Spacecraft thermal control

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Prof. P.Rochus, prochus@ulg.ac.be

Ass. L.Salvador, lsalvador@ulg.ac.be
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Introduction
“Maintain all of the spacecraft’s components within the allowable temperature ranges for all operating modes of the satellite, in all the thermal environments it may be exposed to” (D.G. Gilmore, 2002)

Two facts:

- electronic and mechanical equipment usually operate efficiently and reliably only within relatively narrow temperature ranges. Temperature stability (temporal)
- stringent structural requirements for alignment, pointing,…Limit temperature gradients (spatial)

BUT

- hostile space environments: cold deep space @ 3K, no convection, rapidly changing illumination conditions, harsh heating in space reentry phases…
- electronic dissipative: without convection, more difficult to reach allowable temperatures
- low conductance for interfaces
- most materials have non-zero CTEs and hence temperature changes imply thermal distortion…
Introduction

- In space, only the radiation and conduction modes for heat transfer are predominant (convection in the environment negligible because free mean path of the particles is large)

- In practice
  - We have to adjust external exchanges with the environment (external surfaces, radiators,…)
  - We have to adjust internal exchanges to ease the evacuation of dissipated power and absorbed external power (Sun, Earth albedo, Earth IR,…)
  - Internal surfaces, conductive links,…)

- For some applications (space reentry, planetary exploration, cryogenic instruments) the use of latent heat is mandatory (sublimation by ablation, fusion of a phase-change material, sublimation of He or H₂,…)

- Heat fluxes are the subject of thermal control (not temperatures). These adjustments allow to reach desired temperatures of the SC and subsystems.

- Thermal control is achieved by analysis (predict temperatures) and testing (confirm an validate analysis)
Typical requirements

- Typically electronic cards and mechanical equipment around room temperature (originally designed for terrestrial use, ease of testing,…), detectors usually colder (limit noise). Critical equipments are usually optics, detectors and batteries.

<table>
<thead>
<tr>
<th>Component</th>
<th>$T_{\text{min}}$ (°C)</th>
<th>$T_{\text{max}}$ (°C)</th>
</tr>
</thead>
<tbody>
<tr>
<td>Electronics</td>
<td>-20</td>
<td>+50</td>
</tr>
<tr>
<td>Batteries</td>
<td>-5</td>
<td>+25</td>
</tr>
<tr>
<td>Mechanism</td>
<td>-100</td>
<td>+80</td>
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<tr>
<td>Solar cells</td>
<td>-150</td>
<td>+70</td>
</tr>
<tr>
<td>Infrared detectors</td>
<td>-250</td>
<td>-80</td>
</tr>
<tr>
<td>Fuel</td>
<td>+10</td>
<td>+40</td>
</tr>
</tbody>
</table>

**Component Stability : $\Delta T/\Delta t$**

- Electronic unit : $< 5$ K/h
- CCD camera : $< 0.1$ K/mn
- Cryo telescope : $< 100$ µK/mn

**Component Gradient : $\Delta T$**

- Across optical instrument (1.5m) : $< 5$°C
- Between MMH/NTO tanks : $< 5$°C

- **NB**: a mean temperature 10°C greater can reduce by a factor 2 the life duration of electronic components!
In space, only the radiation and conduction modes for heat transfer are predominant.

Unfortunately the convection is rightly more efficient!

EXAMPLE 1 (DIFFICULTY OF THERMAL CONTROL UNDER VACUUM CONDITIONS)

Let’s consider an electronic box inside an environment at T=25°C, considered as a black body (absorbs all the radiation emitted by the box). Dimensions are h=12cm, L=15cm. The electronic can dissipate 0 to 10W. Three emissivities are chosen (ε=0.5, ε=0.7, ε=0.9). The box is fixed with 4 bolts, the conductance of which is supposed to be 4*0.2 W/K.

We can compare the cooling efficiency in ambient medium such as on Earth versus vacuum in a spacecraft.
- Illustrate the basic problem to face in spacecraft thermal control. In vacuum, we could not dissipate much more power without dramatically increasing the box temperature.

- Thus a solution would be to increase the external emissivity of the box, the bolt conductances, and dissipate the power as much as possible through a spacecraft radiator.
EXAMPLE 2 (CHANGING ENVIRONMENT FOR A CIRCULAR QUASI-POLAR ORBIT)

In near-Earth orbit, the main external heating source is the direct solar flux. We can see it varies from ~1400 W/m² in solar illumination to 0 W/m² when in eclipse!
Thermal Control as mission driver

ISO
SOLAR ORBITER
HST
‘TPS’

JWST
HERSCHEL

PHOENIX MARS LANDER

[1] Introduction
Thermal Control Overview

**Mission parameters**
- Launch/Space Environment
- Orbit, Attitude

**Requirements**
- Powers, temperatures, T gradients, stability dT/dt, …

**Mathematical Models**
- material properties, boundary conditions, stationary, transient, BOL, EOL, Cold cases, Hot Cases, reduced model ...

**Experimental Qualification**
- Thermal vacuum
- Thermal cycling
- Thermal balance

**Thermal control Methods**
- passive
- active

[1] Introduction
[2]

A bit of theory

[2.1]

Radiative heat transfer
Radiative heat transfer important in space applications

Spectral and directional properties

Properties can be integrated over whole range of a variable (or by part over certain spectral regions for spectral properties).
Black body radiation

- Useful concept widely used as a
  - perfect absorber and emitter (independent of direction)
  - standard against which actual surfaces may be compared

- Spectral emissive power given by *Planck distribution*; dependence on \( \lambda \) and T and independent of direction (\( \theta, \Phi \))

Integrating over the wavelength range, one finds the *Stefan-Boltzmann law*

\[
E_b = \sigma T^4 \quad (W)
\]

\[
\sigma = 5.67 \cdot 10^{-8} \quad (W/m^2K^4)
\]

Wien’s law links the temperature of a black body with the wavelength of its maximum emitted power

\[
\lambda_{max} T = 2898 \mu m.K
\]
Kirchhoff’s law

- At the equilibrium (given T)
  \[ \varepsilon_{\lambda, \theta, \phi}(\lambda, \theta, \phi, T) = \alpha_{\lambda, \theta, \phi}(\lambda, \theta, \phi) \]

- How can we find an integrated law?
  \[ \varepsilon_{\lambda} = \int_0^{2\pi} \int_0^{\pi/2} \varepsilon_{\lambda, \theta, \phi} \cos \theta \sin \theta \, d\theta \, d\phi \]
  \[ = \int_0^{2\pi} \int_0^{\pi/2} \alpha_{\lambda, \theta, \phi} I_{\lambda, i} \cos \theta \sin \theta \, d\theta \, d\phi \]

