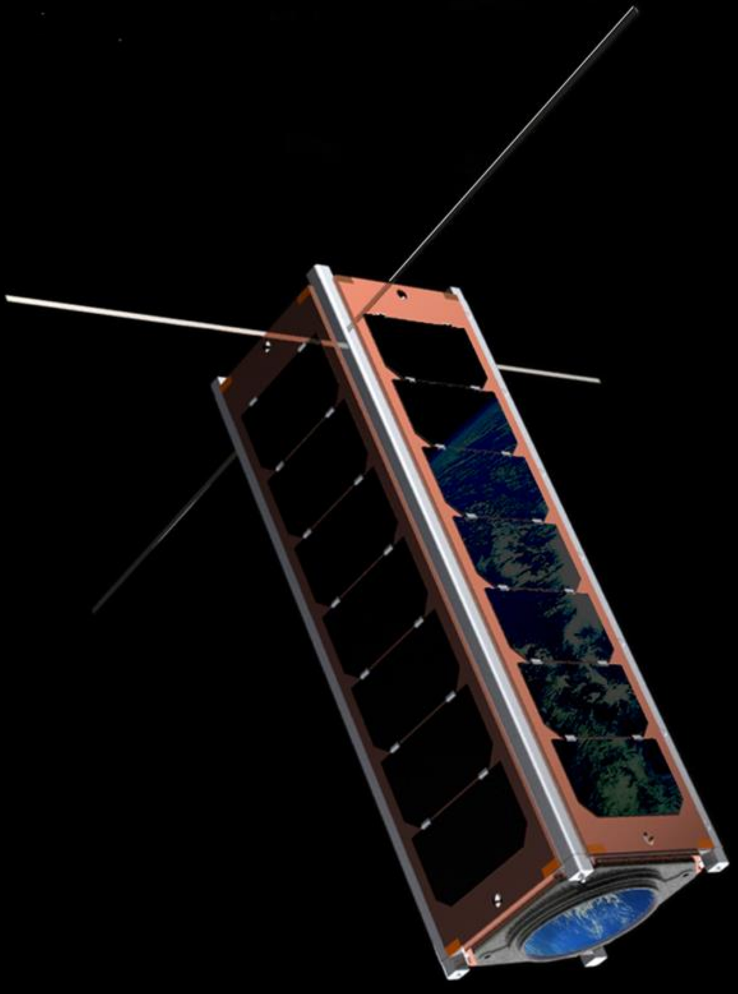

MEA_AM30 Space Technologies/Διαστημικές Τεχνολογίες

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ΠΑΝΕΠΙΣΤΗΜΙΟ ΠΑΤΡΩΝ

UNIVERSITY OF PATRAS

THERMAL ENVIRONMENT AND THERMAL DESIGN

Introduction

In this section we shall:

- review the physics of heat transfer;
- describe a thermal model of a spacecraft;
- study thermal control techniques;
- examine thermal design verification.

Spacecraft Thermal Design

- The Need for Thermal Design:
  - On Earth, objects operate in a relatively benign **thermal environment**.
  - **Heat transfer** is effected *via* one or more of the processes of [ΤΡΟΠΟΙ ΜΕΤΑΦΟΡΑΣ ΘΕΡΜΟΤΗΤΑΣ):

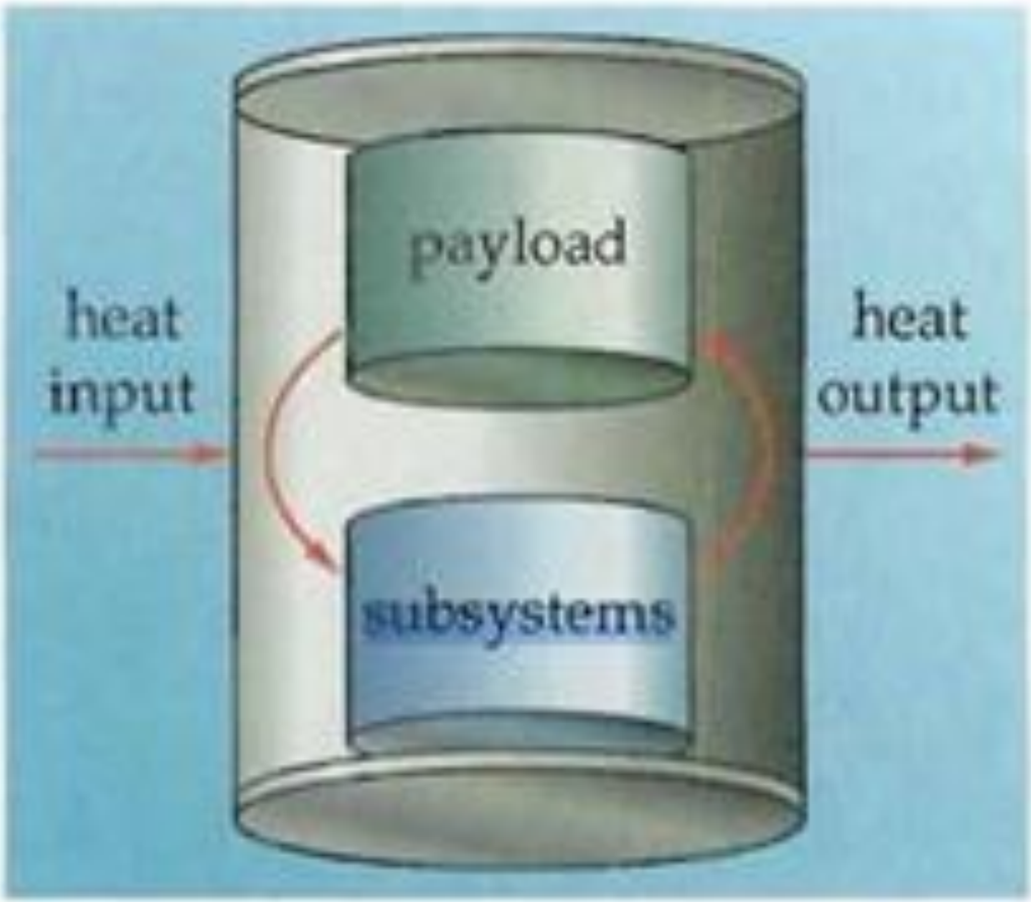
    - conduction (when the objects are in direct contact); **[ΑΓΩΓΙΜΟΤΗΤΑ]**

    - convection (i.e. by the motion of a fluid [liquid or gas]); **[ΣΥΝΑΓΩΓΗ]**

    - radiation (by the interchange of electro-magnetic radiation *-photons).* **[ΑΚΤΙΝΟΒΟΛΙΑ]**

  - The process of heat transfer usually results in a change in an object's **temperature** -which we can measure, and which we often wish to control.

