ENAE 691 SATELLITE DESIGN
THERMAL CONTROL

University of Maryland, College Park
Department of Aerospace Engineering
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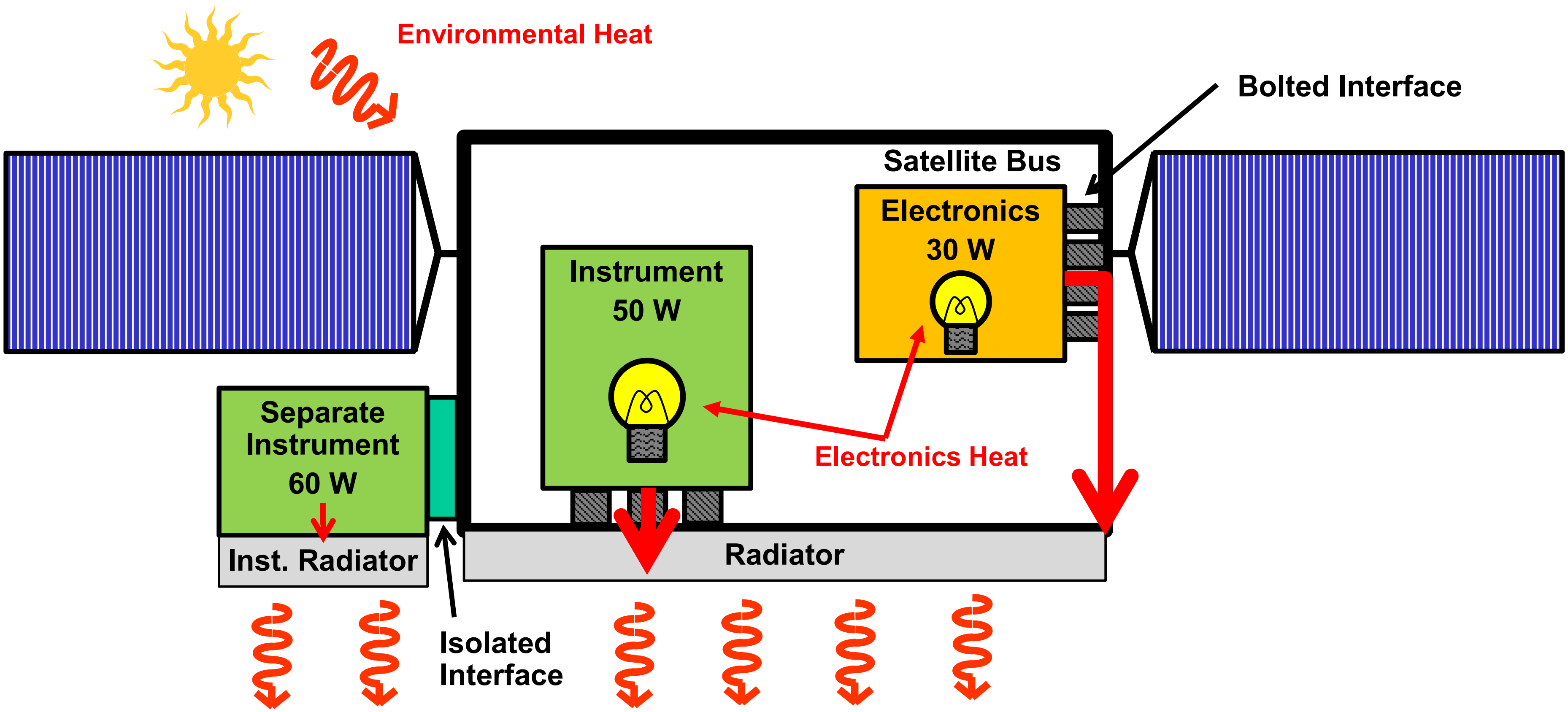
Course Outline

- Heat Transfer Review
- Thermal Design Process
  - Requirements and Constraints
  - Thermal Environment
  - Thermal Challenges
  - Thermal Hardware
  - Radiator and heater sizing
  - Thermal Testing
  - Mass and Power
  - Document and Iterate
- Recap and Course Project Thermal Objectives

What is "Thermal Design" ?

- Thermal design uses the fundamental principles of heat transfer and thermodynamics to manage the heat flow into, within, and from the satellite.