The equation stands if either,

1) irradiation diffuse \( (I_{\lambda, i} \text{ is independent of } \theta \text{ and } \phi) \)
2) surface diffuse \( (\varepsilon_{\lambda, \theta, \phi} \text{ and } \alpha_{\lambda, \theta, \phi} \text{ independent of } \theta \text{ and } \phi) \)

Now integrating over the wavelength range,

\[ \varepsilon = \int_{\lambda = 0}^{\infty} \varepsilon_{\lambda} E_{\lambda, b}(\lambda, T) \, d\lambda = \int_{\lambda = 0}^{\infty} \alpha_{\lambda} G_{\lambda}(\lambda) \, d\lambda = \alpha \]

The equation stands if either,

3) irradiation of a blackbody at T \( (G_{\lambda}(\lambda) = E_{\lambda, b}(\lambda, T) \text{ and } G = E_b(T)) \)
4) gray surface \( (\alpha_{\lambda} \text{ and } \varepsilon_{\lambda} \text{ independent of } \lambda) \)
The gray surface

- $\varepsilon$ is only a property of the surface while $\alpha$ depends on the surface AND the spectrum content of the irradiation $G$!

- The integrated Kirchhoff’s law $\alpha = \varepsilon$ is valid for a diffuse (lambertian) gray surface.

- A gray surface may be defined as one for which $\alpha_\lambda$ and $\varepsilon_\lambda$ are independent of $\lambda$ (approximately) only over certain spectral regions of the irradiation and surface emission and not the overall region.

It can be referred to semi-gray approximation

The gray surface concept underlies the main assumption in spacecraft thermal control.

Indeed the solar spectrum absorbed and reflected by the spacecraft’s surfaces \( \lambda_{\text{max}} (T=5800\,\text{K}) \approx 0.4\,\mu\text{m} \) is very different of the radiation spectrum emitted and exchanged between these surfaces \( \lambda_{\text{max}} (T=50\,\degree\text{C}) \approx 11\,\mu\text{m} \).

The surface can then be supposed to be a gray surface over two `thermal´ domains.

- Visible [0.2\,\mu\text{m} – 2.8\,\mu\text{m}] [800\,\degree\text{C}_1\text{4000}\,\degree\text{C}] (= 95% of Sun energy, <0.5% of IR energy)
- Infrared [5\,\mu\text{m} – 50\,\mu\text{m}] [-200\,\degree\text{C}_3\text{00}\,\degree\text{C}] (=92% of IR energy, < 1% of the Sun energy)

If the surface is gray over these two domains and diffuse, Kirchhoff’s law yields to

\[
\begin{align*}
\alpha_{\text{vis}} &= \varepsilon_{\text{vis}} = '\alpha' \\
\alpha_{\text{IR}} &= \varepsilon_{\text{IR}} = '\varepsilon'
\end{align*}
\]

NB : ‘vis’ also noted ‘sol’
This approach is more or less OK for the majority of space systems in which most areas of the spacecraft or instrument have temperatures in the range roughly between -100°C and +100°C.

Where there are regions with vastly different temperatures and radiative heat transfer is dominant (or very important) then the semi-gray assumption no longer holds.

Examples: cryogenic detector systems, very hot instruments or re-entry surfaces, while the rest of the spacecraft is around room temperature.

Must perform thermal radiative analysis in multiple spectral bands!

Problem: get or determine thermo-optical properties for each of the bands!!
Using the main assumption of spacecraft thermal control, a surface can then be defined by 8 thermo-optical properties:

\[ \alpha_{sol}, \rho_{sol,d}, \rho_{sol,s}, \tau_{sol}, \varepsilon_{IR}, \rho_{IR,d}, \rho_{IR,s}, \tau_{IR} \]

linked by the energy balance relations:

\[ \alpha_{sol} + \rho_{sol,d} + \rho_{sol,s} + \tau_{sol} = 1 \]

\[ \varepsilon_{IR} + \rho_{IR,d} + \rho_{IR,s} + \tau_{IR} = 1 \]

In practice, the solar flux is absorbed (\( \alpha_{sol} \)) and reflected (\( \rho_{sol,tot} = 1 - \alpha_{sol} \)) by the satellite’s surfaces (considered opaques) hit by incident Sun rays. The solar flux then heats these surfaces at a certain temperature (usually around -100°C +100°C) and the surfaces emit an infrared flux (\( \varepsilon_{IR} \)) towards the cold space and other surfaces, a part of which is absorbed (\( \varepsilon_{IR} \)) and another is reflected between the surfaces (\( \rho_{IR,tot} = 1 - \varepsilon_{IR} \)).
Exchanges between surfaces

- **VIEW FACTORS:**

  Fraction of uniform diffuse radiation emitted by a isothermal surface $A_i$ that directly reaches another surface $A_j$

  \[ F_{ij} = \frac{1}{A_i} \int_{A_i} \int_{A_j} \frac{\cos \theta_i \cos \theta_j}{\pi R^2} dA_i dA_j \]

  In practice, view factors are mainly used for preliminary analytical calculus or calculus of more general factors (Gebhardt’s factors)

- **GEBHART FACTORS:**

  Fraction of radiation emitted by an isothermal surface $A_i$ that reaches another surface $A_j$ directly and after multiple reflections with other surfaces $A_k$

  \[ B_{ij} = F_{ij} \varepsilon_j + \sum_k F_{ik} (1 - \varepsilon_k) B_{kj} \]

  Gebhardt factors allow to take into account the direct radiative coupling between surfaces

Exchanges between surfaces

EX: a S/C solar panel (node 2) that sees S/C (node 3) and Deep Space (node 1)

A main drawback of using view factors or gebhart factors is that the irradiation is assumed uniform (can be easily unvalidated). Hence the radiative exchange factors $GR_{ij}$ can be directly computed by a Monte-Carlo Ray Tracing method (MCRT) which is a widely used method.

$$q_{i,j} = GR_{ij} \sigma (T^A_i - T^A_j) \quad (W)$$

$$GR_{ij} = \epsilon_i A_i B_{ij}$$

---

A bit of theory

Conduction
Introduction

- Conduction is also very important in space applications since it allows to balance the internal powers between the components and permits to evacuate the dissipated power through external radiators. Inside a spacecraft it is the main mode of transfer for common temperature [-50°C - +100°C]

- Based on Fourier’s law \( q = -k \nabla T \) (W/m²)

- \( k \) (W/m.K) is the conductivity. Diagonal tensor for isotropic media. In space applications we also have CFRP (Carbon-Fiber-Reinforced Polymer) which is anisotropic!
k varies with temperature and it must be taken into account in cryogenic or space reentry applications for example.