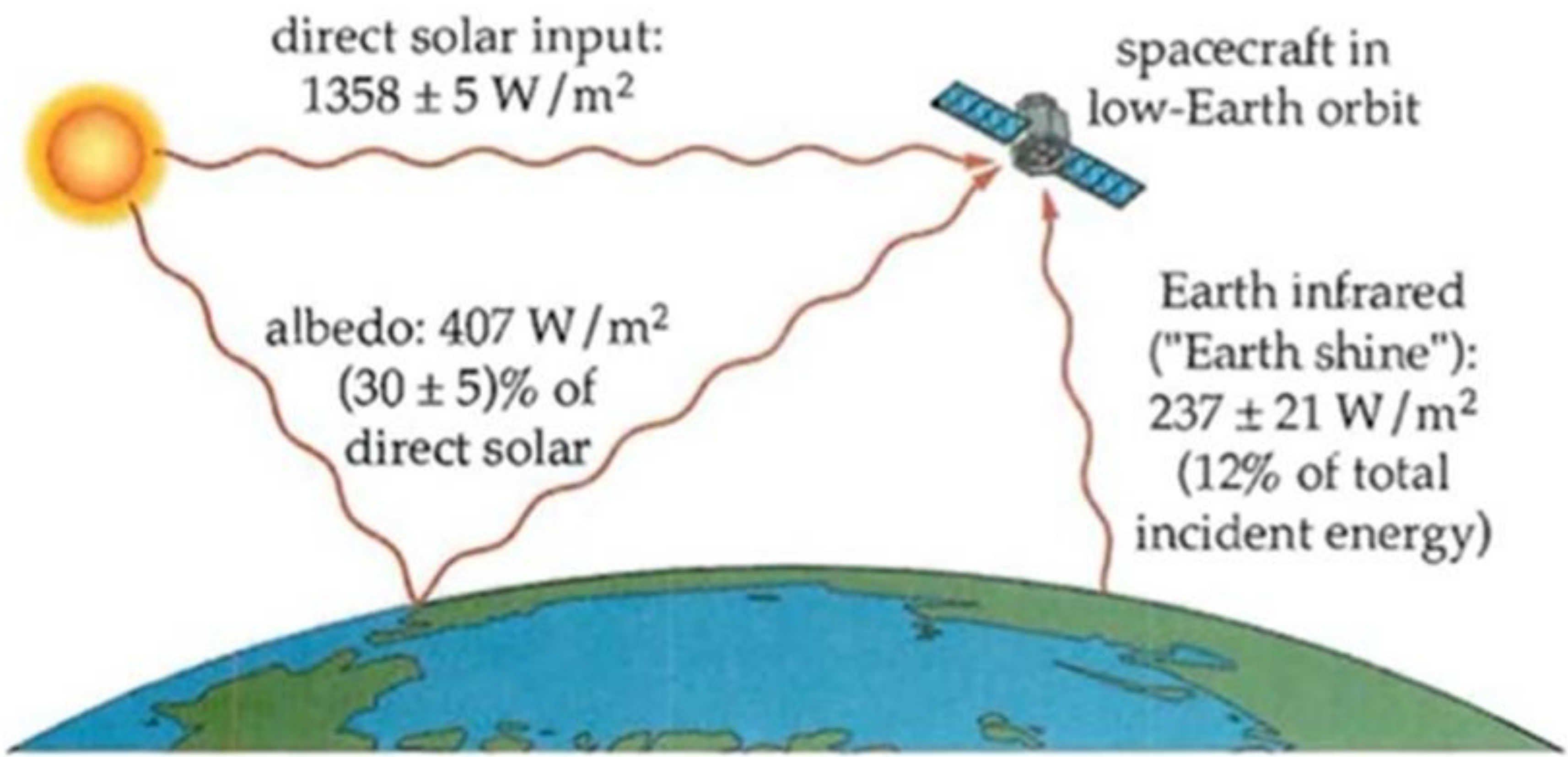
Thermal Balance



Heat Output= Heat Input + Internal Heat

Heat Sources in Space

- Η θερμότητα στο διάστημα προέρχεται από τρεις κύριες πηγές: Τον Ήλιο, Γη, εσωτερικές πηγές



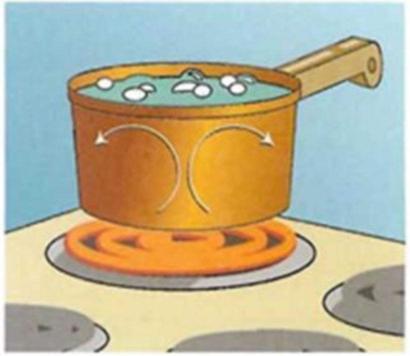
Albedo: Λευκαύγεια/ανακλαστικότητα Γης

Heat Transfer

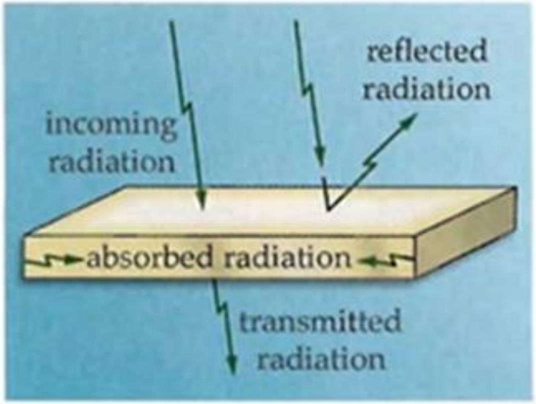
Η θερμότητα μεταφέρεται από το ένα σημείο στο άλλο με:

- Αγωγιμότητα (conduction)

- Συναγωγή (Convection)



- Ακτινοβολία (radiation)



Spacecraft Thermal Design

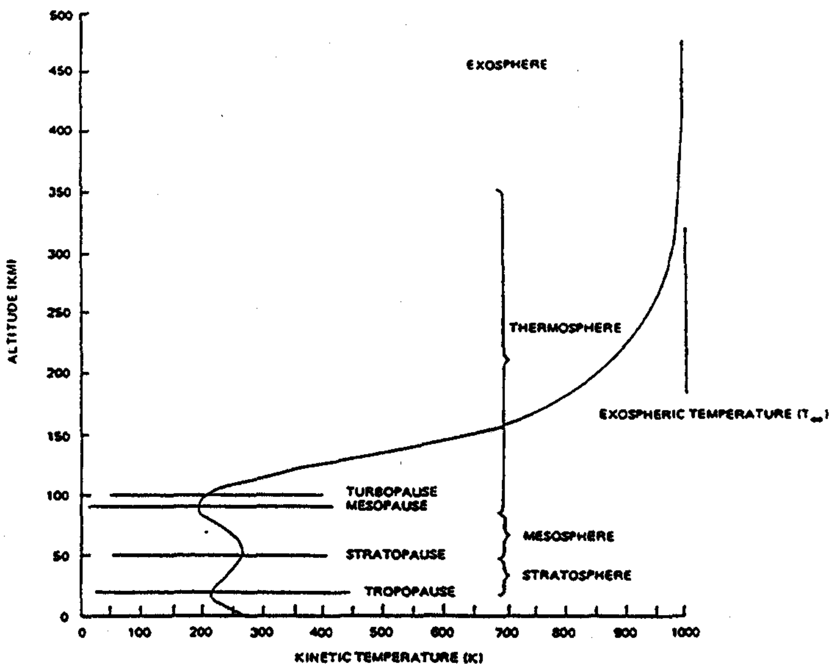
- The temperature limits for typical spacecraft components are:

Electronic Equipment (typical, operating)	0 °C to +70 °C
Electronic Equipment (MILSPEC)	-55 °C to +125 °C
Battery (NiCad Cells)	0 °C to +20 °C
Attitude Control Fuel (Hydrazine -N_2H_4)	+9 °C to +40 °C
Bearing Mechanisms	-45 °C to +65 °C
Solar Cells (typical)	-60 °C to +60 °C

- The aim of the thermal design process to to ensure that all components operate within appropriate temperature bounds, given all the possible thermal environments that the spacecraft may encounter.

Spacecraft Thermal Design

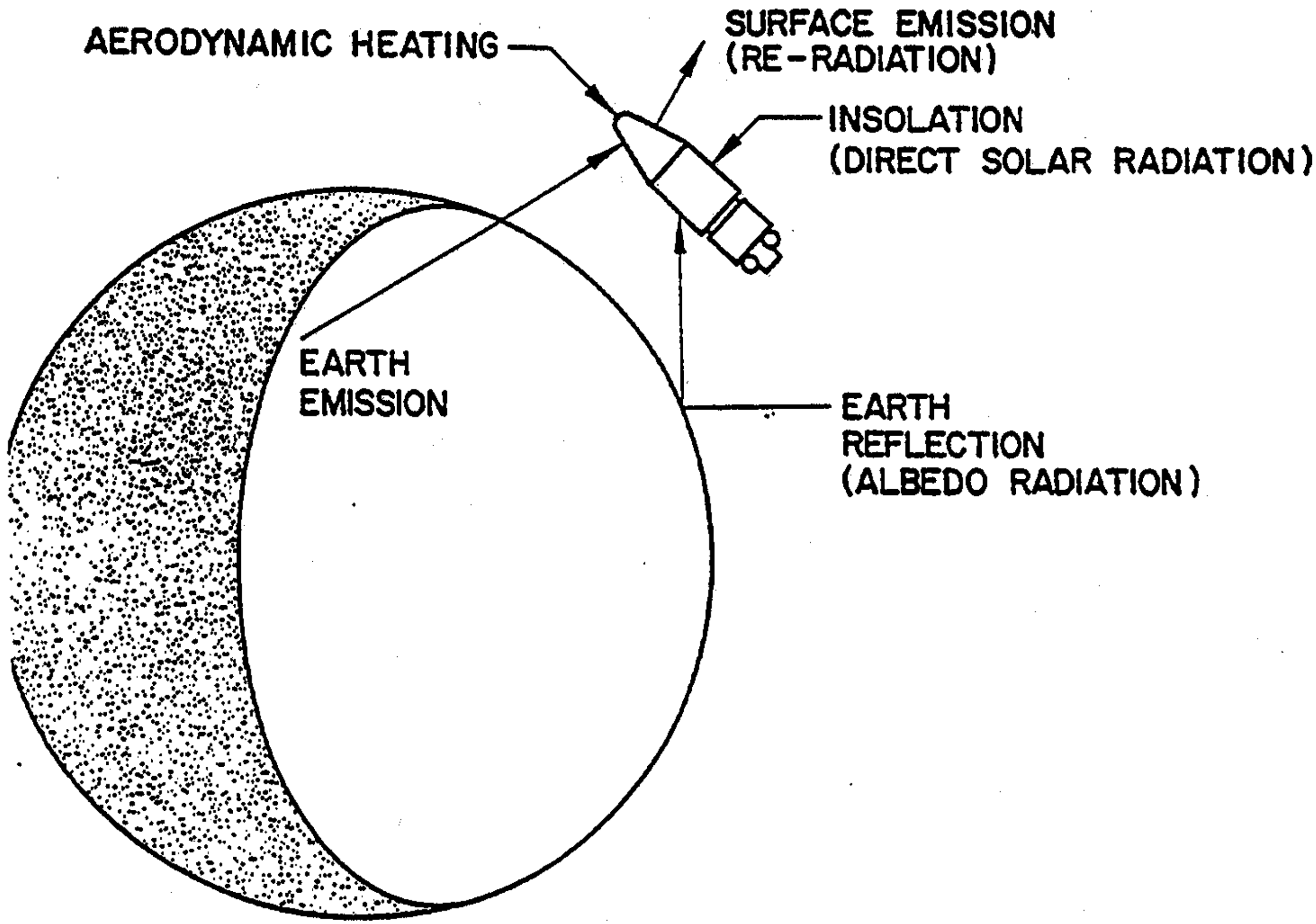
- Factors Controlling Temperature:
  - In near-Earth space, the temperature of the residual atmosphere (**Exosphere**) is ~1000 K (~730 °C).
  - However, spacecraft will <u>not</u> attain thermal equilibrium with the atmosphere as the **free-mean path** of the atmosphere's particles is much larger than the spacecraft dimensions
  - Heat Transfer is by **radiation**.