- The thermal engineer uses thermal control to balance the energy emitted (as IR radiation) against the energy generated internally (from onboard electronics) and the energy absorbed from the environment ➔This maintains all parts of a satellite within their allowable temperature ranges throughout all phases of the mission

REVIEW OF HEAT TRANSFER

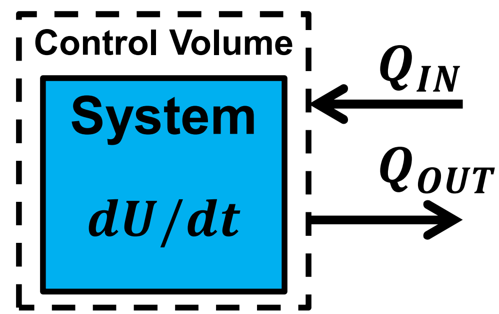
Heat Transfer Background

- **First law of thermodynamics: Energy in a confined system is conserved**

$$\underbrace{Q}_{\text{Heat}} - \underbrace{W}_{\text{Work}} = \underbrace{dU/dt}_{\text{Change in internal energy}}$$

- **For a satellite in orbit, [typically] no work is done by the system, so the net heat into the system equals the change in internal energy (temperature):**

$$Q_{NET} = dU/dt$$



- **Net heat into the system is:**

$$Q_{NET} = \Sigma Q_{IN} - \Sigma Q_{OUT}$$

Primary source of "$\sum Q_{IN}$", from external environmental heating

3 basic kinds of heat transfer apply to "$\sum Q_{OUT}$" in the thermal design of satellites

Three Forms of Heat Transfer

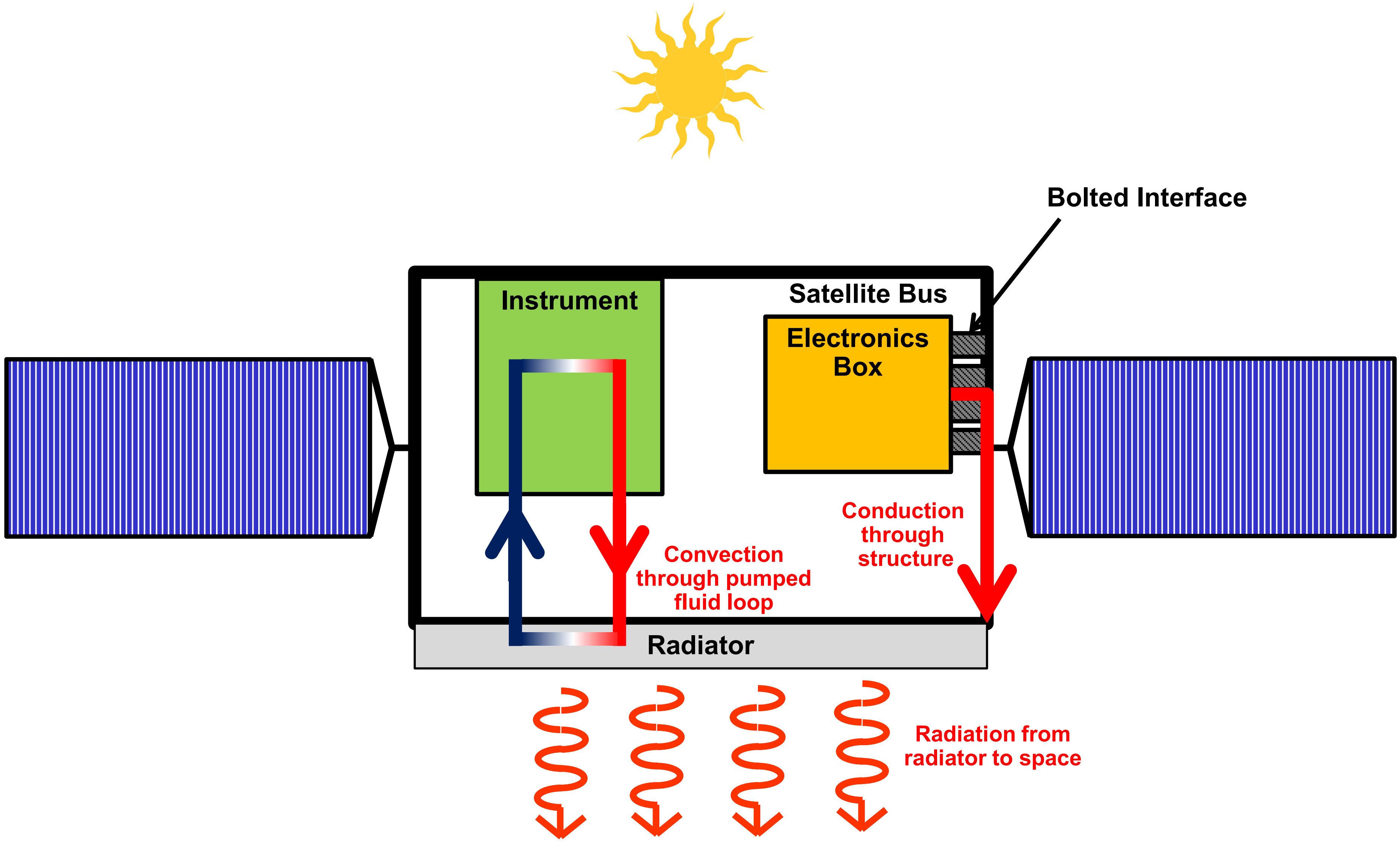
- **Conduction: "*heat transfer* by direct contact of particles of matter"**
  - **This is the fundamental method for getting the heat out of the various electronics boxes, through the S/C to the exterior radiator panels**
- **Convection: "*heat transfer* by movement of a fluid/gas (unlike conduction, fluid/gas currents are additionally involved in convection)"**
  - **Free: (natural) based on differences in fluid/gas density (temperature) in a gravitational or other force field**
  - **Forced: pump, fan, blower acting on fluid/gas**
- **Radiation: "*heat transfer* by the emittance of infrared radiation (IR)"**
  - **This occurs in both air and vacuum**
  - **This is the fundamental method for getting rid of a satellite's waste heat**