We can neglect its variation when a typical range is considered, i.e. [273K-473K] [0°C-200°C]
Lumped parameter method

- Considering a rod of length L, sectional area A and isotropic conductivity k, the equation of conduction implies in steady state:

\[ Q = kA \frac{T_L - T_0}{L} = GL_{0,L}(T_L - T_0) \]

where \( GL_{0,L} \) \((W/K)\) is the conductance across the rod. Other conductances can be calculated and in the general case of complex geometries, shape factors may be used. Discretization techniques are also common: a temperature difference is imposed between two points and the resulting heat flux is calculated leading to the conductance value.

- Lumped parameter method: modelling a continuous medium as a discrete network of nodes. They represent the capacitance of the system linked by conductors that correspond to its conductances. It is linked with finite difference analysis and defines algebraic equations as opposed to analytical solutions.

- The conductive power exchanged between two nodes is:

\[ q_{i,j} = GL_{ij}(T_i - T_j) \quad \text{(W)} \]
[3]

Space Thermal Environment

[3.1]

Introduction
Spacecraft Thermal Environment

Introduction

- From transportation pre-launch to end of mission

- Transportation, pre-launch usually not a problem for the thermal control

- Space is a hostile environment:
  - Absence of convection (ultra-vacuum $10^{-14} \text{ bar} < p < 10^{-17}$) for satellites
  - Deep space @ 3K for satellites
  - Rapidly changing illumination conditions (Solar eclipse, planetary exploration, …)
  - Solar flux diminishes as $1/d^2$ (very high heat fluxes for Solar orbiters, very low for far missions, dramatic changing illumination conditions for space probes, …)
  - Atmospheres and conditions of other planets (planetary exploration, …)
[3]

Space Thermal Environment

[3.2]

Launch environment
Launch thermal environment

- SC thermal control systems usually designed to environment on orbit. Yet temperatures during initial phases must be predicted to ensure limits will not be exceeded.

- Thermal conditions before fairing jettisoning (< 150 km)

  ✓ For the first few minutes the environment surrounding the SC is driven by the payload-fairing temperature which rises rapidly to 90-200°C as a result of aerodynamic heating.

  ![Diagram of thermal conditions](image)

  Thermal flux density radiated by the fairing does not exceed 1000 W/m² at any point for Ariane 5 and Vega and 800 W/m² for Soyuz.

  Only cause temperature rise on relatively low-mass exposed components (solar arrays, insulation blankets, antennas, …)

  ✓ Within 2 to 5 min after liftoff aerodynamic heating drops low enough and the fairing may be jettisoned.
Launch thermal environment

- **Aerothermal flux and thermal conditions after fairing jettisoning (> 150km)**

  ✔ After fairing jettisoning, free molecular heating (FHM) occurs: heating is modeled as collisions of the body with individual molecules

  \[ Q_{\text{FHM}} = \alpha \left( \frac{1}{2} \right) \rho V^3 \]

  \( \alpha = \) accommodation coeff. \((0.6 \leq \alpha \leq 0.8\) but \(\alpha=1\) recommended for conservatism). The atmospheric density \(\rho\) highly variable and calculated by atmospheric models. Velocity during launch ascent calculated using launcher-trajectory simulations.

  ✔ Usually, these simulations are conducted by specialists who supply the thermal engineer with curves of worst-case heating versus time. With such a curve and a knowledge of the SC attitude relative to the velocity vector, one may calculate the heat load on the SC by simply multiplying the heating rate by the cross-sectional area of the surface in question and the cosine of the angle between the surface normal and velocity vector.

  \[ Q = Q_{\text{FHM}} A_c \cos (\mathbf{n}, \mathbf{v}) \]  

(W)
For European launchers, the nominal time for jettisoning the fairing is determined in order not to exceed the aerothermal flux of 1135 W/m². Typically the aerothermal flux varies from 1135 W/m² to less than 200 W/m² within 20 seconds after the fairing jettisoning.

Solar radiation, albedo, and terrestrial infrared radiation and conductive exchange with LV must be added to this aerothermal flux. While calculating the incident flux on spacecraft, account must be taken of the altitude of the launch vehicle, its orientation, the position of the sun with respect to the launch vehicle, and the orientation of the considered spacecraft surfaces. A specific attitude with respect to the sun may also be used to reduce the heating (e.g. ‘barbecue roll”).

ARIANE 5: Typical GTO mission
[3]

Space Thermal Environment

[3.3]

On orbit
On orbit

- On orbits, the principal forms of environmental heating are
  1) Direct solar flux
  2) Albedo (Sunlight reflected) [diffuse-specular]
  3) IR energy (emitted from planet and its atmosphere) [diffuse]

- Overall thermal control of a satellite on orbit is thus achieved by balancing the incoming direct sunlight, albedo and IR emission on the S/C plus the dissipated power within the S/C and the exchanges with Deep Space (IR radiation)

- For interplanetary missions, planet IR and albedo loads added to solar load for short periods of time during flybys. Thermal mass of the vehicle often damps out temperature rises of most components except maybe lightweight components.

- NB : an additional FHM can occur for low perigee orbits (<180km for Earth)
Direct sunlight is the main source of heating in space

Diminishes as $1/d^2$
Concept of reference sphere ($\alpha=1$ ; $\varepsilon=1$): rough indication of how ‘hot’ or ‘cold’ the local thermal environment is.
The Sun can be considered as

- A point source infinitely far away
- A point source at finite distance
- A finite-size sphere at finite distance
**Planet albedo**

- Fraction of incident sunlight on a planet reflected back to space (diffuse) which depends on
  - Planet’s surface reflectivity characteristics
  - View factor with planet (SC views a hemispherical cap)
  - Beta angle (minimum angle between orbit plane and solar vector)

- Some spectral absorptions within atmosphere (vapor of water, CO2, O3...)

- Highly variable (spatial and temporal)
  - Different surface and atmospheric properties. Especially true for Earth!
  - Albedo heat flux decreases as S/C moves along its orbit away from subsolar point. Local incident heat flux is reflected from surface inclined with respect to solar vector.

