiators** placed near it. They are also placed near instruments that need a colder temperature than the rest of the body. **Heaters** are placed around sensitive instruments and components such as valves in the propulsion system, which must not freeze. The locations of the specific heaters are less important than the total power input to the system when they are part of the MLI covered body. For external components, such as the thruster clusters on Cassini, **Radioisotope Heater Units** are used. [64]

In case of New Horizons, the heater system is independent of the spacecraft computer and can operate even in case of spacecraft safe modes. Generally, internal electronics produce most of the necessary heat to maintain operational temperatures. **Louvers** are used to further control the internal temperature of the body as a whole, rather than individual components. They are open most of the time in the inner part of the Solar System and rarely open in the outer regions. [64] [65] [67] [68]

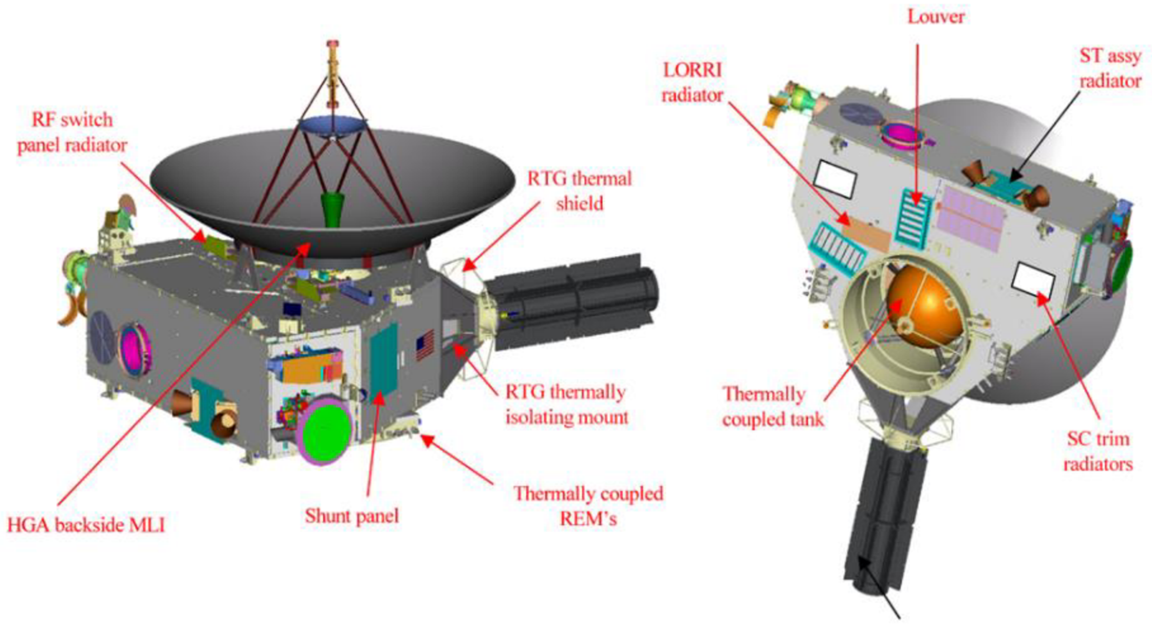


Figure 6-2 Key thermal design features of the New Horizons spacecraft [67]

The **High Gain Antenna** (HGA), which is necessary for communication with Earth from these distant regions, is large in diameter, painted white and insulated from the body with MLI. Thanks to that, it can be used as a sunshade during the inner parts of the flight path, when low gain antennas are sufficient. The insulation of the antenna also prevents heat leaks while in the Outer Solar System. There is often additional solar insulation around the HGA. [64]

The spacecraft do not have any additional protection against planetary radiation from the planets or the moons during flybys. Equal temperature distribution is achieved by changing the positioning the spacecraft, or its rotation in case of spin stabilized ones. [64] [69]

Appendages are generally passively controlled using appropriate surface finishes alone, or in combination with insulation. [68]

6.2 Spacecraft With Solar Arrays

New developments in solar cell technology have allowed their use in missions reaching as far as the orbit of Jupiter (5.2 au). These cells are optimized for low-intensity / low-temperature conditions. Since the solar irradiation this far is around 4 % of the value on Earth, the panels are very large (e. g. 27 m on JUICE). The first to use it was Rosetta (see chapter 4.4), although it only reached beyond the orbit of Jupiter in the hibernation phase. [69] [70]

Despite the increased weight of solar panels compared to plutonium-powered generators, the vehicles' masses are still within acceptable launch limits. It is cheaper, eliminates the need for an environmental impact statement that is required for nuclear-powered spacecraft, and does not rely on constrained supplies of plutonium-238, the isotope used in RTGs. The lack of the RTGs excess heat also means that more power is required to keep the spacecraft warm – about half the power is used for that. [71]

Unlike RTGs, however, ionizing radiation can damage solar panels. The orbiters are able to use solar power because its high-inclination polar trajectory enables it to avoid most of the high radiation zones that are concentrated over Jupiter's equator. The orbits pass through Jupiter's intense magnetosphere, which gradually degrade the solar panels as the mission progresses. [70]

Temperature control on the flight path using the HGA is identical to the one in chapter 7.1. [69] [70]

A notable difference compared to the previous category is that they often lack a unified compartment with a global thermal control system. Most of the components are contained inside one MLI wrapping, which covers the spacecraft. But within that, they are kept within special compartments (so called vaults), which can each have their own temperature and small thermal control system. It is partially insulated from the others with thermal separators and has independent control software. These are a small instrument pen with heat pipes, heaters, and a radiator. The hydrazine tanks are also contained in a partially independent section with MLI. [69]

Since these designs have all been developed within the last few years, it is likely that latest generation, more precise instruments require more extreme operating temperatures, so they must be mounted more independently (note that most instruments in the previous category operated at a common temperature in the range of 0-50 °C). Gimballing capabilities on independent mounts allow the HGA to be pointed at Earth at all times while performing science. The lack of RTG heat, which was used to heat the fuel tanks, means that they require additional insulation. Finally, none of the solar powered spacecraft fly past the orbit of Jupiter, so they do not require the most conservative heat control.