Fried eggs are heated by conduction from the pan

Blowing across a hot cup of coffee cools it by convection

Curly fries (and other fast "food") are heated by radiation from a lamp

Three Forms of Heat Transfer on a Satellite



Basics - Conduction Heat Transfer

- This is the primary method of transferring heat from the sources within the spacecraft to the external surfaces where it is radiated into space

- One dimensional conductive heat flow through a material is defined by the equation:

$$Q(T) = \left[k(T) \frac{A}{\Delta x} \right] \Delta T$$

Where:

$Q(T)$	heat flow (function of temperature)
$k(T)$	thermal conductivity of material (function of temperature)
x	coordinate of length
A	cross-sectional area

Note: For most spacecraft, the material conductivity used in analysis is constant since they are near-ambient. However, for temperature excursions ± 50°C from ambient, temperature dependency should be used.

Conduction Heat Transfer – Material Properties

- All materials are conductive, but some have much better conductivity than others
  - Metals vs. Non-metals
- Alloy differences can also affect thermal conductivity
- The properties typically used for thermal analysis are: Density, ρ; Thermal Conductivity, k; Specific Heat, C_P
- Low conductivity materials used as isolators, while higher conductivity materials may be used as spreaders or heat straps

Solid Material Thermal Properties

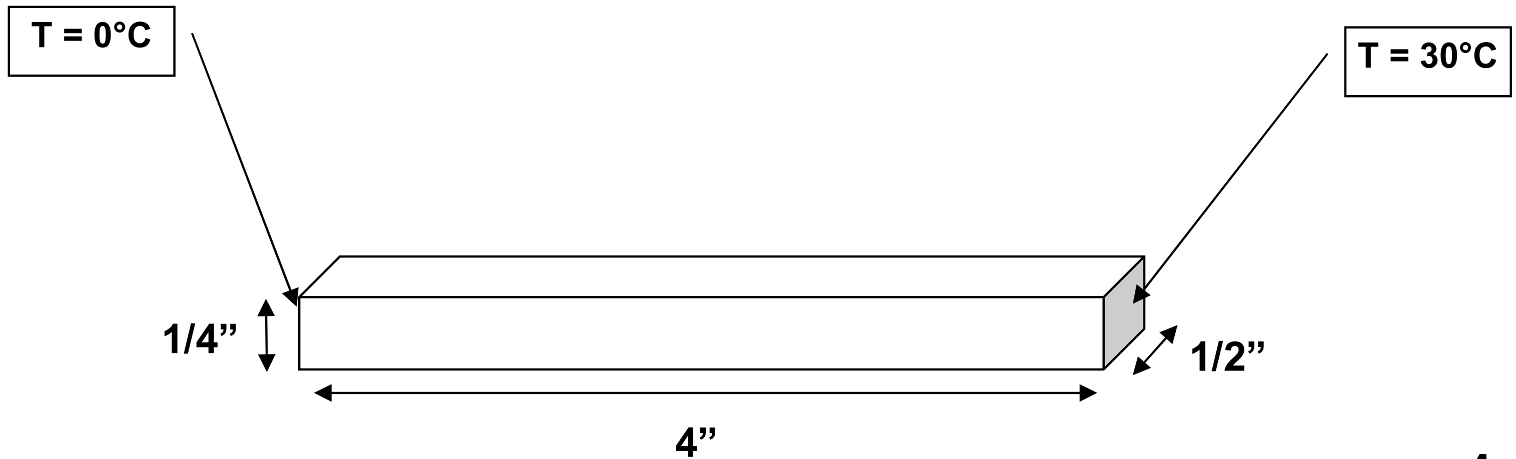
Material	ρ (kg/cm^3)	k (W/cm°C)	c_p (W-hr/kg°C)
Aluminum			
208	0.00277	1.212	
222	0.00277	1.333	
242	0.00277	1.506	
295	0.00277	1.437	
B295.0	0.00277	1.610	
308	0.00277	1.419	
319	0.00277	1.142	
355	0.00277	1.506	
C355.0	0.00277	1.419	
356	0.00277	1.593	