- SC views only a small portion of full planet. It is exposed to rapidly changing conditions that occur usually faster than the thermal inertia of most SC with respect to orbital period. It is thus common to take an orbital average value. Not a problem for massive well-insulated components but maybe for lightweight ones.
For practical purposes, we can assume that the planet reflects the solar rays uniformly with a mean albedo. Then the incoming albedo loads at orbit varies as $1/R_{\text{orbit}}^2$

$$q_{\text{albedo}} = q_{\text{sol}} a \cos(\theta) \cdot (R_p/R_{\text{orbit}})^2 \quad (\text{W/m}^2)$$

This method is only correct for preliminary calculations with a SC modelled by a point or a simple plate facing the planet. If we have a surface the albedo load can be non zero even if surface is perpendicular to planet because there exists a non zero view factor!

Or we explicitely use the view factors with the planet

$$q_{\text{albedo}} = q_{\text{sol}} a \cos(\theta) \cdot F_{\text{planet}} \quad (\text{W/m}^2)$$

✓ Dependence in $1/R_{\text{orbit}}^2$

✓ Dependence in surface’s orientation

(/!\ in reality even if the satellite has just come above the shadow part of the planet , i.e. $\theta \geq 90^\circ$, reflected rays may still hit the satellite. Particularly true for Earth polar orbits where the polar caps can cause forward scattering.)
Planet IR emission

- **IR energy emitted by a planet (diffuse) which depends on:**
  - Local surface temperatures
  - View factor with planet (SC views a hemispherical cap)

- **Some spectral absorptions within atmosphere (vapor of water, CO2, O₃...)**

- **Variable (spatial and temporal)**
  - Increasing cloud cover tends to lower emitted IR
  - Highest values of IR emission will occur in regions next to ecliptic plane (maximum solar heating) and will decrease with latitude

- Much less severe localized variations than for albedo. SC views only a small portion of full planet. It is exposed to changing conditions that occur usually faster than the thermal inertia of most SC with respect to orbital period. It is thus common to take an orbital average value. Not a problem for massive well-insulated components but maybe for lightweight ones. This is thus common to take a mean temperature value of the planet.

  // some planets don’t have a temperature value more or less constant over their entire surface as Earth
• A planet is in balance with the incoming direct solar flux and deep space

• Balance maintained fairly well on a global annual average basis, but IR emission at any given time from a particular point can vary considerably depending on factors such as the local temperature of surface and the amount of cloud cover

• For rapidly rotating planets (and preferably with atmosphere), the entire surface can be considered at approximately the same temperature.

\[
T_{\text{planet}} = \sqrt[4]{\frac{(1 - a) Q_{\text{sol}}}{4\sigma\epsilon}}
\]

• For slowly rotating planets, the back surface is considered too cold to emit a significant amount of energy

\[
T_{\text{planet}} = \sqrt[4]{\frac{(1 - a) Q_{\text{sol}}}{\sigma\epsilon}}
\]
Planet IR loads

- For most practical purposes, we can assume that the planet radiates uniformly from whole cross-sectional area. Then the incoming IR emission at orbit varies as \(1/R_{\text{orbit}}^2\)

\[
q_{IR} = \sigma \varepsilon T_{\text{planet}}^4 \cdot \left(\frac{R_p}{R_{\text{orbit}}}\right)^2 \quad \text{(W/m}^2\text{)}
\]

This method is only correct for preliminary calculations with a SC modelled by a point or surface facing planet. If we have a surface the IR load can be non zero even if surface is perpendicular to planet!

- Or we explicitly use the view factors with the planet (one usually assumes that the planet emits as a black body: \(\varepsilon=1\))

\[
q_{IR} = \sigma \varepsilon T_{\text{planet}}^4 \cdot F_{\text{planet}} \quad \text{(W/m}^2\text{)}
\]

- Dependence in \(1/R_{\text{orbit}}^2\)
- Dependence in surface’s orientation

- A temperature distribution can be implemented if the variations are large (e.g. slowly rotating planets…) or if analysis prove it’s necessary
Received flows depend on the attitude control => attitude control and S/C thermal control are closely linked

3-axis stabilized satellites

Spin-stabilized satellite
SC in low orbits routinely enter eclipse periods and this cause a great fluctuation in the direct solar flux received

Umbra is totally eclipsed by the planet and penumbra is only partially obscured. We assume spherical planet, no atmosphere influence on eclipse

A SC passes behind the terminator line of the planet if \[ \vec{r}_{\text{sun}} \cdot \vec{r}_{\text{sat}} < 0 \]

We can develop [see D.A. Vallado]

- A simple geometrical shadow analysis
- A traditional shadow analysis (using classical orbital elements and a shadow function)
- A simple line of sight geometrical method
- Useful parameter for visualizing the orbital thermal environment (eclipses and albedo loads) particularly for low orbits

- The orbit beta angle is the minimum angle between the orbit plane and the solar vector:

\[ \beta = \sin^{-1}(\cos \delta_s \sin RI \sin(\Omega - \Omega_s) + \sin \delta_s \cos RI) \]

\(-90^\circ \leq \beta \leq 90^\circ\)

(\(\delta_s\) declination of the sun, \(RI\) orbit inclination, \(\Omega\) right ascension of the ascending node, \(\Omega_s\) the right ascension of the sun)

- As \(\beta\) increases, SC passes over areas of the planet further from the subsolar point, thereby reducing albedo loads. Yet the SC is also in the sun for a larger percentage of each orbit as a result of decreasing eclipse times.

- \(\beta\) varies continuously with time because of orbit nodal regression and change in sun’s right ascension and declination over the year
[3]

Space Thermal Environment

[3.4]

Earth environment
Direct solar flux

- Sunlight is the greatest source of environmental heating on most S/C in Earth orbit

- Varies mainly because Earth orbit is elliptic (± 3.5% variation)

- Little variation with 11-year solar cycle (± 1% variation)

- Uncertainties on the values (± 0.4% variation)

- Flux at mean Sun-Earth distance (1 AU) falling at right angles = Solar constant (SC) = 1367.5 W/m²

The Earth albedo can be approximated by a mean value of $a=0.33$

Highly variable (spatial and temporal): different surface and atmospheric properties

<table>
<thead>
<tr>
<th>Surface</th>
<th>Albedo (a)</th>
</tr>
</thead>
<tbody>
<tr>
<td>Snow</td>
<td>0.8 – 0.9</td>
</tr>
<tr>
<td>Clouds</td>
<td>0.4 - 0.8</td>
</tr>
<tr>
<td>Deserts</td>
<td>0.25 – 0.30</td>
</tr>
<tr>
<td>Forests</td>
<td>0.05 – 0.10</td>
</tr>
<tr>
<td>Oceans</td>
<td>0.04 – 0.05</td>
</tr>
</tbody>
</table>