However, some spacecraft use a unified compartment, such as Juno. It still needs dedicated insulation and heating for the fuel tank, and has several instruments housed separately, but most components and instruments are contained together, thermally linked within the spacecrafts body. Since the body serves as a radiation shield, it likely led to the use of this unified thermal control system, as the components must be already enclosed together. [70]

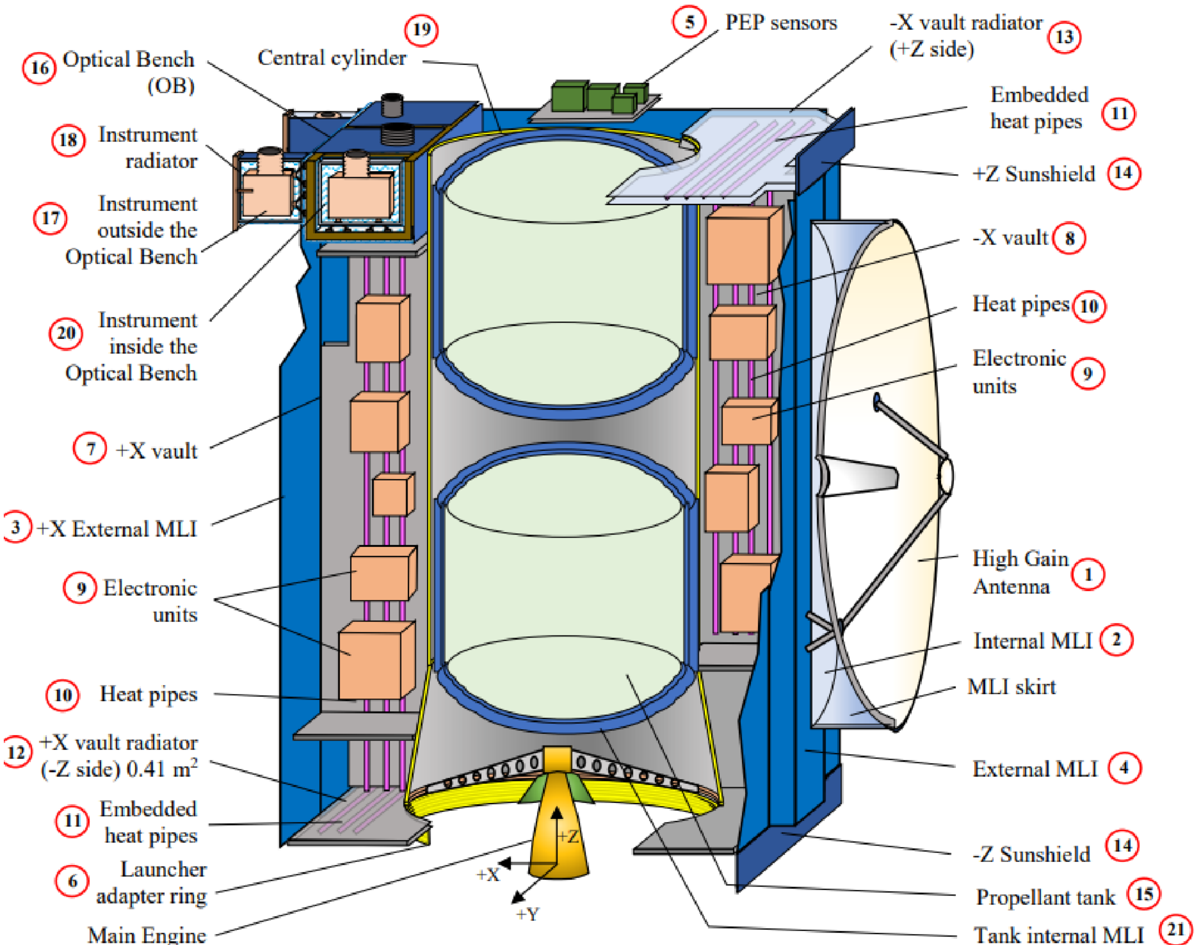


Figure 4. JUICE spacecraft thermal architecture.

Figure 6-3 Figure 6 3 JUICE thermal control system overview. Notice the compartments, so called vaults. [69]

7 Miniature Satellites

In most TCS designs in the previous chapters, their most defining characteristics were linked to the environments in which they operated. In contrast, the TCSs of miniature satellites is driven primarily by their small size. Their flight path plays a lesser role, yet not negligible. Nevertheless, due to technological limitations, no miniature satellites have flown in orbits below 0.5 au or in the Outer Solar System; at least not for more than a few hours, which would call for a dedicated TCS. [72]

The line between “regular” probes and miniature ones, from a thermal control perspective, is not completely clear. Especially in LEO, larger satellites have TCSs that share many similarities with miniature satellites. Generally, miniature satellites have the equipment and instruments more densely packed, which requires more focus on heat transfer solutions and leaves less room for thermal protection against extreme hot and cold. There are several types of probes in which we can clearly point out that they have TCSs with miniature satellite characteristics, such as CubeSats, NanoSats and FemtoSats. [72]

	Mass Class Name	Kilograms (kg)
Smallsats	Femto	0.01 – 0.09
Smallsats	Pico	0.1 – 1
Smallsats	Nano	1.1 – 10
Smallsats	Micro	11 – 200
Smallsats	Mini	201 – 600
	Small	601 – 1,200
	Medium	1,201 – 2,500
	Intermediate	2,501 – 4,200
	Large	4,201 – 5,400
	Heavy	5,401 – 7,000
	Extra Heavy	> 7,001

Figure 7-1 Categorization of satellites. [73]