Mean Atmospheric Temperature as a Function of Altitude

Spacecraft Thermal Design

- Spacecraft Thermal Environment

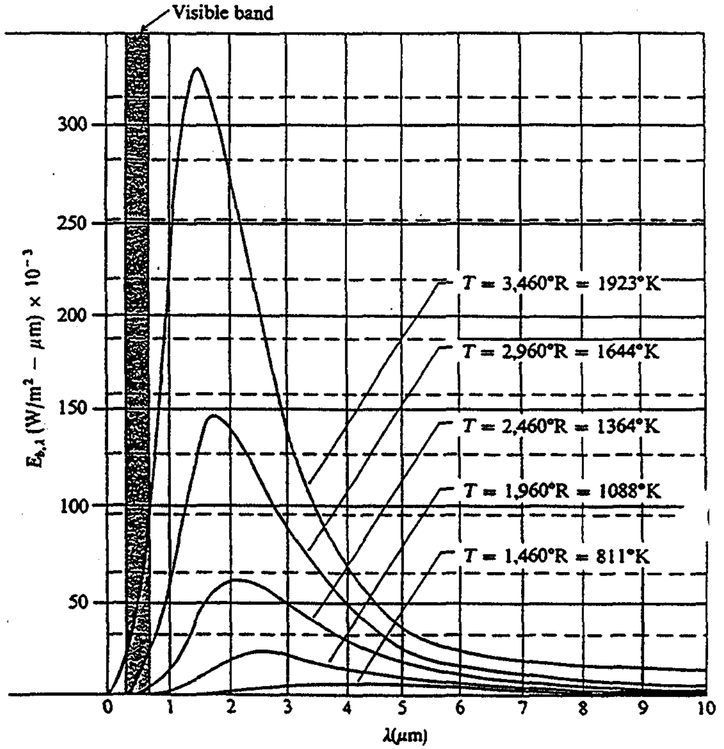


Earth's reflectance/albedo Λευκαύγεια/ανακλαστικότητα

Spacecraft Thermal Design

- Blackbody Radiation:
  - The radiation emitted by a body is a function of its **absolute temperature**.
  - The **Planck curve** describes the **radiance** (power per unit area per unit wavelength) of a perfect radiator (**Blackbody/μέλαν σώμα**) as a function of temperature.
  - The spectrum shows a peak emission at a wavelength given by **Wien's Law**:

$$\lambda_{peak} T = 2898 \ [\mu m. K]$$



Spacecraft Thermal Design

- Stefan-Boltzmann Law:
  - If we integrate the Planck function over all wavelengths [μήκος κύματος] (i.e. find the area under the curve), then we find the total power emitted by a blackbody per unit area. The result is remarkably simple:

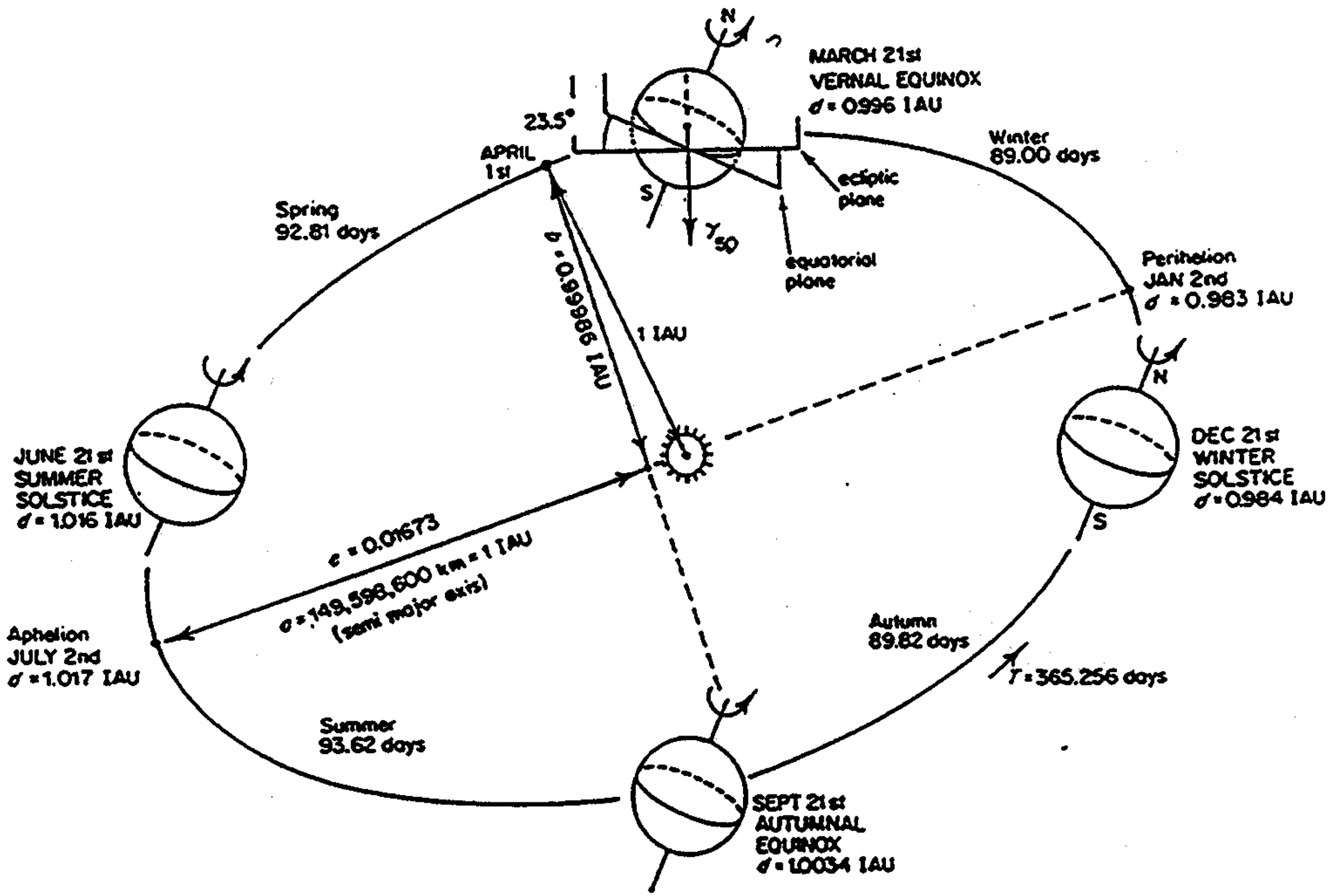
    $\mathbf{E_b(T) = \sigma T^4}$ [$W.m^{-2}$]

    where: $\sigma = 5.67 \times 10^{-8}$ $W.m^{-2}\,K^{-4}$ *(Stefan-Boltzmann Const.)*

  - Taking your body surface temperature to be ~300K (27 °C) - then you have a peak emission at 9.7 μm (infra-red) and you are radiating 459 W for every m^2 of your body area. Fortunately the room is also radiating heat at you. If the room is at ~20 °C, then you only have a net loss of ~0.1-0.2 mW per m^2 due to radiation.

Spacecraft Thermal Design

- The Earth's Orbit around the Sun:



Spacecraft Thermal Design

- Emissivity (ικανότητα εκπομπή):
  - The **spectral hemispherical emissivity** is the ratio of a body's actual **radiance** (also known as its **spectral radiation flux**), q_λ to the radiance of a blackbody (also known as the **blackbody emissive power**), $E_{b\lambda}$, at the same temperature:

    $$\varepsilon_\lambda = q_\lambda (T) / E_{b\lambda} (T)$$

  - Averaging over all wavelengths gives the **hemispherical emissivity**, ε, - otherwise known as just the **emissivity** or sometimes the **emittance**.
  - An emissivity of 1 means that the body emits radiation perfectly; an emissivity of 0 means that it does not emit radiation at all. All real objects are somewhere in-between.