A356.	0.00277	1.593	
A380.	0.00277	1.004	
A413.0	0.00277	1.212	
443	0.00277	1.454	

Source: Spacecraft Thermal Control Handbook, Vol I, Appendix B

Material	Thermal conductivity (W/m K)*
Diamond	1000
Silver	406
Copper	385
Gold	314
Brass	109
Aluminum	205
Iron	79.5
Steel	50.2
Lead	34.7
Mercury	8.3
Ice	1.6
Glass,ordinary	0.8
Concrete	0.8
Water at 20° C	0.6
Asbestos	0.08
Fiberglass	0.04
Brick,insulating	0.15
Brick, red	0.6
Cork board	0.04
Wool felt	0.04
Rock wool	0.04
Polystyrene (styrofoam)	0.033
Polyurethane	0.02
Wood	0.12-0.04
Air at 0° C	0.024
Helium (20°C)	0.138
Hydrogen(20°C)	0.172
Nitrogen(20°C)	0.0234
Oxygen(20°C)	0.0238
Silica aerogel	0.003

Conduction Heat Transfer – Sample Problem

- **How much heat flows through a perfectly insulated 6061-T6 aluminum bar (1/2” x 1/8” x 4” long) if one end is at 30°C and the other at 0°C ?**



$$Q(T) = \left[k(T)\frac{A}{\Delta x}\right]\Delta T$$

Thermal conductivity, k: 3.1 W/in-°C

Solution:

$$Q = [k * A / \Delta X] * \Delta T$$

$$Q = [3.1\mathrm{W/in°C} * (0.25 * 0.5)\mathrm{in}^2/4\mathrm{in}] * (30°\mathrm{C} - 0°\mathrm{C})$$

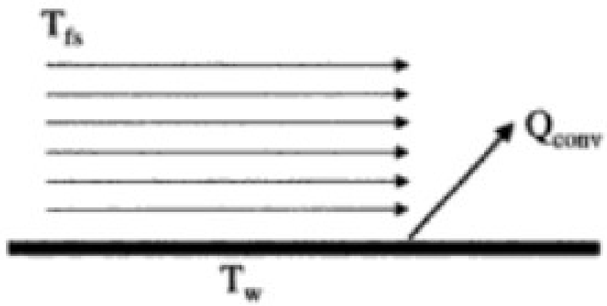
$$Q = [(3.1 * 0.125 / 4) * 30]\mathrm{W}$$

$$Q = 2.9\mathrm{W}$$

Basics - Convection Heat Transfer

- **One dimensional convective heat flow from a surface to a fluid is defined by the equation:**

$$Q(T) = A[h(T)]\Delta T$$



Where:

Q(T)	**heat flow (function of temperature)**
h(T)	**convection coefficient (function of temperature)**
A	**cross-sectional area**

Note: Most of the thermal issues requiring convective heat transfer occur during testing or during assembly, e.g. – acoustics or EMI where fans aren't used and the "sealed" up spacecraft can be operated for hours.