Thermal control technology for miniature satellites is primarily based on passive methods, that is, surface finishes, radiators, and, sometimes, heat pipes. The active methods like liquid pumps are not typically used at this moment. There is a comparison between a GEO satellite and a 3U CubeSat in figure 7-1. As shown, not every aspect scales the same. While their volume is, for instance, 10^4 times smaller, their power consumption is just 10^2 times smaller. This leads to high power density and relatively low radiator surfaces. The simplest way to remove waste heat from components is to mount heat dissipating components on a radiator inner surface. As the components in a small sat are packed and often away from the radiators, it calls for advanced heat transfer and management. [72]

Thermal dissipation for high power small satellites is challenging using only a body-mount radiator panel, and this is best addressed by increasing the radiating area by means of deployable panels. If deployable radiator panels are used, it is very important to ensure that the thermal conductivity is higher for the hinges used. Flexible thermal straps along with proper mechanical hinges, which have variable bending angles, can be used to deploy radiators. Flexible thermal straps can be considered for thermal connection between the heat dissipating element and the deployable radiator. [74]

	GEO com	3U cube sat	Ratio GEO/cube
Satellite dimension [m]	$3.0 \times 3.0 \times 6.0$	$0.1 \times 0.1 \times 0.3$	
Satellite volume [m^3]	54.0	0.003	18,000
Radiator surface [m^2][a]	42.88	0.06	715
Power [W]	5000	50	100
Power density [W/m^3]	92.6	16,666.7	0.6%
Efficiency	60%	60%	
Waste heat	2000	20	100
Rejected heat flux [W/m^2]	46.6	333.3	0.14

[a] Only two largest surfaces are used as radiators.

Figure 7-2 Comparison of a GEO communication satellite and a 3U CubeSat. [72]

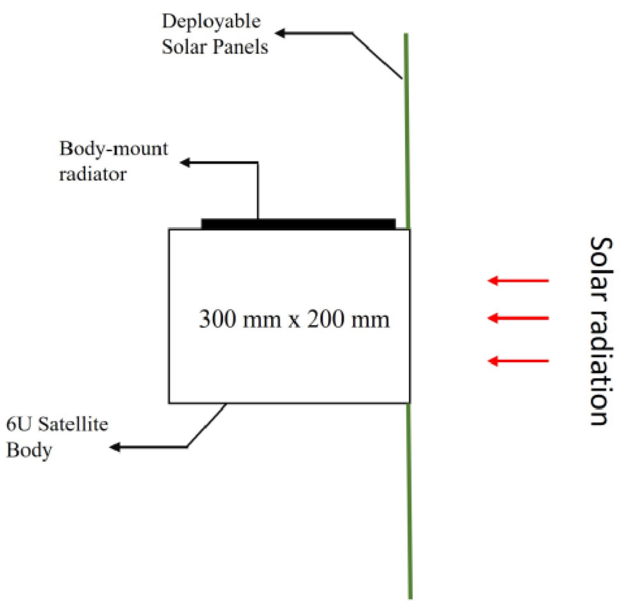


Figure 7-3 Placement of a body-mounted radiator on a 6U CubeSat. [74]

A miniature satellite can have thermal controls divided into several zones. Components from the same group can be connected to one radiator, which could have the highest temperature and consequently the smallest size. Batteries usually operate in smaller temperature range than other electronics and lower operating temperatures. They often have a separate thermal control system with dedicated radiator and heater. Optical elements do not tolerate large temperature changes; large temperature gradients across the optics must be avoided to prevent thermal distortion. [72]

The creation of several zones with different temperatures allows some radiators to operate at higher temperatures. Thanks to this, they can have a smaller surface area than if they used one common radiator. Additionally, a deployable radiator can be used to increase the cooling area, but it requires the use of heat pipes, so they are not used very often. [72]

Phase change material (PCM) absorbs energy during melting and releases the energy during solidification. A general application of PCM thermal control is for cyclically operating components that are operating on an On-Off cycle. The same is true for environment heating of the probe. A PCM accumulator can store solar energy when the radiator is facing the Sun and release energy during shadow periods. Storing energy at the peak load and releasing it later significantly reduces required radiator size. [72]

In flat satellites, usually FemtoSats, only one side is exposed to the sun at any time. The shaded side can easily radiate excessive heat into space. A thin silica gel layer is used to strengthen the heat conduction ability between the two sides. [75]

External **MLI** reduces thermal radiation to space and protects against excessive heat fluxes such as Sun and thruster plume heating. Internally, MLI blankets are used to create areas of separate thermal control, to reduce temperature gradients across the spacecraft, and to reduce heater power consumption. [72]

Heat is conducted from hot to cold regions. Typically, components inside a CubeSat are mounted on shelves that are attached to the space frame. Rough estimations indicate that the thermal conductivity of such a pass is about 0.5 W/K. For transfer of a couple of Watts, a temperature drop across the pass is not significant. However, transfer of 20 W would require a temperature difference of 40 °C. In order for heat to be conducted to the radiator, it would have to be very cold, which would lead to a significant increase in the radiator size. Creation of a thermal path with minimum resistance, using **highly conductive strips**, is thus very important for CubeSats with power above 20-30 W. [72]

A **heat pipe** is the best solution for transferring more than 10 W of heat across these types of junctures. Heat pipes are useful for several applications, such as conducting heat from a component mounted on a shelf to an external surface, transferring solar load from a sunny side to a shaded side, connecting an external panel with a deployable radiator, spreading heat over radiator to reduce temperature gradients along its surface. [72]

For external surfaces, OSRs, white paints and other treatments are used to minimize absorbed solar energy while maximizing heat rejection. Black paint is commonly used on internal surfaces of the spacecraft to enhance radiative heat transfer among internal components and radiative exchange with the internal surfaces of radiators. [72]