<table>
<thead>
<tr>
<th>Short-Term Albedo Correction</th>
<th>Orbit-Average Albedo Correction</th>
</tr>
</thead>
<tbody>
<tr>
<td>Position from Subsolar Point (deg)</td>
<td>Add Correction</td>
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</tr>
<tr>
<td>20</td>
<td>0.02</td>
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<tr>
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<tr>
<td>80</td>
<td>0.20</td>
</tr>
<tr>
<td>90</td>
<td>0.31</td>
</tr>
</tbody>
</table>

Albedo increases for large beta angles because sunlight is reflected off Earth with more forward scatter at low angles of incidence that occur near the terminator (correction must be considered)

Hand calculations can be carried out with a mean albedo of $0.33 (0.31-0.39)$ considering

- Earth VF (from abacus or explicit calculation)
- $\left(\frac{R_{\text{Earth}}}{R_{\text{orbit}}}\right)^2$ (only good if SC modelled by 1 point, or plate facing Earth)
- Lower albedo values near tropical regions and higher values in polar regions

- Long duration polar missions should consider the albedo increase at the poles
The Earth IR radiation can be approximated by a **255K black body** \((\sigma T_{Earth}^4 \sim 240 \text{ W/m}^2)\)

- Variable emission (spatial and temporal):
  - Increasing cloud cover tends to lower Earth-emitted IR because cold and effectively block upwelling radiation from Earth’s warmer surface below
  - Highest values of IR emission will occur in tropical and desert regions

- Hand calculations can be carried out with a IR radiation of 240 W/m² multiplied by
  - Earth VF (from abacus or explicit calculation)
  - \((R_{Earth} / R_{orbit})^2\) (only good if SC modelled by 1 point, or plate facing Earth)
- Higher loads near tropical regions and lower loads in polar regions since colder

- Long duration polar missions should consider the IR load reduction at the poles

LEO IR loads: 3-hours period

- We can see that the IR load can vary quite significantly for shorter periods.

- A low-mass device with a good radiative coupling to these environmental fluctuations (good Earth view factor) might exceed a temperature-stability limit that is particularly tight!
- Spinning cylindrical satellite in a 555km-altitude LEO

- Earth-emitted IR considered constant over Earth (independent of orbit inclination, RAAN or beta angle)

- If beta increases, eclipse time and albedo loads decrease

---

**Fig. 2.19. Cylinder in low Earth orbit.**
As orbit altitude increases, environmental loads from Earth (IR and albedo) decrease rapidly. Yet could be significant for cryogenic systems!

~3 W/m² albedo load (subsol point)

~5.5 W/m² IR load

During summer and winter the sun’s declination causes Earth’s shadow to be cast above or below the satellite orbit making eclipses impossible.

Typically 3-axis stabilized SC. Thus maximum load on North/South face is \( q_{\text{sol}} \times \sin(23.4^\circ) \) and the other surfaces receive a load that varies as a cosine variation from no load to full load.
Thermal control methods

Introduction
In order to maintain the components within allowable ranges of temperatures and to adjust correctly the powers, we need several means.

Two main categories:

- Passive methods (no mechanical moving parts, no moving fluids, no power needed)
- Active methods

**Advantage of passive methods**: simplest, reliable, low mass, low power, low costs, no constraints on SC design and tests

**Necessity of active methods**: allowable range of $T^\circ$ is tight and/or precise, large powers to be evacuated, variable environment, cryogenic applications, …
Thermal control methods

Coatings
Coatings (or surface finishes)

- External surfaces of a SC radiatively couple the SC to space.
  Because these surfaces are also exposed to external sources of energy (sunlight, albedo, planet IR), their radiative properties must be selected to achieve an energy balance at the desired temperature between SC internal dissipation, external sources of heat, and reradiation to space.

- Radiation is also very important inside the SC where we want adjust the powers as well.

- For these purposes, SC thermal designs employ wavelength-dependent thermal-control coatings, main interesting properties of which are: IR emissivity and solar absorptivity.
The equilibrium temperature of a surface in space only exposed to solar flux depends on the ratio $A_s/A$ and $\alpha/\varepsilon$ (one assumes here $T_{DS}=0K$ and normal incidence)

\[ T = \sqrt[4]{T_{DS}^4 + \frac{\alpha A_s \cos \theta}{\varepsilon A} \frac{q_s}{\sigma}} \]
Different types

- Large range of properties available as function of $\alpha/\varepsilon$ ratio

- Allow to ‘play’ with the ratio $\alpha/\varepsilon$ to adjust the powers and find allowable temperatures
Different types

- Thermal coating fall into four basic categories:
  - Solar absorber ($\alpha/\varepsilon > 1$) ['hot coatings']
  - Solar reflector ($\alpha/\varepsilon << 1$) ['cold coatings']
  - Flat absorber ($\alpha/\varepsilon \sim 1$)
  - Flat reflector ($\alpha/\varepsilon \sim 1$)
In practice

- Selection of coatings is mainly driven by $\alpha / \varepsilon$ ratio, degradation, ease of application, mass, electrical conductivity, cost

- Paints are most commonly used. All paints have high-$\varepsilon$, the choice is rather driven by $\alpha$

- Electronic boxes inside SC and structural panels to which they are attached usually painted to achieve a high $\varepsilon$ (black paint is a conventional choice). Therefore one can dissipate heat from electronic or make uniform the temperatures inside the SC or payload

- Internal temperature-sensitive components that do not dissipate much (propellant lines, tanks,…) often have low-$\varepsilon$ finish (bare or polished Alu, Gold)

- Outer-cover layer of insulation blankets usually with low-$\alpha$ to high- $\alpha$, high-$\varepsilon$ (black Kapton, aluminized Kapton,…) 

- Radiator coatings have low-$\alpha$ and high-$\varepsilon$ (second-surface mirrors, white paint, silvered Teflon, aluminized Teflon…)

- Also if discharge can be a problem, we can use conductive (electrically) coatings
Coating degradation

- Thermal-control finishes affected in orbit by
  - UV radiation
  - Charged particles
  - High vacuum
  - Contaminant films

- The general result is an increase in solar absorptivity with little or no effect on IR emissivity

- This is a problem, for example radiators that are oversized to handle high solar loads at end-of-life cause the SC to run much cooler in the early years of the mission, sometimes necessitating use of heaters to avoid undertemperatures of electronic components
Thermal control methods

[4.3] Insulation
MLI blankets prevent both excessive heat loss from a component and excessive heating from environmental fluxes, rocket plumes and other sources. They may also protect against micrometeoroids, atomic oxygen, electron charge accumulation.