- **This is not typically a consideration for thermal design of satellites, except for ground cooling, onboard pumped fluid loops, or missions to planets with atmospheres**

Convection Heat Transfer – Sample Problem

- **Can calculate convective heat transfer coefficients, etc.**
  - **Use simple calculations, or full CFD analysis to determine velocities, pressures, etc.**

- **But usually, only simple calculations suffice:**
  - **Consider a vertical 3 m^2 radiator panel with 200W of heat generated by the electronics mounted on the inside, that is sealed up so the heat must be transferred through the radiator panel to the external air. If the room is ambient temperature (20°C), and no forced air is available, how warm will the outer surface be? (Note: typical free convection rate is 5 W/m^2°C)**

$$\boldsymbol{Q}(\boldsymbol{T}) = [\boldsymbol{h}(\boldsymbol{T}) * \boldsymbol{A}]\Delta \boldsymbol{T}$$

Solution:

$$Q = [h * A] * \Delta T$$

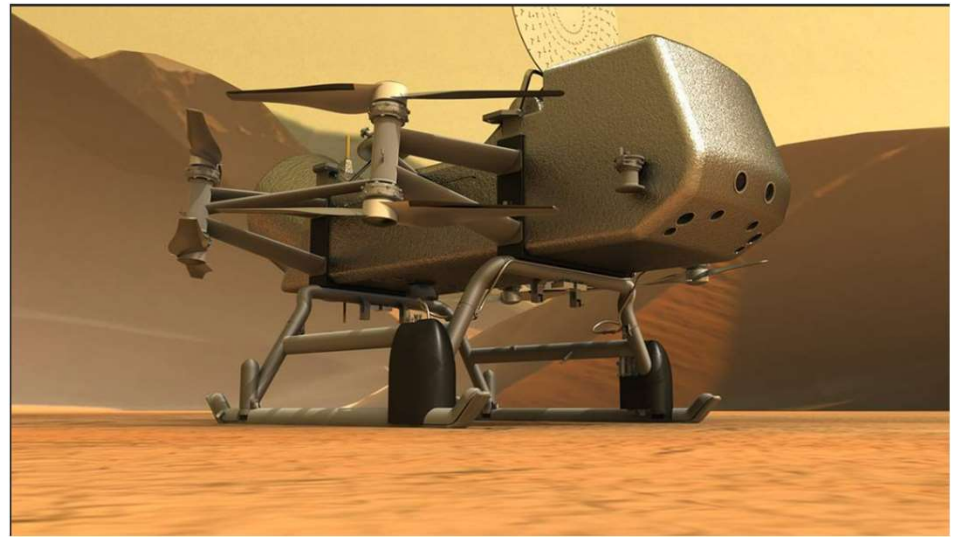
$$200\mathrm{W} = [5.0\mathrm{W/m^2\,°C} * (3.0)\mathrm{m^2}] * (T_{\mathrm{PANEL}} - 20°\mathrm{C})$$

$$T_{\mathrm{PANEL}} = 20°\mathrm{C} + [200\mathrm{W}/(5.0\mathrm{W/m^2\,°C} * (3.0)\mathrm{m^2})]$$

$$T_{\mathrm{PANEL}} = 33°\mathrm{C}$$

Convection in Spacecraft: NASA's Dragonfly Mission

- **Dragonfly Mission Overview:**
  - 8-bladed rotorcraft to visit Saturn's moon, Titan.
  - Launch planned for 2027 and arrival in 2034
  - Titan's surface temperature is around -179 Celsius.
  - Titan has a nitrogen based atmosphere and the surface pressure is also 50 percent higher than Earth's.



Artist's Impression of Dragonfly on Titan's surface **www.nasa.gov/dragonfly**

- **Dragonfly's Thermal Design:**
  - Uses convection as it's main heat transfer mechanism
  - Detailed CFD (computational fluid dynamics) models predict temperatures instead of traditional spacecraft radiation/conduction modeling