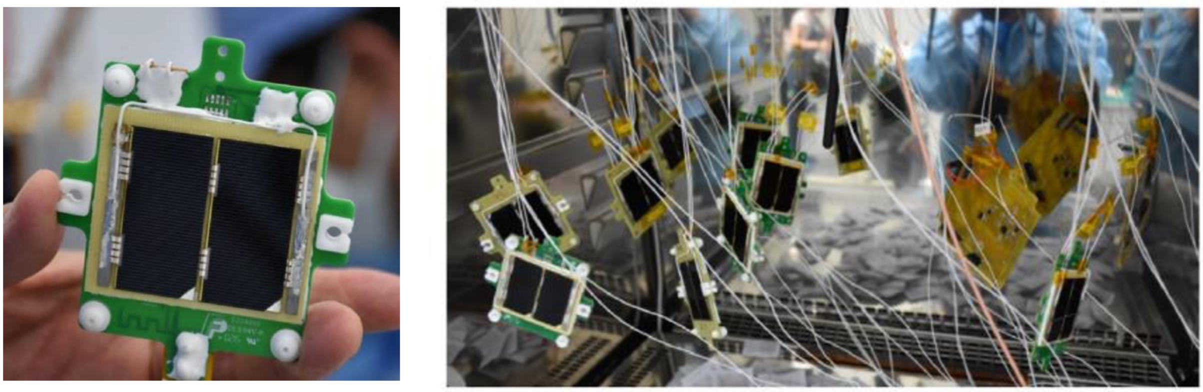


Figure 7-4 Femto-satellites in high-low temperature chamber. [75]

Some miniature satellites can carry IR imagers, which need to be cooled down to 100K. Active methods must be used to achieve such low temperatures, for instance, using a micro-cryocooler. For missions with such IR payloads and stringent thermal requirements, active thermal control systems are used. Satellite active thermal control systems rely on input power for operation and have been shown more effective in maintaining the required temperature within required limits, but this increases the total power budget of the satellite and eventually the cost. In general, small satellites cannot generate more power due to limited solar array size. [74]

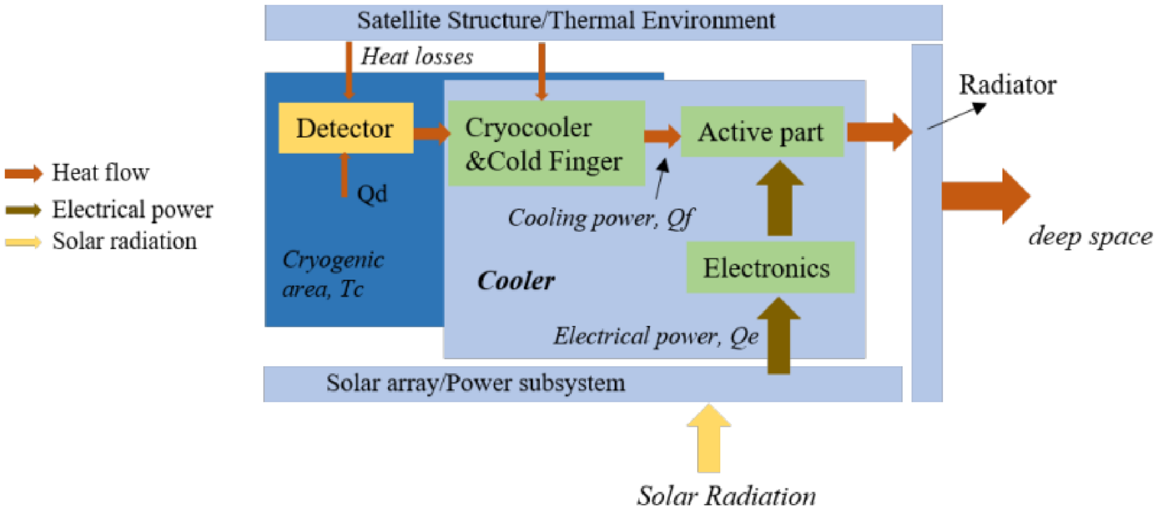


Figure 7-5 The heat balance in a microsatellite with a cryocooler. [74]

8 Space Observatories

Space observatories, i.e., space telescopes, are built around a set of scientific instruments which detect electromagnetic radiation, from radio to gamma rays. These instruments usually need cryogenic temperatures as low as 5.5 K for proper operation. In addition, the spacecraft must be kept at a uniform temperature to avoid thermal distortion, which would not allow precise scientific measurements. [76] [77]

From a thermal control perspective, the best orbits for the telescopes are in deep space, either an Earth trailing orbit or in a Lagrange point, usually L2. There they receive a constant solar flux which can be easily shielded. An alternative is a Sun-Synchronous Orbit (SSO), where they receive albedo and thermal IR radiation from Earth, in addition to the roughly constant solar flux. For the values, see chapter 3. Another alternative is a generic LEO. [78] [79] [80]

The spacecraft are divided into many thermal compartments. The solar array and sunshades present the first layer of protection. The bus is divided into several sections for operational and scientific equipment. The telescope section is thermally decoupled from the bus. It contains active cooling system and the science instrument assembly.[76] [81]

Figure 8-1 Cutaway view of the Spitzer cryogenic telescope assembly showing the details of its mounting to the spacecraft bus and internal details of the structure. Conductive isolation from the bus is accomplished using struts (yellow). [76]

In the **spacecraft's bus**, arious parts need to be heated in order to operate. The bus is divided into sections with different temperature ranges. Many sections are equipped with their own heaters, heat pipes and radiators. To avoid heating the telescope, the bus is thermally decoupled form it. The TCS is also required to maintain the spacecraft bus structure at a very uniform temperature. Changing temperatures would cause distortion and misalignment between the star trackers and the telescope. [81] [82]

On Spitzer, to maintain an isothermal structure, heat pipes were imbedded in the structural panels to distribute heat from warm areas to cold areas. Similarly, the upper deck contains a heat pipe that attaches to each of the scientific instrument assembly strut mounting pads in order to maintain a pad temperature difference of no greater than 1 K. [82]