Spacecraft Thermal Design

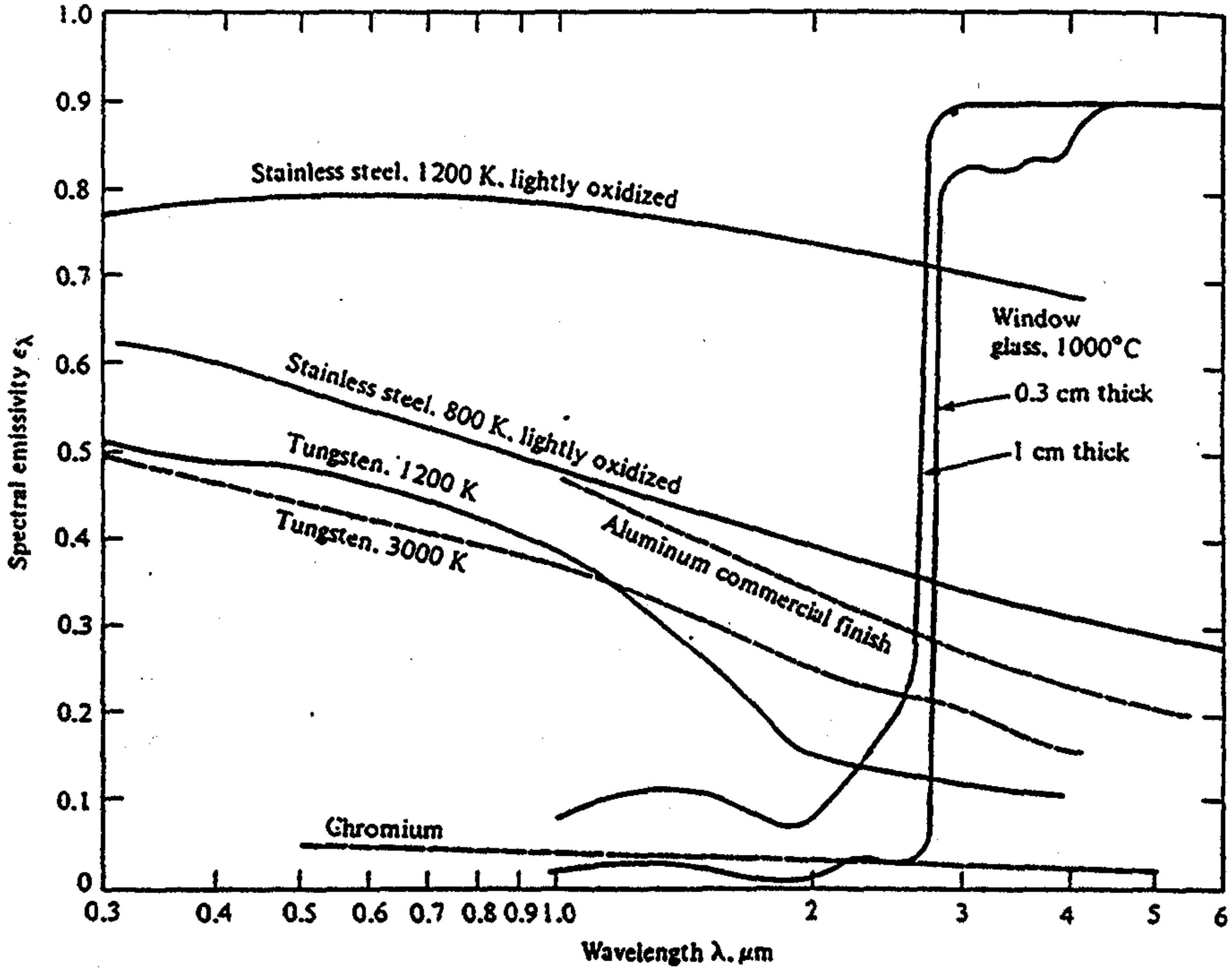
- Emissivity:
  - The **emissivity** is used as a weighting factor in order to calculate the flux emitted by an object. Thus:

    $\mathbf{E_b = \sigma T^4}$ [W. m^{-2}] (for a blackbody)
    $\mathbf{E = \varepsilon \sigma T^4}$ [W.m^{-2}] (for a body of emissivity ε)

  - In general, paints, organic materials and glass have high emissivities (e.g. black paint, ε~0.90; white paint, ε~0.95); polished metals have low emissivities (e.g. silver ε~0.01; aluminium ε~0.05); oxidised metals have higher emissivities (e.g. oxidised aluminium ε~0.2).

Spacecraft Thermal Design

- Emissivity:



Spacecraft Thermal Design

- Absorptivity (ικανότητα απορρόφησης) :
  - If q_λ^i (T) is the spectral radiation flux incident upon a surface, and q_λ^a (T) is the amount of radiation absorbed, then the **spectral hemispherical absorptivity**, α_λ is defined as:

$$\alpha_\lambda = q_\lambda^a(T) \,/\, q_\lambda^i(T)$$

  - Averaging over all wavelengths gives the **hemispherical absorptivity**, α, - otherwise known as just the **absorptivity** or sometimes the **absorptance**.
  - An absorptivity of 1 means that the body absorbs radiation perfectly (perfectly black); an absorptivity of 0 means that it does not absorb radiation at all - i.e. it is perfectly **reflective** or perfectly **transparent**.

Spacecraft Thermal Design

- Absorptivity:
  - The **absorptivity** is used as a weighting factor in order to calculate the power absorbed by an object. Thus:

    $\mathbf{Q}_b = \mathbf{A}_{projected} \cdot \phi$ [W] (for a blackbody)

    $\mathbf{Q} = \alpha\, \mathbf{A}_{projected} \cdot \phi$ [W] (for a body of absorptivity α)

    where $A_{projected}$ = the area projected towards the flux, ϕ.

  - Paints can have a wide range of absorptivities
    (e.g. black paint, $\alpha \sim 0.95$; white paint, $\alpha \sim 0.2$);
    polished 'silver-coloured' metals have low absorptivities
    (e.g. silver $\alpha \sim 0.1$; aluminium $\alpha \sim 0.2$);
    'coloured metals' have higher absorptivities
    (e.g. gold $\alpha \sim 0.5$).

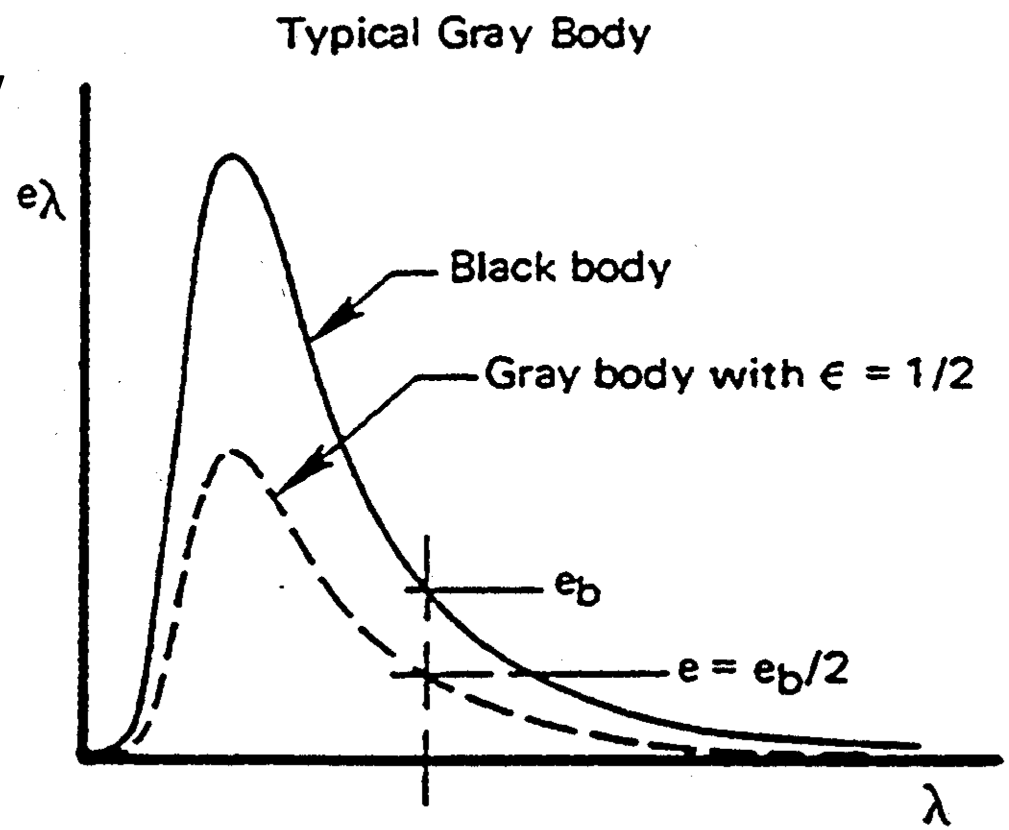
Spacecraft Thermal Design

- Reflectivity and Transmissivity (ανακλαστικότητα/ Διαπερατότητα):
  - An object may reflect radiation **specularly** (like a mirror); **diffusely** (like a sheet of white paper), or somewhere in-between. For a diffuse reflector (**Lambertian**) then we can define the **hemispherical reflectivity**, ρ, in a similar way to the absorptivity - in this case the numerator is the amount of radiation reflected.
  - Similarly, an object may be semi-transparent (like glass), and we may define the **hemispherical transmisivity**, τ, similarly, with the numerator now the amount of radiation transmitted.
    - For an opaque object: $\alpha + \rho = 1$
    - For a semi-transparent object: $\alpha + \rho + \tau = 1$