- MLI are often cut-out to provide areas for radiators.

- MLI must be vented to allow the gases to evacuate during launch.

- MLI is composed of multiple layers of low-ε films (Mylar sheets) each with vacuum-deposited aluminium finish on one-side or multiple layers of low-ε films (Mylar sheets) each with vacuum-deposited aluminium or gold on both sides with additional flat spacers (silk or Dacron). One can also replace the flat spacers by embossed aluminized Kapton to increase performances.
MLI outer cover

- One of the most important factor influencing choice of MLI

- Should be opaque to sunlight, generate minimal amount of particulate contaminants, be compatible with environment and temperatures to which it is exposed during mission

- **EX:**
  - Moderate $\alpha/\varepsilon$ requirement for comfortable temperature in Sun (aluminized Kapton, Beta cloth)
  - Very low $\alpha/\varepsilon$ requirement to minimize impact of sun (silvered Teflon)
  - Electrostatic discharge requirement (black Kapton)
Thermal control methods

Radiators
SC waste heat is ultimately rejected to space by radiator surfaces that can be SC structural panels, flat-plate radiators, deployable radiators.

They reject heat by IR radiation which strongly depends on the temperature.

Most SC radiators reject between 100 and 350W of internally generated electronics waste heat per m². Weights typically from almost nothing to around 12kg/m².

Sizing considering operating $T^\circ$, worst-case waste heat, environmental heating, radiative and conductive interactions with other SC surfaces.

\[ Q = A\varepsilon\sigma T_{rad}^4 \]
[4]

Thermal control methods

[4.5]

Bolts, straps, braids,...
- The bolts are used for structural requirements but also for thermal control since it is the simplest conductive method.

- Straps are essential and common to link a dissipative unit such as an electronic box to a radiator.

Thermal control methods

Heaters
Heaters

- Heaters are often required to protect components under cold-case environmental conditions (operational mode) or to make up for heat that is not dissipated when an electronics box is turned off (non-operational mode)

- Most common type is the patch heater which consists of an electrical-resistance element sandwiched between two sheets of flexible electrically insulating material such as Kapton

- Control: usually mechanical or electronic thermostat
  - Temperature at which thermostat is turned on is called: set point
  - Difference between temperatures at which thermostat turns on and off: dead band
Heaters

- Control: usually mechanical or electronic thermostat
  - Temperature at which thermostat is turned on is called: set point
  - Difference between temperatures at which thermostat turns on and off: dead band

- A small dead band reduces the temperature swing of the device being heated and reduces power consumption a little (since average temperature lower). On the other hand, a small dead band also increases number of cycles on the thermostat itself and decreases its reliability. Dead band are usually larger than 4°C.

- Kapton-type heaters can dissipate up to about 10 W/cm² at ambient temperature

- Kapton-type heaters can be used up to 250°C ($T_{\text{melt}}$)
Thermal control methods

Louver
Louvers

- Useful to dump more / less power to space and accommodate large variations of energy (internal powers, interplanetary missions, ...) with little temperature change and saving heater power.

- Principle: low-\(\epsilon\) parallel blades can rotate and uncover a high-\(\epsilon\) radiator. As a result, there is variation of \(\epsilon\).
[4.8] Heat pipes
Principle

- The common application is one requiring a physical separation of the heat source (hot component) and sink (generally spacecraft radiator).

- Main advantage is a very large heat transfer capability with a limited difference of temperature and limited mass. Main drawback is the difficulty of testing.

- **Principle**: closed two-phase liquid-flow cycle with evaporator and condenser to transport relatively large quantities of heat from one location to another without electrical power.

When heat is applied to evaporator, liquid in a wick structure uses this energy to evaporate. High temperature and corresponding high pressure in this region result in flow of the vapor to condenser. At this cooler region of wick structure, vapor condenses, giving up its latent heat of vaporization. Capillary forces in wicking structure then pump the liquid back to evaporator. The mid-region can be considered adiabatic.

- Three major components in such a device:
  - a working fluid,
  - a container
  - a wick structure
Latent heat of vaporization for most working fluids is high so that only small amounts of fluid need to flow to transport significant quantities of heat. This also entails a small temperature difference between the evaporator wall and the condenser wall.

Selection of working fluid is mainly based on an operating temperature range and boiling temperature that must be adequate, and the fluid must be stable over the entire range.

### Table: Working Fluid Properties

<table>
<thead>
<tr>
<th>Fluid</th>
<th>Melting Point (°C)</th>
<th>Boiling Point (°C)</th>
<th>Critical Point (°C)</th>
</tr>
</thead>
<tbody>
<tr>
<td>Methane (CH₄)</td>
<td>-182.5</td>
<td>-161.8</td>
<td>-82.6</td>
</tr>
<tr>
<td>Methanol (CH₃OH)</td>
<td>-97.9</td>
<td>64.8</td>
<td>240.0</td>
</tr>
<tr>
<td>Acetone (CH₃COCH₃)</td>
<td>-93.2</td>
<td>56.25</td>
<td>235.1</td>
</tr>
<tr>
<td>Ammonia (NH₃)</td>
<td>-77.7</td>
<td>-33.4</td>
<td>132.4</td>
</tr>
<tr>
<td>Water (H₂O)</td>
<td>0 (.05)</td>
<td>100</td>
<td>374.2</td>
</tr>
</tbody>
</table>
Thermal control methods

Pumped fluid loops
Pumped fluid loops (PFL)

- Usually for large powers to be dissipated, environmental changes, cryo applications,…

- PFLs are devices that provide efficient transfer of a large amount of thermal energy between two points by means of forced liquid convective cooling

- Cooling can be accomplished by the use of a coolant as the thermal energy transport agent

- The coolant absorbs the dissipated energy from a component and transfers it to a heat sink

- Final rejection process depends on wheter the coolant is expendable (rejected from the space vehicle) or nonexpendable (working fluid is recirculated within the system once its thermal energy has been radiated to space via a radiator)
Spacecraft thermal analysis

Introduction
Lumped parameter method

- We exposed the lumped parameter method. This general network allows us to discretize a model into several nodes.