The telescope section consists of several shells that protect the scientific instruments from outside thermal loads. Each shell is cooler than the one surrounding it. The shells often have their active cooling system. For instance, Kepler propane and ammonia flowing through pipes embedded in the spacecraft's exterior panels. Other telescopes use helium vapor cooled shells. [76] [78] [81]

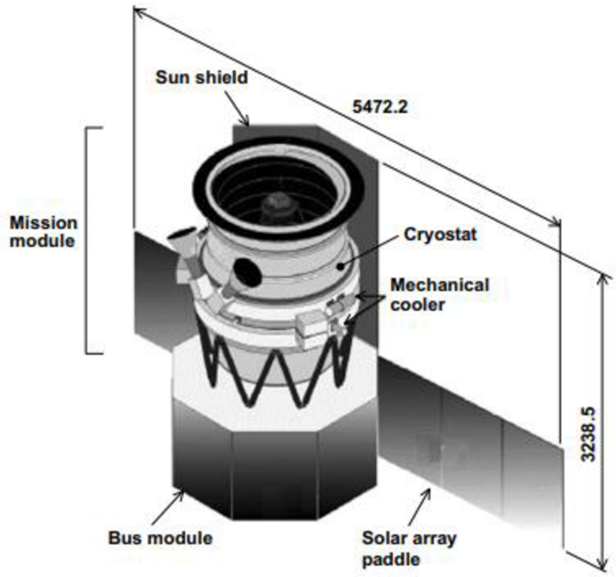


Figure 8-2 Akari satellite overview. [79]

The scientific instruments are insulated from the telescope using thermally non-conductive struts. The instruments are cooled by venting helium vapor. The helium tank operates at a temperature of 1.2 K. The parasitic heating to the helium tank is minimal, only in the order of 10^{-3} W. The heat produced by the detectors is in the order of 10^{1} W. It must be removed by the heat pipe and ultimately a radiator mounted on the cold side of the spacecraft, which is usually the anti-sun facing side of the outer shell. The size of the radiator is the factor that limits the operating temperature of the detectors. The radiator itself can cool down the instruments to 30-40 K. [76] [83]

There are heaters which warm up the helium when colder operating temperatures are needed. The increased boil-off rate leads to more helium vapor which cools down the instruments. The observatory's cold lifetime is determined by the initial mass of the helium bath on. [76] [79]

In addition to the helium system, the Akari space observatory uses mechanical coolers to cool a thermal radiation shield. For redundancy, two mechanical coolers are mounted on the outer shell to reduce the parasitic heat load to the helium tank and the scientific instrument assembly. Using mechanical coolers, observations are possible even after the helium is depleted. This hybrid design prolonged cryogen lifetime from two years to three years. [79]

The **solar panel** is radiatively decoupled from the spacecraft by MLI and a **sunshade**. All of this contributes to a low effective emissivity and minimal radiation heat leak into the. A similar shielding technique is used for the top deck of the bus to shield it from the bottom of the telescope. Low thermal conductivity struts are used to mount the telescope to the spacecraft bus and minimize conduction heat leak. [82]

The design of the sunshade assembly is critical because kilowatts of power are incident on the Sunward side of the shield. Through careful design and construction, the incident solar power impinging on the observatory is either reflected or reradiated so that the inward heat flow is reduced by a factor of 10^4 before it reaches the outer shell. [76]

The deployed sunshield is the key to James Webb Space Telescope's thermal design and consists of five aluminized Kapton layers separated in a V-groove configuration. The sunshield completely shades the cryogenic areas from the Sun over a wide range of viewing angles. This new design reduces the solar heating by a factor of 10^5. The 200 W of internally generated heat are dissipated into deep space to prevent it from hitting the cold side of the sunshade. Otherwise, it could reflect back onto the cryogenic radiators. [83]

Space observatories in LEO, such as the Hubble Space Telescope, face temperature swings of the exterior of ~250 °C. They do not use sunshades. Instead, extensive use of a reflective, low-emissivity MLI to wrap the majority of spacecraft's exterior is vital to their passive thermal control. The the Hubble Space Telescope's coating reflects several kilowatts of Solar and Earth radiation, and total internal heat load is 450 W from a 3.1 m^2 radiator. [84][83]

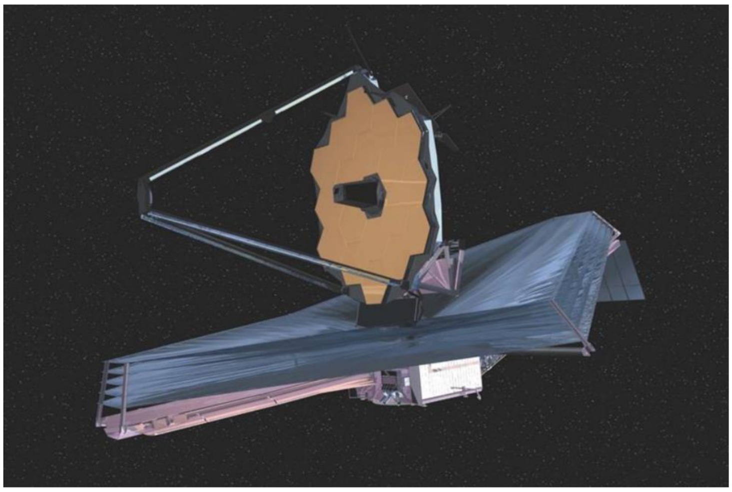


Figure 8-3 James Webb Space Telescope deployed. [83]