Spacecraft Thermal Design

- Gray body:
  - In thermal analysis the **gray body assumption** is often made to simplify the problem -that is α_λ, ε_λ, *etc.* are assumed to be uniform over the wavelength range of interest.
  - Under the grey body assumption, **Kirchhoff's Law** applies -that is

$$\alpha = \varepsilon$$



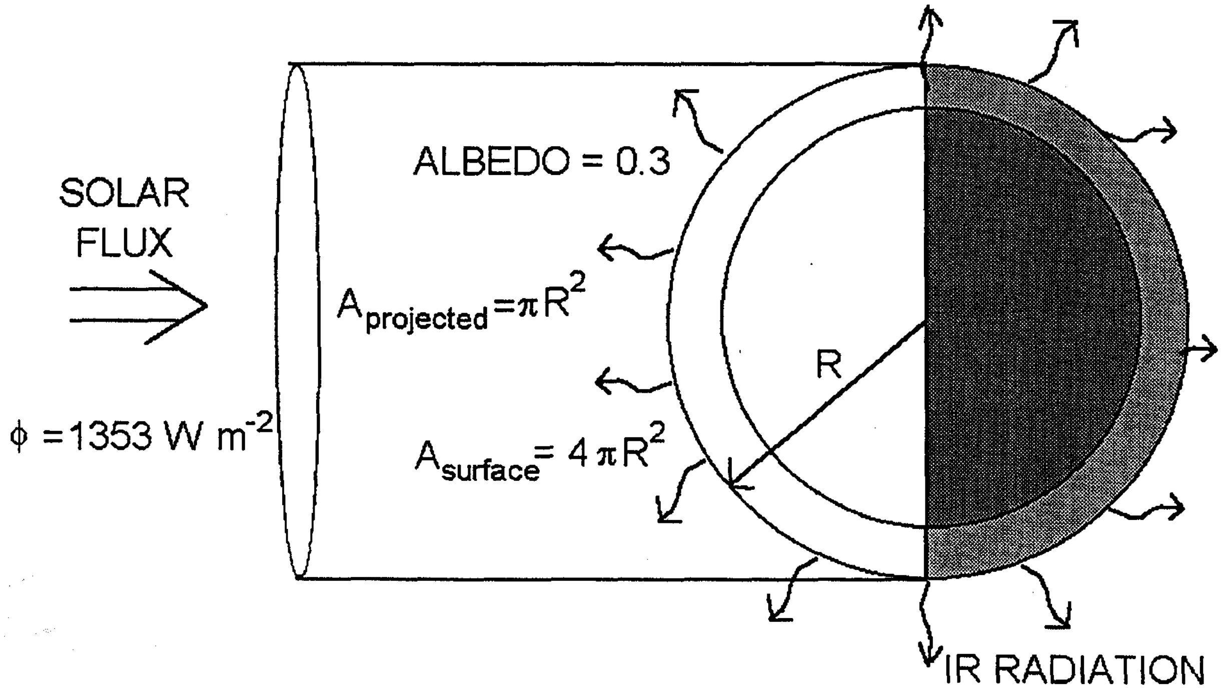
Spacecraft Thermal Design

- Predicting the Temperature of Earth:
  - We can use the blackbody relationships to predict the temperature of the Earth:
  - The **albedo** of Earth is ~30% - that is approximately 30% of the solar flux is reflected straight back to space on average.
  - Thus, the effective **absorptivity (α)** of Earth is 70% (as $\alpha + \rho = 1$ for an opaque body).
  - So the total **power absorbed** [W] by the Earth is:

    $Q_{absorbed} = 70\% \, \phi \, A_{projected}$; where $\phi = 1353$ W. m^{-2} at 1 AU

  - Assuming Earth to be a sphere, $A_{projected} = \pi R^2$; R= Radius of Earth.

Spacecraft Thermal Design



Spacecraft Thermal Design

- Now assuming the Earth to have no internal source of heat, the planet will reach **equilibrium temperature** when the thermal power radiated, $Q_{radiated}$, is equal to the thermal power absorbed,
  $Q_{absorbed}$ (i.e. *power* in = *power* out).

- The **power radiated** is:
  $Q_{radiated} = A_{surface}\, \varepsilon\, \sigma\, T^4$ [W]; (ε = 1 assuming Earth is a blackbody)

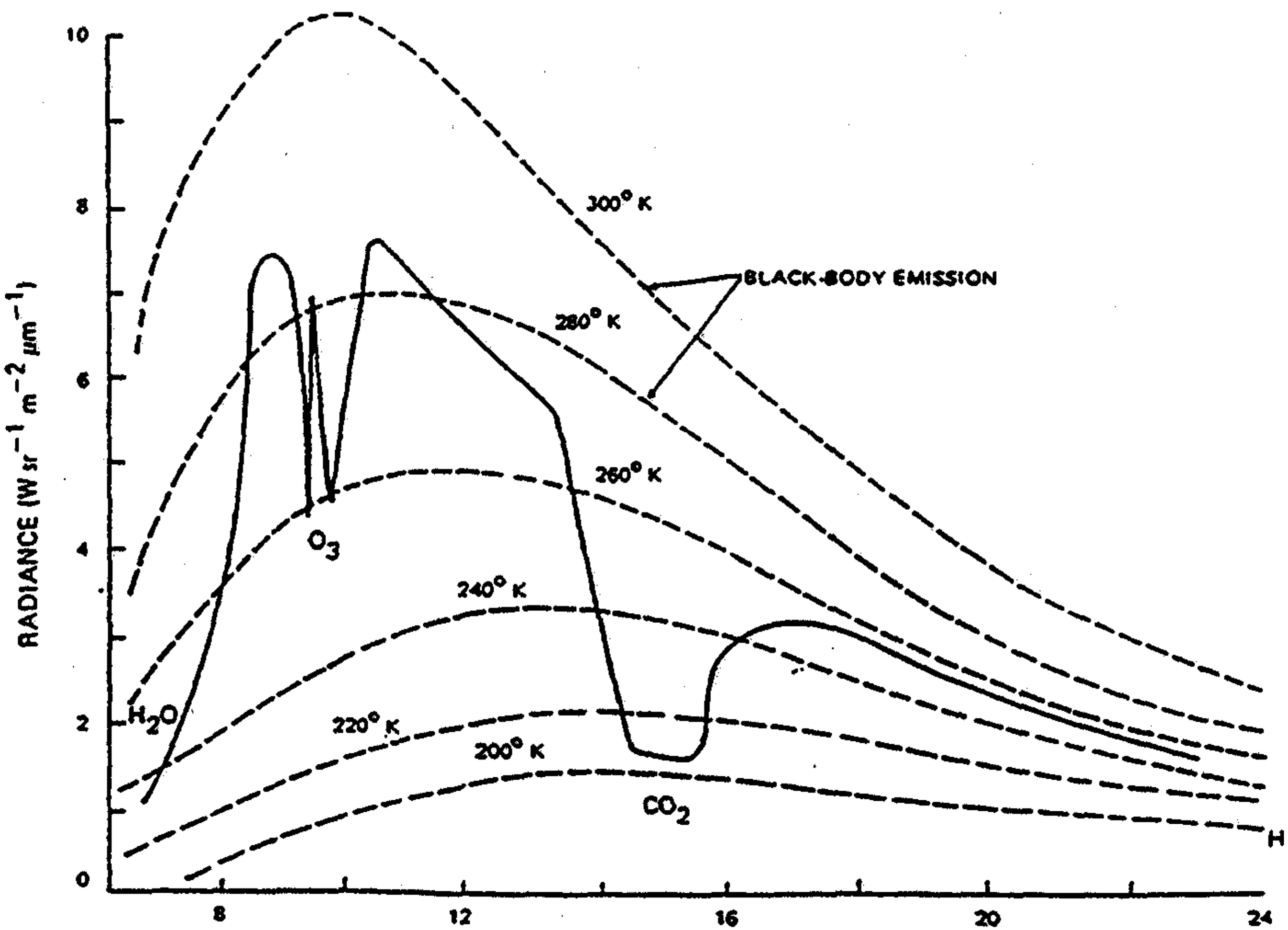
- Thus,

  $\alpha\, \phi\, \pi\, R^2 = 4\, \pi\, R^2\, \varepsilon\, \sigma\, T^4$ => $\mathbf{T^4 = (\alpha/\varepsilon)\, .\, \phi\, /\, 4\sigma}$
  => <u>T = 255 K</u> (-18 °C)

- i.e. Earth should be a frozen planet!

Spacecraft Thermal Design

- Earth Thermal Emission:

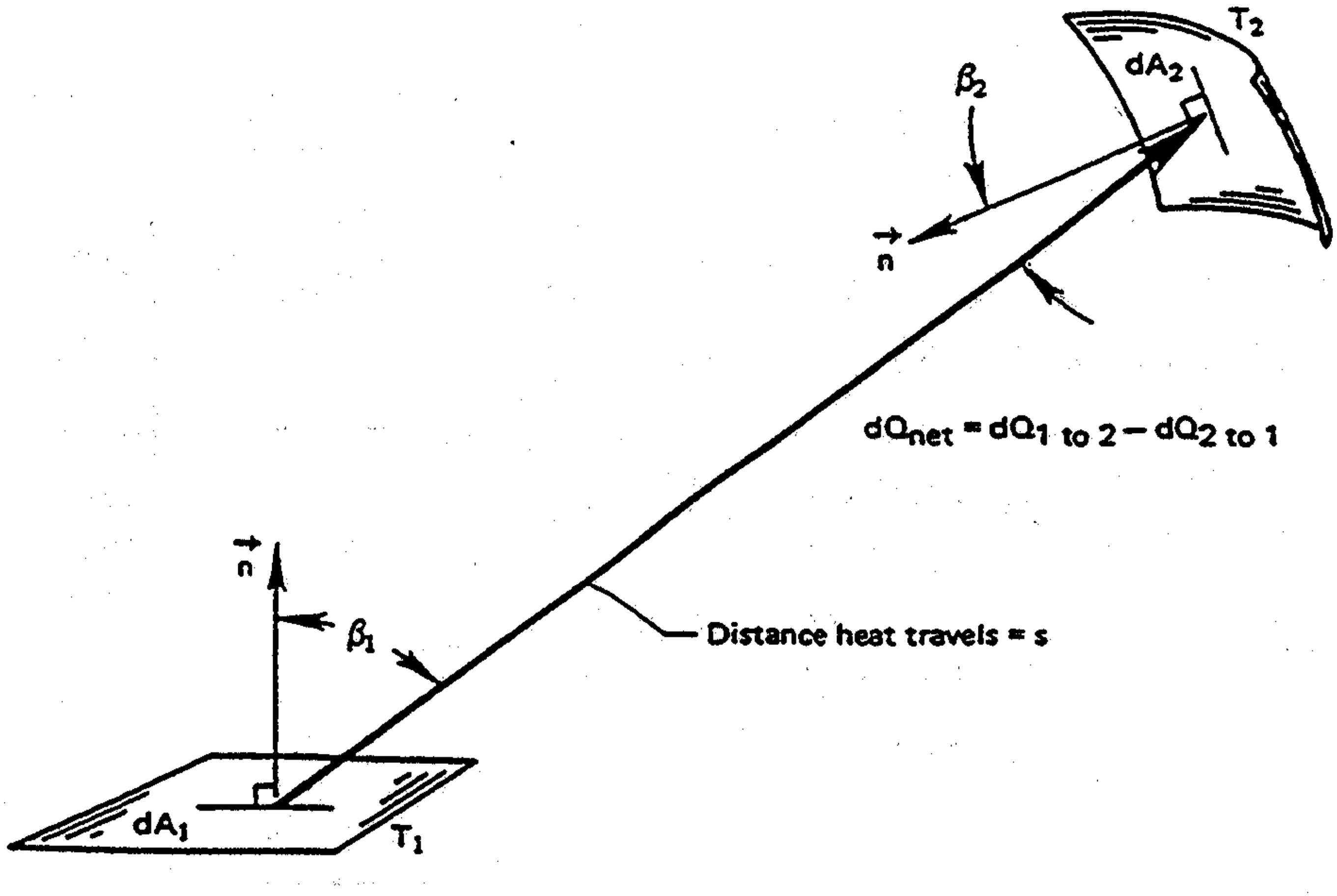


Spacecraft Thermal Design

- Can we use this same method to predict the temperature of a spherical spacecraft -e.g. *"Sputnik-1"*?
- No - there are other sources of heat - both internal and from the Earth (i.e. **albedo radiation** and Earth's own **thermal emissions**).
- Problem: For an Earth-orbiting satellite - the Earth is not a "point source" of heat - simple projection is not appropriate - hence the concept of a **view factor**:
- A view factor represents *the fraction* of *the radiative energy leaving one surface that strikes another surface directly.*

Spacecraft Thermal Design

- View Factor Geometry:



Spacecraft Thermal Design

- The view factor from surface 1 to surface 2 is given by:

$$F_{1\text{-}2} = \frac{1}{A_1} \int_{A_1} \int_{A_2} \frac{\cos\beta_1 \cos\beta_2}{\pi s^2} \, dA_1 dA_2$$

- The view factor from surface 2 to surface 1 is given by:

$$F_{2\text{-}1} = \frac{1}{A_2} \int_{A_2} \int_{A_1} \frac{\cos\beta_2 \cos\beta_1}{\pi s^2} \, dA_2 dA$$

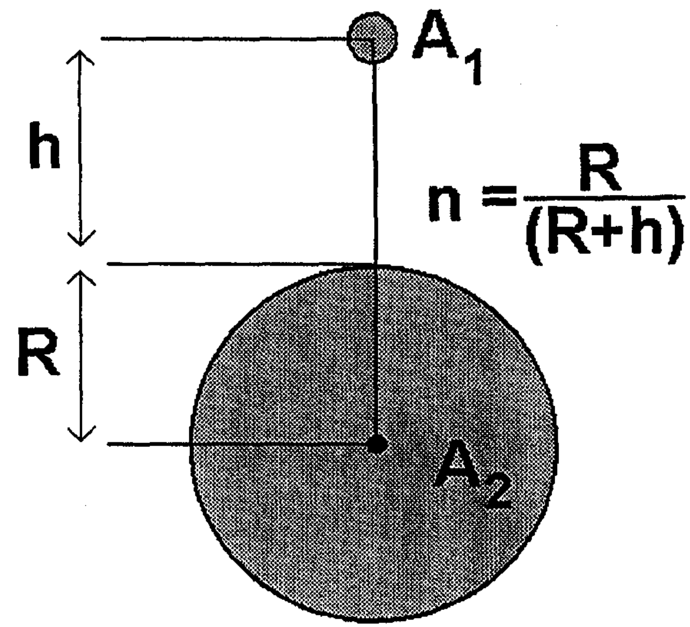
- Thus, by symmetry we can write the **reciprocity relation**:

$$\mathbf{A_1 F_{1\text{-}2} = A_2 F_{2\text{-}1}}$$

Spacecraft Thermal Design

- View factors are difficult to calculate except for the simplest geometries – and these are tabulated in any heat transfer textbook.
- For example, the view factor from a spherical surface element, A_1, of radius *r*, to a sphere of radius *R* and of surface area A_2, where the element is a distance *h* above the sphere, is given by:

$$F_{1\text{-}2} = \frac{1}{2}\left[1 - \sqrt{(1 - n^2)}\right]$$



Spacecraft Thermal Design

- We can now use these results to find the heat power which impinges upon a spherical satellite (e.g. *"Sputnik-1")* due to **thermal emission (IR)** from the Earth:

  (Satellite height, h = 227 km; Radius of Earth, R = 6378 km)

- Step 1 : *Find the heat power at the Earth's "surface":*

Taking the Earth's blackbody temperature to be 255 K, we can use **Stefan-Boltzmann** to get the flux emitted at the Earth's surface, ϕ_{Earth}:

$\phi_{Earth} = E_b = \sigma T^4 = 5.67 \times 10^{-8} \times (255)^4 = \underline{240}\ W.m^{-2}$

Taking the Earth's surface area to be A_2, then the total power emitted is $\underline{240\ A_2}$ watts.

Spacecraft Thermal Design

- Step 2 : *Find the fraction of this power which hits the satellite:*

  The fraction required is simply the **view-factor** F_{2-1}.

  But, we are given $F_{1-2} = (1/2)\,[\,1 - (1 - n^2)^{1/2}]$, where

  $n = R / (R+h) = 6378 \text{ km} / (6378 \text{ km} + 227 \text{ km}) = \underline{0.9656}$

  So we must use **reciprocity**:

  $A_1 F_{1-2} = A_2 F_{2-1}$ $\Rightarrow F_{2-1} = (A_1/A_2)\, F_{1-2}$

- Step 3: *Combine this fraction with Earth's surface heat power to get the heat power at the satellite:*

  Heat power at the satellite $= (240\, A_2) \,.\, (A_1/A_2)\, F_{1-2}$
  $= 240\, A_1 \times 0.37 = \underline{89\, A_1}$ watts (A_1 = surface area of satellite)

Spacecraft Thermal Design

- The equation of state for the **thermal balance** of a satellite is:
  Heat power absorbed = Heat power radiated + (mc) dT/dt
  where, the heat power absorbed (Q_{in}) comprises:

	$Q_{insolation}$	$= \alpha_s \, \phi_{solar} \, A_{projected}$
+	Q_{albedo}	$= f\,(\alpha_s, \phi_{solar}, A_{surface},$ solar direction, *etc*.)
+	$Q_{Earth\text{-}IR}$	$= \varepsilon_{IR} \, \phi_{Earth} \, F_{satellite\text{-}Earth} \, A_{surface} \; [\alpha_{IR} = \varepsilon_{IR}]$
+	$Q_{intemal}$	= internal power dissipation

The heat power radiated (Q_{out}) = $\varepsilon_{IR} \, A_{surface} \, \sigma \, T^4$
"(mc)" is the **heat capacity** (mass x specific heat capacity) of the satellite.