- The conservation of energy for each node permits to deduce the temperatures in the model

\[
Q_i + \sum_j GR_{ij}\sigma(T_j^4 - T_i^4) + \sum_j GL_{ij}(T_j - T_i) = C_i \frac{dT_i}{dt}
\]  

- \(Q_i\) can be either a dissipative power or an incoming power (Solar flux absorbed and reflected)

- The linear conductances \(GL\) can take into account the conduction and convection

- Boundary conditions can be:
  - constant temperature (Dirichlet), e.g. deep space
  - constant flux (Neumann), e.g. flux towards deep space

- Any finite difference model is based on this equation

---

Before any numerical simulations are carried out, hand-calculations should be performed (useful to assess the parameters and solutions) and can be done for a few nodes (lumped parameter method)

First the implementation consists in a GMM (geometric mathematical model) which is the geometry of the design implemented as shells (basic entity) with their thermo-optical properties, orbit and environment parameters and contains the results of the radiative analysis.

The radiative analysis is carried out by calculating the VF or REFs. Once the radiative couplings are known, all the external radiative fluxes (solar, albedo, planet IR) and internal radiative conductances are given.

Then the TMM (thermal mathematical model) which is the expression of the network of nodes (lumped parameter method) can be fully defined with the conductive conductances

\[
Q_{i} + \sum_{j} GR_{ij} \sigma (T_{j}^4 - T_{i}^4) + \sum_{j} GL_{ij} (T_{j} - T_{i}) = C_{i} \frac{dT_{i}}{dt}
\]

The equation can then be solved with the finite difference method (commonly used) to predict the temperatures.
A lot of uncertainties come from:

- Uncertain parameters (k, α, ε, contact conductances, Qdissipated…)
- Uncertain parameters of space environment (solar flux, IR, albedo, coating degradation, …)
- Physical parameters of the SC (lengths, …)
- Numerical model (discretization, calculations of radiative couplings, …)
- Parameters of ground testing to correlate mathematical model

In theory, a sensibility analysis could be performed for the temperature as function of most influent parameters. However, spacecraft thermal engineer rather simplify the analyses by considering ‘worst cases’ scenarios

- **Hot case** (α max at the EOL, components ON, maximum environmental heat loads, …)
- **Cold case** (α min at the BOL, components OFF, minimum environmental heat loads, …)

If required combined cases can be done to provide more scenarios (eg, hot case with components OFF) and give a better idea of the design behavior.

We can run the analyses with the most influent parameters considering these cases

In addition we commonly add to the worst cases predicted temperatures a safety margin of 17°C during design phase reduced to 11°C after testing
As for the mechanical analyses, a reduced model of a subsystem is necessary

Need to decrease number of nodes in the global model of a S/C. Existence of several teams (eg. one for an instrument, another one for the S/C). S/C team only interested in interfaces with instrument and some of its components

Typically we want to keep some relevant nodes (ie associated with critical components or if a large gradient of temperature exists). The global balance of the instrument (power in = power out) shall be respected and the powers through the interfaces with S/C identical
[5]
Spacecraft thermal analysis
[5.2]
Examples
Example 1: electronic unit on a platform
Example 2: circular polar orbit
Example 3: EUVI Focal Plane Assembly

<table>
<thead>
<tr>
<th>Surface Description</th>
<th>Surface Coating</th>
</tr>
</thead>
<tbody>
<tr>
<td>Radiator (6063 Al)</td>
<td>Front: White Paint NS-43C</td>
</tr>
<tr>
<td></td>
<td>Back: Gold Irridite</td>
</tr>
<tr>
<td>Cold Finger (6063 Al)</td>
<td>Gold Plated</td>
</tr>
<tr>
<td>CCD Cold Cup (6063 Al)</td>
<td>Black Anodize internal Gold Plating External</td>
</tr>
<tr>
<td>CCD Chip Carrier (Invar)</td>
<td>As Machined</td>
</tr>
<tr>
<td>Cold Cup Radiation Shield (6063 Al)</td>
<td>Gold Plating</td>
</tr>
<tr>
<td>Structure (6AL-4V Titanium)</td>
<td>Gold Plating</td>
</tr>
</tbody>
</table>

Example 3: EUVI Focal Plane Assembly

OP HOT CASE

NOP Hot Decontamination Case

Example 3: EUVI Focal Plane Assembly

**OP Hot Case**

-81 °C
-36 °C
-73 °C
-79 °C
-83 °C
-21 °C

-0.17 W
-0.25 W
-0.47 W
-0.09 W
0.56 W
0.39 W
1.86 W
1.39 W
0.05 W
0.54 W
2.35 W
1.91 W

**NOP Hot Decontamination Case**

23 °C
25 °C
77 °C
22 °C
23 °C

0.33 W
11.4 W
10.77 W
0.43 W
0.12 W
0.02 W
0.12 W
11.12 W
11.0 W
11.10 W
0.02 W
11.1 W
0.02 W
0.17 W

84 °C
74 °C
47 °C
21 °C
21 °C

9.79 W
25 °C
25 °C

18 °C
17 °C

0.05 W
0.17 W
0.39 W
0.30 W
0.39 W
0.27 W
0.10 W
0.07 W
0.05 W
0.02 W
0.02 W
0.05 W
0.17 W

81 °C
-79 °C
-83 °C
-21 °C

0.09 W
0.47 W
0.27 W
0.11 W

18 °C
17 °C
-6 °C
-36 °C
-37 °C
-79 °C
-85 °C

9.79 W
84 °C
-11 °C

Spacecraft thermal testing

Introduction
Why?

- Thermal modeling involves many assumptions and uncertainties and the design relies on a worst case approach.
- Space missions are very costly and we have only one SC that will flight.
- Tests are thus essential for the verification and to ensure the satellite achieves all the requirements so that the mission is not jeopardized!
1. **Engineering or Development Tests** are conducted throughout a satellite design in each subsystem to validate new design concepts or perform measurements to reduce uncertainties.

2. **Qualification Tests** are conducted to demonstrate that the design implementation and manufacturing process have resulted in hardware and software that meets specification requirements with sufficient margins.

3. **Flight Acceptance (FA) Tests** are conducted to verify conformance to specification requirements and provides quality-control assurance to detect workmanship deficiencies, manufacturing errors or any latent defect that would be detected by normal inspection techniques.
   - Acceptance testing is less severe than qualification testing and are conducted under environmental conditions no more severe than those expected during the mission.
   - Nominal test program = acceptance tests following qualification tests. But often costly, time-consuming,…

**Alternative methods** are protoqualification (or protoflight) testing (introduce higher risks compared to the nominal but increasing safety factors and development tests can mitigate the induced risks.).

The tested protoflight model (PFM) is thus considered eligible for flight. Protoflight testing accomplishes thus in one test the combined purposes of design qualification and flight acceptance and results from a combination of the two strategies.
The above strategies concerned both thermal and vibration tests. More specifically, thermal test are divided into three tests:

1. **Thermal Vacuum Test (TV)** is performed under vacuum and subjects the satellite (or equipment) to worst hot/cold temperatures including adequate margins. Its purpose is the performance verification through functional testing because it is the most realistic ground simulation of the in-orbit environment.