9 Comparison of Thermal Control Systems

Environment	Solar distance	type	characteristics	heat balance*
Low Earth Orbit	1.0 au	Flat comsat	Body-mounted radiators Dense thermal coupling	mostly rejection, 10^2 to 10^3 W
Low Earth Orbit	1.0 au	Box-shaped satellite	Body-mounted radiators External thermal compartments for specific instruments	mostly rejection, 10^2 to 10^3 W
Sun-Synchronous Orbit	1.0 au	SSO satellite	Body-mounted radiators External thermal compartments for specific instruments	rejection, 10^2 to 10^3 W
Geostationary Earth Orbit	1.0 au	GEO comsat	Body-mounted radiators with heat pipes	rejection, up to $1.1\text{x}10^4$ W
Geostationary Earth Orbit	1.0 au	GEO comsat	MPFL Body-mounted and deployable radiators	rejection, up to $3\text{x}10^4$ W
Lunar orbit	1.0 au	Single-module spacecraft	Preheating capability or louver	alternating, 10^2 W
Lunar orbit	1.0 au	Modular spacecraft	Separated thermal modules Preheating capability	alternating, 10^2 W
Venus orbit	0.7 au	---	Highly reflective coating Solar panels with OSRs Louvers or cold side with radiator Large number of heaters	rejection, $10^{\check{s}}$ W
Mars orbit	1.5 au	---	Significant heating capability Heat-preserving insulation Heat-rejection hardware	heating, 10^2 W
Asteroid and comet orbits	2-5 au	---	Louvers Heat-preserving insulation	heating, 10^2 W
Orbits below 0.5 au	perigee < 0.5 au	Spacecraft with sunshade	Sunshade Solar panels with OSRs Heat-resistant appendages	rejection, up to 10^6 W

Orbits below 0.5 au	perigee < 0.5 au	MLI wrapped	Several MLI wrappings Louvered radiator with 10^2 heat pipes Heat-resistant appendages	rejection, up to 10^4 W
Orbits below 0.5 au	perigee < 0.5 au	Spin-stabilized	Spin 15-60 rpm Covered with OSRs Heat-resistant appendages	rejection, up to 10^4 W
Outer Solar System	operation phase > 5 au	Spacecraft with RTG	RTG (source of heat), protection Radioisotope heater units Heat-preserving insulation Louvers HGA as sunshield	heating, 10^2 W (20 % is waste from RTG)
Outer Solar System	operation phase > 5 au	Spacecraft with solar arrays	Louvers HGA as sunshield Heat-preserving insulation	heating, 10^2 W
Microsatellites	1.0 au	---	Body mounted or deployable radiators PCM Careful heat conduction	mostly rejection, 10^1 to 10^2 W
Space Observatories	1.0 au	LEO spacecraft	MLI wrapping Telescope protected by several shells Active cryogenic cooling system	rejection, 10^3 W
Space Observatories	About 1 au	SSO and deep space spacecraft	Sunshade Telescope protected by several shells Active cryogenic cooling system	rejection, 10^3 W

*Heat balance during the spacecraft's operation: *Rejection* means that the spacecraft primarily rejects heat, usually by dissipating internal, reflecting external or both, although it may perform heating in other parts simultaneously. *Mostly rejection* means that the spacecraft sometimes changes mode, and it primarily preserves and adds heat. *Alternating* means that the spacecraft alternates between primarily rejecting heat and primarily heating. *Heating* means that during the operational phase, heat is mostly generated and radiating it away is less desirable.

Earth Orbit

Satellites in Low Earth Orbit (LEO) must tackle radiation coming from the Sun and Earth, a total of 1-2 kW from different directions. They are also subjected to frequent day/night cycles. Since they use 10^2 to 10^3 W to power the electronics, much of the TCS is oriented towards heat dissipation. They use thermal couplings, heat pipes and usually body mounted radiators. The periods in Earth's shadow allows them to cool down the radiators and get rid of the excess heat. They survive these periods thanks to their thermal inertia. They also make use of additional heaters, especially when some of the electronics are not in operation, and thus cannot utilize excess heat.

In the last decade, flat satellite buses have become increasingly popular, since they can be easily stacked and launched in large constellations. One side is usually sun-exposed, while the Earth-pointing side serves as a radiator. The flat layout maximizes the surface area.

TCSs for spacecraft in Sun-Synchronous Orbits are similar to the ones in LEO, but with larger radiators, since they cannot cool down in Earth's shadow. They still accommodate heaters to compensate for inactive electronics, and for emergency cases.

Spacecraft in a Geostationary Earth Orbit (GEO) are mostly for communication, and their equipment uses 10^3 to lower 10^4 W, 60 % of which is dissipated as heat. That is the highest amount among all unmanned spacecraft. The solar flux is the same as in LEO, but radiation from Earth is of much less importance. However, they sometimes experience eclipses, which last up to 1.2 hours. They have large, rectangular bodies with radiators mounted on two sides, connected to the other instruments via heat pipes. Satellites with over 11 kW of power use deployable radiators. Mechanically pumped fluid loops are used for even high-power heat management.

In general, spacecraft in Earth orbits do not experience extreme environmental variations throughout the year. Their TCSs are optimized for dissipating the high internal heat.

Inner Solar System

Probes in the Inner Solar System, more specifically in orbits from Venus to the outer parts of the Asteroid Belt, are scientific missions with power ranges of 100 to 2000 W. They experience a wide range of solar radiation along their paths, from about 2.6×10^3 W at Venus to about 1×10^2 W at 4 au. They often make use of louvered radiators to cover this need for flexibility in thermal design. Recent spacecraft in hotter environments, like the Moon and Venus, manage to combat temperature changes without louvers.

Spacecraft in lunar orbit are not subjected to the changing solar irradiation that other Solar System probes experience along the way. However, they must tackle the day/night cycles and the uneven thermal IR radiation from the Moon's surface. They usually point towards the nadir, and therefore do not place radiators on that side. Both the hot case (heat rejection) and cold case (heat preservation) are of high importance on these missions. During lunar eclipses they experience several hour periods of shadow. The spacecraft are usually preheated several orbits before, in order to remove strain on the TCS which would call for additional thermal protection. An alternative is using louvers.