At **thermal equilibrium** (max. or min. temperature),
dT/dt = 0

Spacecraft Thermal Design

- Finding the maximum temperature of *Sputnik-1:*
  - ***Data*** - *Sputnik-1* was a 29 cm radius polished aluminium sphere (α_s=0.2; ε_{IR}=0.05) with a minimum orbital altitude of 227 km. Let $Q_{albedo} = Q_{Earth\text{-}IR}$, $Q_{internal}$ = 5 W and ϕ_{solar} = 1353 Wm^{-2}.
  - ***Solution:***

    $Q_{insolation} = \alpha_s \phi_{solar} A_{projected} = 0.2 \times 1353 \times \pi(0.29)^2$
    = <u>71.5 W</u>

    $Q_{Earth\text{-}IR} = \varepsilon_{IR} \phi_{Earth} F_{sat\text{-}Earth} A_{surface} = 0.05 \times 89 \times 4\pi(0.29)^2$
    = <u>4.7 W</u>

    Q_{in} = 71.5 + 4.7 + 4.7 + 5 = <u>85.9 W</u>

    $Q_{out} = \varepsilon_{IR} A_{surface} \sigma T^4 = 0.05 \times 4\pi(0.29)^2 \times 5.67 \times 10^{-8} \times T^4$

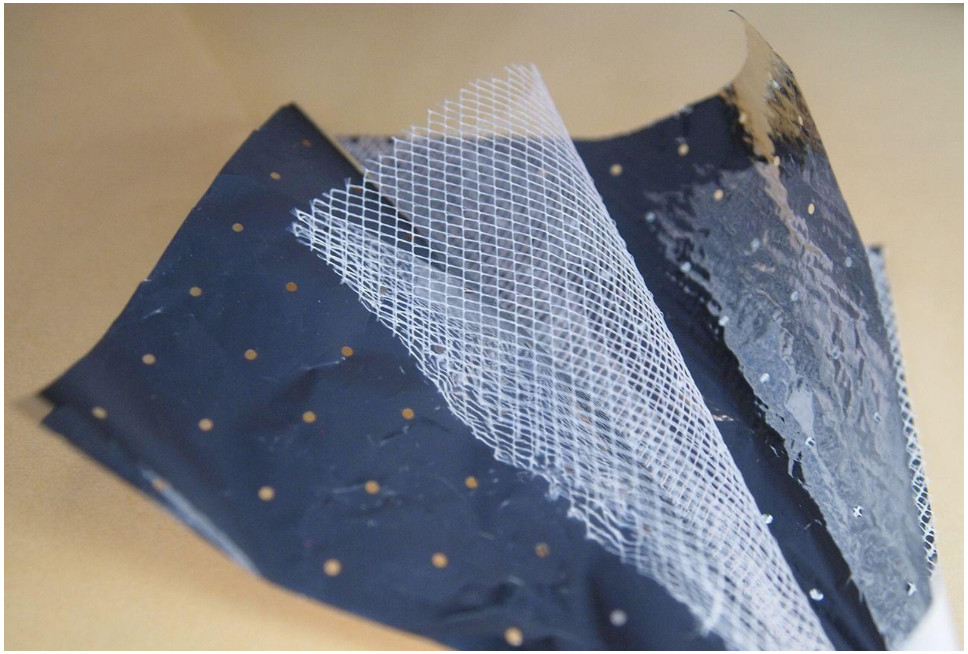
    $\Rightarrow T^4 = 85.9/3 \times 10^{-9}$

    (as $dT/dt = 0$ at $T_{max} \Rightarrow Q_{in} = Q_{out}$)

    => T = <u>411 K</u> (138 °C)

Passive thermal control
Εξωτερικός θερμικός έλεγχος – Παθητικός Έλεγχος

MLI – Multi Layer Insulation

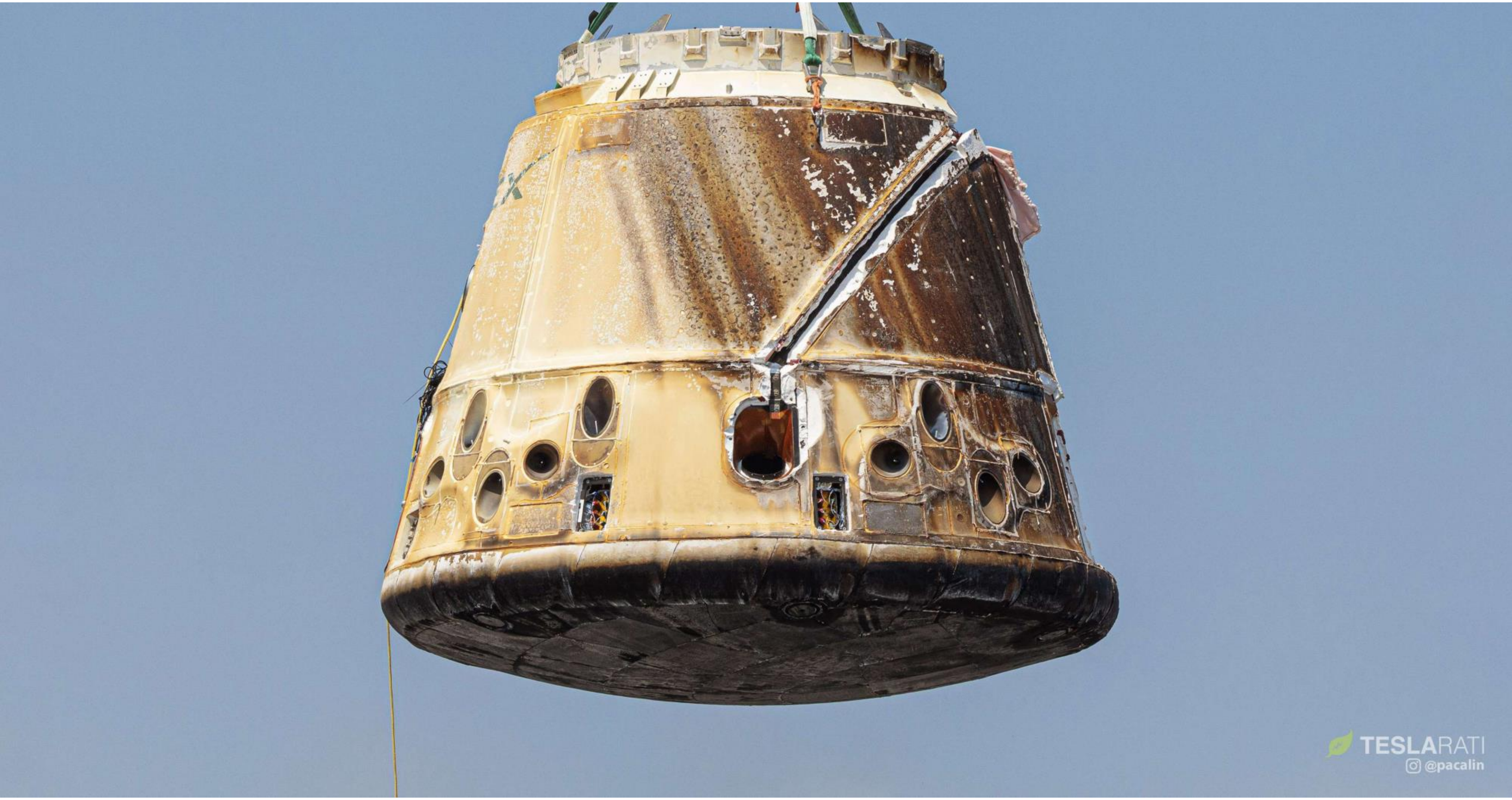


ΠΑΝΕΠΙΣΤΗΜΙΟ ΠΑΤΡΩΝ UNIVERSITY OF PATRAS

Θερμική Ασπίδα (Ablation Shield) - Apollo

Θερμική Ασπίδα (Ablation Shield) – Dragon SpaceX

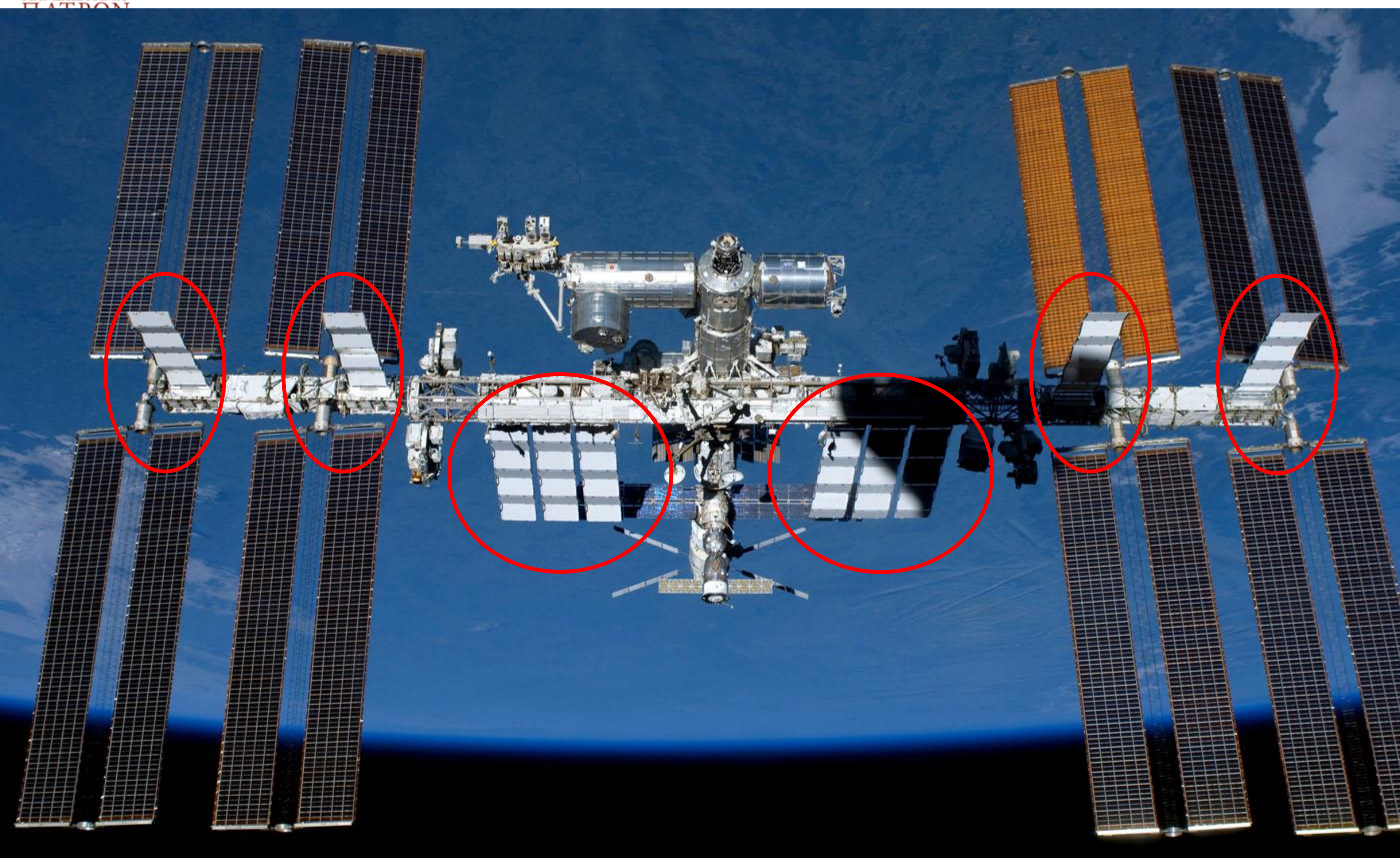
Θερμική Ασπίδα (Ablation Shield) – Dragon SpaceX