2. **Thermal Cycling Test (TC)** that can be performed under ambient pressure, subjects the satellite (or equipment) to a series of cycles of hot and cold temperature plateaus. Its main purpose is to reveal latent workmanship defects due to environmental stress.

3. **Thermal Balance Test (TB)** is generally conducted as a part of the thermal vacuum test. It has two primary purposes: demonstrate the ability of the thermal control system to maintain temperatures within the specified operational limits and provide data for the TMM correlation. These three tests can be combined in one test called thermal vacuum cycling test during which the TB test is also performed.
To account for the uncertainties, safety margins are applied to the worst case predicted temperatures. The resulting temperature constitute the acceptance temperature range.

To this acceptance temperature, extra margin are added for protoqualification testing.

Qualification temperature range are usually the same as the protoqualification range but additional qualification margins can be added to increase environmental conditions over that expected during the lifetime or include the cycles duration as well as any other increase in severity to demonstrate the robustness of the design.


Spacecraft Thermal Control
First ambient temperature and decrease of pressure. Then baking sequence (up to $T_{NO\text{-max}}$). After a dwell time $t_E$, decrease to max start up temperature ($T_{SU\text{-max}}$) and stabilized at max operating temperature ($T_{Q\text{-max}}$). Once stabilized (time $t_E$) the functional and performance tests (including Thermal Balance tests) are performed.

After that it is switched off again to decrease and maintained at the minimum non-operating temperature ($T_{NO\text{-min}}$) during a time $t_E$. The temperature is then increased to the minimum start-up temperature and the equipment is switched on. When stabilized at the low operating level ($T_{Q\text{-min}}$), and after the time $t_E$, the functional tests in cold case are performed.

SC is then cycled between $T_{Q\text{-max}}$ and $T_{Q\text{-min}}$ until the number of cycles required is achieved. Functionally tests are again performed during the last cycle at $T_{Q\text{-max}}$ and $T_{Q\text{-min}}$ after which the temperature is raised to ambient conditions and the final functional and performance test can be performed.
As defined above, the protoflight approach combines acceptance and qualification: levels are the qualification ones while duration are as acceptance ones. Nevertheless 6 cycles could be performed.

<table>
<thead>
<tr>
<th>Parameter</th>
<th>Condition / Value</th>
</tr>
</thead>
<tbody>
<tr>
<td>Start cycle</td>
<td>Hot</td>
</tr>
<tr>
<td>n (number of cycles)</td>
<td>8</td>
</tr>
<tr>
<td>$t_E$ (dwell time at $T_{\text{hot}}/T_{\text{cold}}$)</td>
<td>2 h</td>
</tr>
<tr>
<td>$dT/dt$ (temperature rate of change)</td>
<td>&lt; 20 °C/min</td>
</tr>
<tr>
<td>Stabilization criterion</td>
<td>1 °C/h</td>
</tr>
</tbody>
</table>

$a$ $T_{\text{hot}} = T_{Q-\text{max}}$ or $T_{\text{NO-\text{max}}}$.  
$b$ $T_{\text{cold}} = T_{Q-\text{min}}$ or $T_{\text{NO-\text{min}}}$.  

<table>
<thead>
<tr>
<th>Levels</th>
<th>Duration</th>
</tr>
</thead>
<tbody>
<tr>
<td>Qualification margins</td>
<td>Acceptance margins</td>
</tr>
<tr>
<td>10 °C</td>
<td>5 °C</td>
</tr>
<tr>
<td>Qualification</td>
<td>Acceptance</td>
</tr>
<tr>
<td>8 cycles</td>
<td>4 cycles</td>
</tr>
</tbody>
</table>
[6]
Spacecraft thermal testing
[6.2]
Examples
Examples

- ROSETTA
- Planck
- BepiColombo
- EIT
Examples

ESTEC LSS (Large Space Simulator)
[7] Spacecraft thermal design examples
Satellite “skin”

[Image of SOHO, SPOT-5, ARTEMIS, XMM, ISO, PROBA 1, with labels for EIT RADIATOR, MLI, FRONT BAFFLE RADIATOR, radiators, and MLI]

[7] Spacecraft thermal design examples
Hubble Space Telescope

[7] Spacecraft thermal design examples
Infrared Astronomical Satellite (IRAS) Sunshade

- Inner shield surface specularly reflects Earth Limb IR
- Black painted radiators have high emissivity, no solar exposure
- Inner cone self-emission less than $1 \times 10^{-16}$ W at the detector
- Twenty-four low heat leak fiberglass rods support sunshade outer stage off of outer-shell girth ring
- Outer MLI blanket protects outer radiator from solar
- Inner radiator isolated from other radiator by staged MLI blankets
- Sunshade protects aperture from ram effect molecular contamination

IRAS external thermal control finishes

- Outer radiator (97 to 136 K)
- Vacuum deposited gold on Coilzak

Worst case Earth limb IR
Worst case solar

MLI blanket outer service
- Second surface silvered Teflon

Inner radiator (72 to 106 K)
Intermediate radiator (82 to 112 K)

Aperture
Sunshade support structure

Sunshade three-stage radiative cooler

Spacecraft thermal design examples
Planck satellite

Spacecraft thermal design examples

Telescope

Payload Support Structure

V-Grooves "Tulip"-Extension

Service Module
Planck satellite

Coolers

40K: passive cooling ≈2W

18K: H₂ J-T Sorption pumps (JPL, USA) ≈1W

4K: He J-T Mech. Pump (RAL, UK) ≈15mW

1.6K: J-T expansion 0.5mW

0.1K 3He/4He dilution 0.2μW

(AL, CRTBT, IAS, France)
Conclusion

[8]
Spacecraft Thermal Control is applicable at any level (S/C, subsystem, payload,…)

Radiative and conductive heat transfer are predominant in space

Space is a hostile thermal environment

Hand calculations are recommended for preliminary analyses but numerical simulations are necessary for more precise predictions of temperatures

**Philosophy of thermal control**

- adjusting the powers exchanges with space environment (radiative control) and inside the spacecraft (conductive and radiative control)
- with many means of control (passive or active)
- using a "worst case" approach
- adding margins to take account the uncertainties!
- finally testing to ensure that the design satisfies the initial requirements

**Much more detail is provided in the course “Conception d’expériences spatiales”, 2nd master and practical with ESATAN-TMS are given**
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