Venus probes are mostly built for heat rejection once they reach their destination. They receive strong solar radiation as well as albedo radiation (Venus' albedo is 0.8). The IR thermal radiation is less significant. The probes are covered with highly reflective coatings, with precautions taken to avoid reflections on other parts of the spacecraft. Solar panels have stripes of Optical Solar Reflectors (OSRs) that cover around 30 %. There are radiator placement methods – either by using louvers to protect them from the Sun, or by keeping the probe oriented in such a way that allows a permanently shadowed side with radiators, and even cryogenic radiators. The satellites may be placed on orbits that take them on the night side or far from the planet for cooling. The spacecraft are cold-biased, meaning that their TCS should limit the maximum temperature around Venus. The spacecraft might therefore get too cold during some phases and the TCS must tackle it by using heaters. Examples are the initial flight, or situations when less instruments are operational. The internal components of the TCS are similar to other Inner Solar System probes.

Mars' thin, relatively cloudless atmosphere is highly transmissive to IR. It causes nightside planetary IR to be much lower than that on the dayside. However, in typical Mars' orbits, it only presents a difference in the order of 10^1 W/m^2. Mars presents a relatively stable environment without extreme temperature differences or spikes. Mars is the last planet in our Solar System near which a spacecraft will experience significant environmental heating. Nevertheless, the TCS are mostly concerned with preserving heat. Thermal control consists mostly of heaters, Multi-Layer Insulation (MLI) and more absorptive coatings. Around 50 % of the power is used for heating. During the cruise phase, a similar amount of heat is *rejected* in the beginning, since the probes receive significantly more heat from the Sun (orders of 10^2 W).

Probes heading towards asteroids and comets receive much less Solar radiation, so most of their TCS is for keeping the spacecraft from freezing. Only 15-40 % of the power is used for heating, because the TCS is so focused on preserving heat. Note that it is around 50 % for the studied Mars probes. Their orbits, however, get them Closer to the Sun, often multiple times. Large louvers are essential for the operation and thermal management in these widely varying conditions.

Orbits below 0.5 au

For orbits below 0.5 au, spacecraft must implement significant measures in odrer to face the extreme radiation. Spacecraft with sunshades are capable of handling the highest heat loads, rejecting 10^4 to 10^6 W of radiation. Only a tiny fraction, around 0.01 % of the sunlight reaches the spacecraft, not accounting for appendages and instrument openings. They must be three-axis stabilized, so that they can keep the sunshade pointing in towards the Sun. Behind the sunshade, the TCS of the buses are similar to other probes in the Inner Solar System.

Spacecraft wrapped in MLI, such as BepiColombo's MPO, lack a sunshade and instead are wrapped in thick MLI to combat radiation from multiple directions, which reaches the order of 10^4 W. Of those, around 1 % leaks through the MLI and must be rejected through a large, shielded radiator. There are almost 100 heat pipes connected to it, a significant increase compared to the previous category. The spacecraft is three-axis stabilized.

In contrast to that, spin-stabilized spacecraft combat heat by spinning fast, at 15-60 rpm, compared to the 2-3 rpm of Outer Solar System probes. They are mostly covered in OSRs. They can reject heat in the order of 10^4 W. Around 90 % is reflected, the rest needs to be radiated away.

All spacecraft flying below 0.5 au take more measures to combat the high external heat. Solar arrays are covered with OSRs and tilt to reduce the exposed area. Appendages are covered in MLI and made of heat resistant materials. The Parker Solar Probes even uses coolant pumps to cool the solar panels. The internal power balance is in the order of 10^2 to lower 10^3 W, except for the mentioned Parker Solar Probe, which pumps $6x10^3$ W just from the solar panels during the closest Solar approach. Because they reject most of the solar radiation, during the outer phases, they need additional heating in the orders of 10^2 W.

Outer Solar System

In the Outer Solar System, spacecraft with Radioisotope Thermoelectric Generators (RTGs) can utilize waste heat to help warm the spacecraft, usually the fuel tanks. However, due to the challenges of efficiently spreading the heat, only up to about 1 % (order of 10^1 W) of the waste heat is used for that. About 200 W of approximately 4000 W that a single RTG produces will be utilized to power the electronics, while the rest is radiated away. The spacecraft must be shielded from this IR radiation. Most of the effective power is used by heaters when the electronics are not in operation. Spacecraft are equipped with up to three RTGs. In addition, they might be equipped with radioisotope heaters which do not need an electrical power source. These are mounted mostly near instruments outside the bus. In the Galileo spacecraft, they amount to 20 % of the 500 W that the spacecraft utilizes.

In recent years, solar panels are often used instead of RTGs for reasons unrelated to thermal control. However, they do not provide the additional benefit of waste heat to warm the fuel tanks.

In general, around half of the useful power (order of 10^2 W) is utilized to heat the spacecraft. When the spacecraft is flying in the Inner Solar System, much of the solar radiation is rejected by the High Gain Antenna, which serves as a sunshade. Louvers are open in this phase, and they fully close in the Outer Solar System to conserve heat. The bus can be a single, unified, thermally coupled compartment, which simplifies design and helps conserve heat. The bus can also comprise of several smaller compartments with their respective Thermal Control Subsystems, which allows for multiple instruments with different temperature requirements.

Miniature satellites

The compact size of miniature satellites (CubeSats, Microsats, etc.) is the primary driver behind their TCS design. Power and surface area scale differently, and as a result, microsatellites have much higher power-to-surface ratio than big satellites. The line between “regular” probes and miniature ones, from a thermal control perspective, is not completely clear. Especially in LEO, larger satellites have TCSs that share many similarities with miniature satellites. The line starts to blur around the MicroSat range, 11-200 kg, 10^2 W. Generally, miniature satellites have the equipment and instruments more densely packed, which requires more focus on heat transfer solutions and leaves less room for thermal protection against extreme hot and cold. Thermal control technology for miniature satellites is primarily based on passive methods, that is, surface finishes, radiators, and, sometimes, heat pipes. A lot of focus is placed on the thermal transfer between parts, or its prevention. In such compact sizes, even single bolts matter.