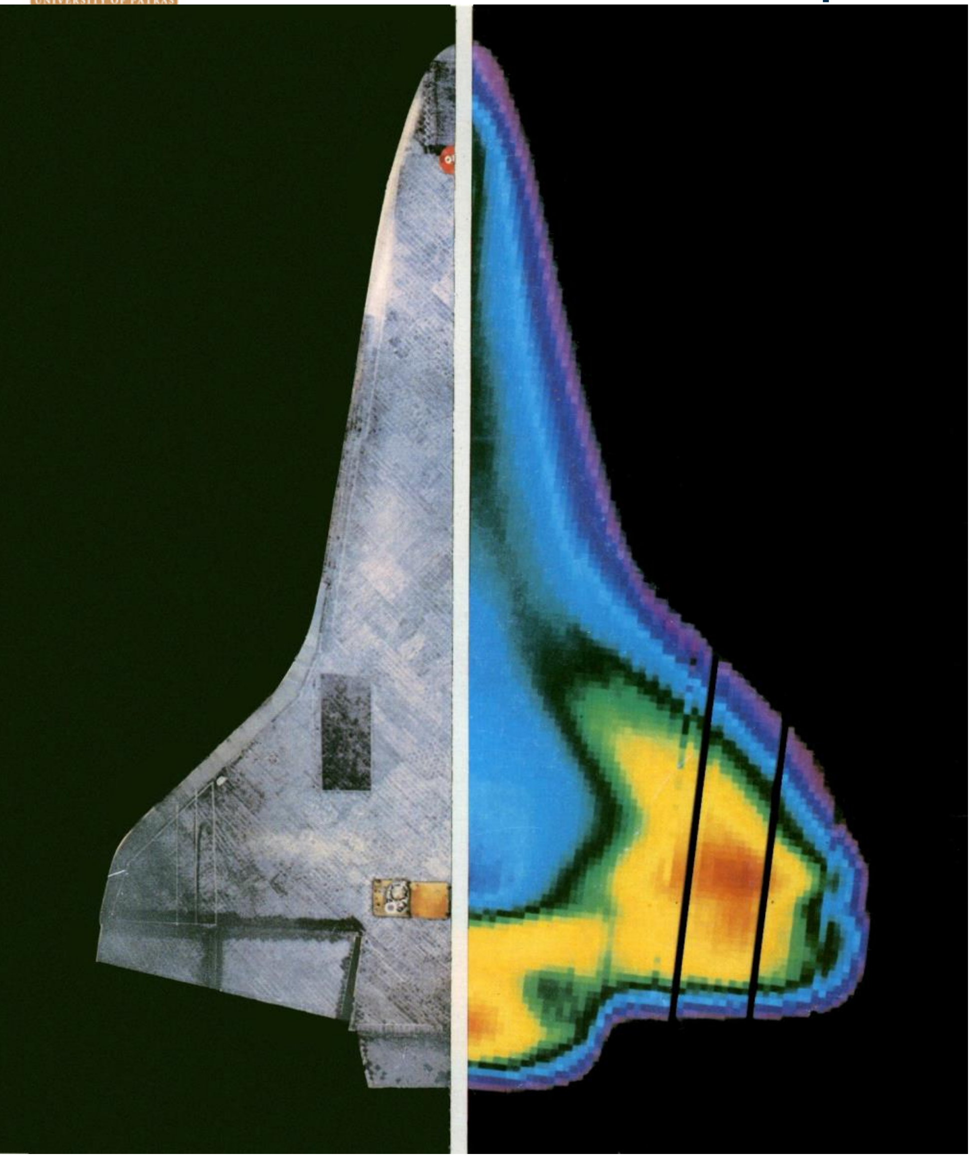
Θερμικά Πλακίδια (Thermal Tiles)

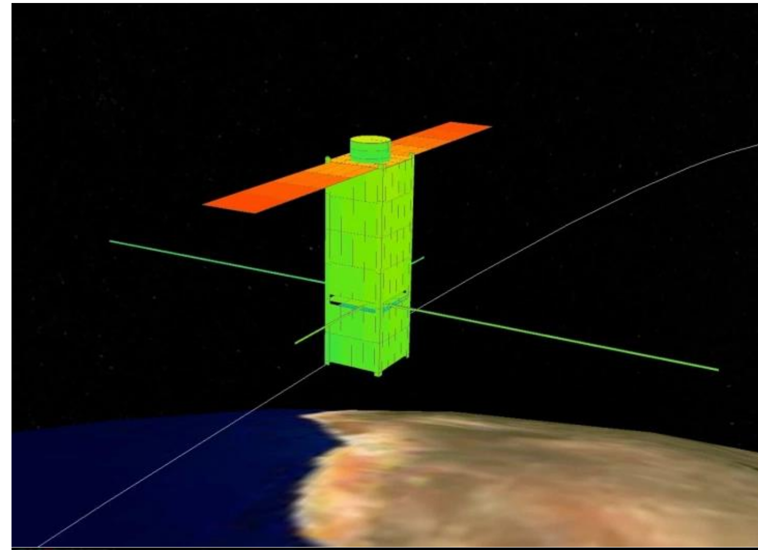
https://www.youtube.com/watch?v=Pp9Yax8UNoM

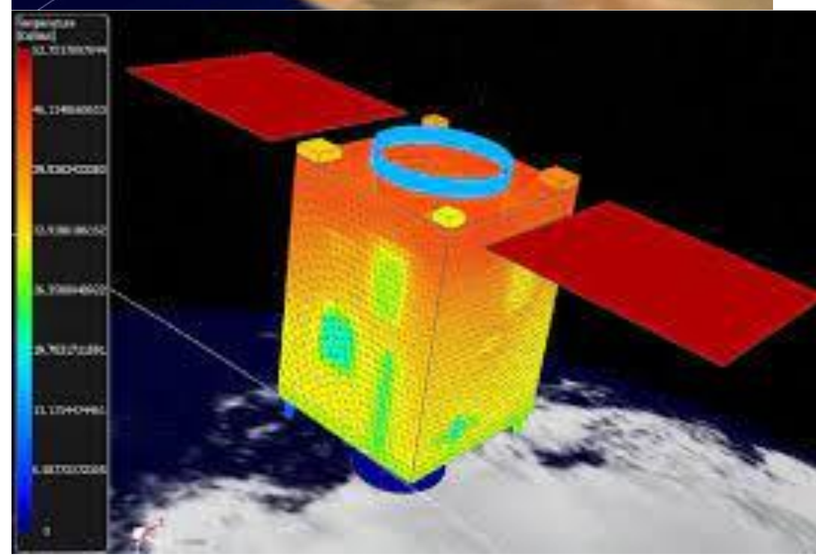
Ακτινοβολητές - Radiators



Θερμικός Σχεδιασμός



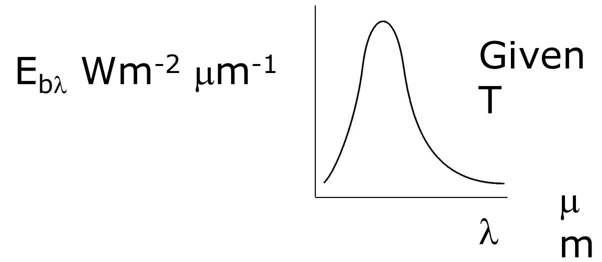




- https://www.esatan-tms.com/

Summary

- Blackbody Radiation:



- Stefan-Boltzmann Law:

$$E_b(T) = \int_{\lambda=0}^{\infty} E_{b,\lambda}(T)\, \delta\lambda$$

$$= \sigma T^4 \quad w.m^{-2}$$

σ = Stefan's Constant
= 5.67 x 10^{-8} $w.m^{-2}.k^{-4}$

- Real Bodies:

Total Emissivity $\quad \varepsilon \equiv \int_0^{\infty} \frac{E_\lambda(\lambda,T)\,\delta\lambda}{\sigma T^4}$

- Gray body:

ε_λ independent of λ