As mentioned above, the high power-to-surface ratio makes the TCS more focused on heat dissipation. When body mounted radiators are not enough, they increase the surface are with deployable radiators. Nevertheless, microsatellites carry heaters to combat cold temperatures in shadows or when instruments are off. Phase changing materials are also often used to store heat during peaks and release it during lows. Depending on the spacecraft, it may reduce radiator area (hot case) or heater power (cold case).

A downside of microsatellites is that they work best only around Earth and cis-lunar space, where the solar distance illumination does not change much over the year. Probes operating closer to the Sun lack the necessary surface area and shielding. Microsatellites flying outwards require more insulation, and especially large antennas. Both shielding and antennas favor larger spacecraft. At the time of writing this thesis, only a handful of microsatellites have flown past Earth's vicinity. Most use their parent spacecraft's TCS and operate independently only for a few hours or days in deep space. Several others operate in heliocentric orbits at the distance of around 1 au.

The TCS on FemtoSats (0.01-0.09 kg) is purely passive, as their power is only 10^{-1} to 10^{0} W. They are flat and very small. One side is illuminated by the Sun while the other radiates heat away. These two sides are thermally coupled with conductive paste/tape.

Observatories

Space observatories, i.e., space telescopes, are built around a set of scientific instruments which detect electromagnetic radiation. These instruments usually need cryogenic temperatures as low as 5.5 K for proper operation. In addition, the spacecraft must be kept at a uniform temperature to avoid thermal distortion, which would not allow precise scientific measurements.

Observatories in Lagrange points, Earth-trailing orbits and Sun-synchronous orbits have a solar array coupled with a sunshade, which protects the telescope from the solar heating, reducing it by a factor of up to 10^{5}. Telescopes in LEO are completely wrapped in a reflective insulating coating. For better heat management, the bus is divided into thermal compartments. The bus and the shields are thermally decoupled from the telescope assembly. Heat dissipation from internal instruments is in the order of 10^{1} to 10^{2} W.

The telescope is made of several layers, each being colder than the one surrounding it. The detectors are usually thermally decoupled too. They are connected to a cryogenic radiator that faces deep space and can cool down the instrument assembly to 30-40 K. The instruments are also connected to the cryogenic cooling system, which is typically helium, and it can further cool down the detectors to 5.5K. The operational longevity of the spacecraft depends mostly on the rate of depletion of the coolant. Some observatories use mechanical in conjunction to extend the longevity after helium is depleted.

Conclusion

This thesis has provided an overview of thermal control system trends for probes and microsatellites based on flight-proven designs. Some of these designs are given a closer look. Commercial spacecraft tend to follow trends in thermal control system design quite closely. Scientific spacecraft try to use flight-proven heritage thermal control too whenever possible. This is especially apparent in microsatellites, which are often based on the CubeSat architecture. However, the unique requirements of scientific missions sometimes require pushing the boundaries of thermal control design, leading to the incorporation of specialized thermal control features and subsystems.

Orientation of the spacecraft and its flight and orbital path present a challenge for thermal control, but it can be used to its advantage, to control the environmental exposure. Most spacecraft have a Sun-exposed face and a cold face with radiators. The distance from the Sun that the spacecraft reaches is one of the main drivers behind the thermal control system, and thus it was the primary of the criteria for classification throughout this thesis. In orbits with a perihelion below 0.5 au, probes must be protected from the intense radiation. On the other hand, probes in the Outer Solar System receive very little Solar radiation and are concerned with preserving heat. Spacecraft in the rest of the Inner Solar System alternate between heating and rejecting heat. Earth orbits have been given a special category, since they accommodate mostly communication satellites, which dissipate large amounts of internal heat. The last two categories have been grouped based on their configuration: microsatellites cover thermal control systems for spacecraft with small dimensions and high power density, and space observatories cover telescopes. An outcome of this thesis is a table listing and comparing the characteristics and power balance, supported by summaries of main points of each category.

Thermal control systems generally group instruments that operate in similar temperature ranges into common compartments. When it is possible, even the whole spacecraft is made into a unified thermal compartment. This simplifies the design and testing process, saves insulation mass, and adds redundancy. Nonetheless, some specialized instruments need dedicated vaults or subsystems. These can be cryogenic equipment, appendages and gimballing equipment, or equipment with strict thermal control, such as fuel tanks or batteries.

In general, spacecraft strive to have mostly passive thermal control systems, because those need no power supply to operate. Most have at least Multi-Layer Insulation to protect them from the large temperature swings in space. However, all studied spacecraft are equipped at least with electric heaters. These are usually in two classes – software operated for fine-tuning thermal control, and thermostat operated for emergency cases and/or simplicity. Rovers and spacecraft with high internal dissipation in the order of kW, generally communication satellites, use active cooling loops. External units, such as antennas and mounts, are insulated from the spacecraft, because they must withstand greater temperature ranges of a couple hundred K.

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List of Abbreviations

ESA	European Space Agency
GEO	Geostationary Orbit
GSO	Geosynchronous Orbit
HGA	High-Gain Antenna
IR	Infrared
LEO	Low Earth Orbit
LGA	Low-Gain Antenna
LRO	Lunar Reconnaissance Orbiter
MGA	Medium-Gain Antenna
MLI	Multi-Layer Insulation
MMO	Mercury Magnetospheric Orbiter
MOSIF	Mercury Magnetospheric Orbiter's Sunshield and Interface Structure
MPFL	Mechanically Pumped Fluid Loop
MPO	Mercury Planetary Orbiter
OSR	Optical Solar Reflector
PCM	Phase-Changing Material
RTG	Radioisotope Thermoelectric Generator
S/C	Spacecraft
SSM	Second Surface Mirror
SSO	Sun-Synchronous orbit
SSL	Space Systems/Loral
TCS	Thermal Control System
UV	Ultraviolet