Basics - Radiative Heat Transfer

- Thermal radiation is electromagnetic radiation that is emitted solely based on ***temperature*** and requires no intermediary medium. Heat transfer from a satellite, and within, is due to thermal radiation.
- Governed by Planck's Law; when electromagnetic radiation is integrated over all wavelengths, it gives:

$$Q_{blackbody} = \sigma \varepsilon A T^4 = \sigma A T^4$$

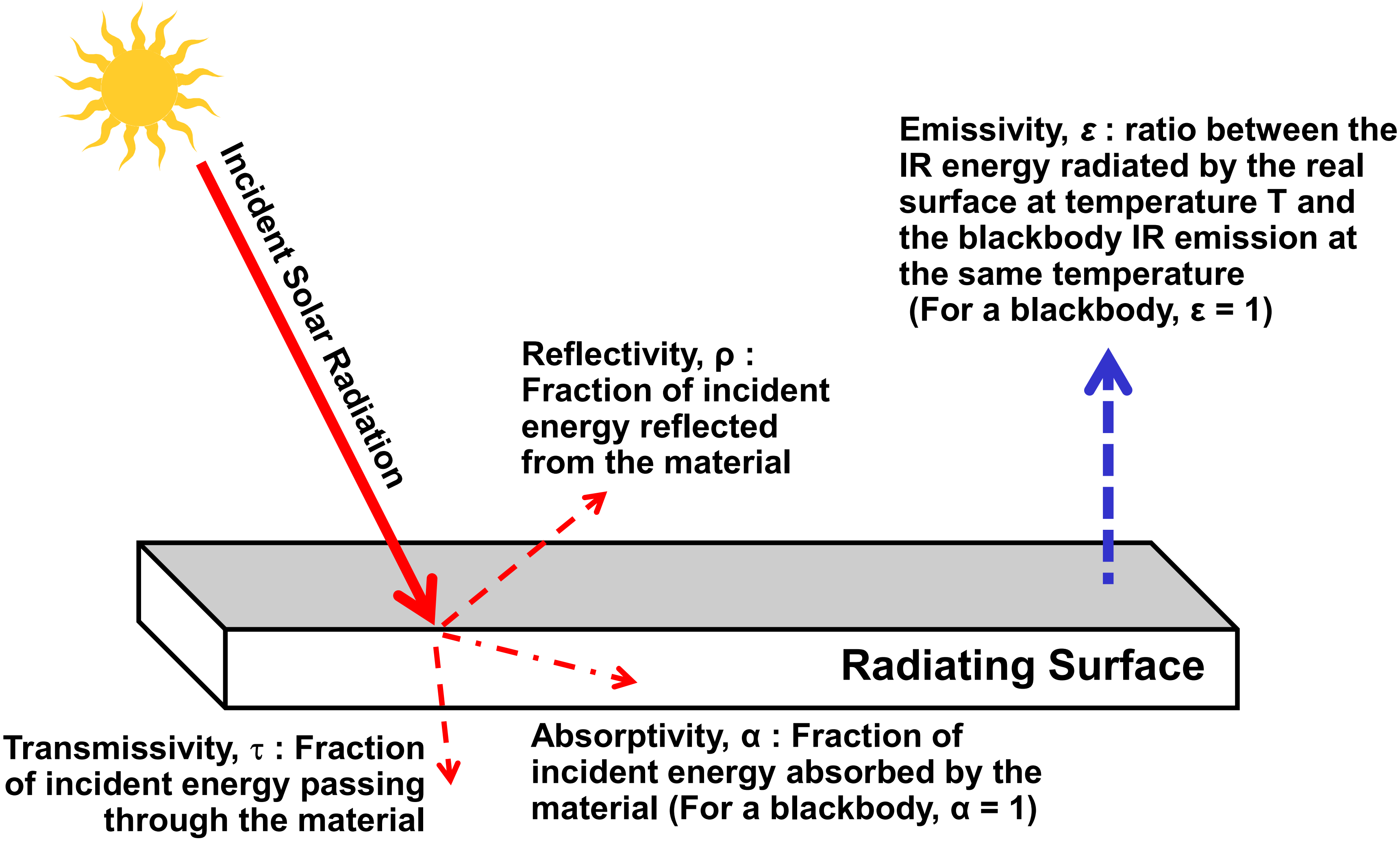
for one dimensional radiative heat flow from a blackbody surface.

Where:

Q	emissive power from a surface, or from one surface to space
σ	Stefan-Boltzmann constant, 5.67×10^{-8} W/(m^{2}K^4)
A	area of surface radiating
ε	Emissivity [blackbody = 1.0; real world "gray" surfaces <1.0]
T	temperature of blackbody surface

- A blackbody is an ideal body absorbing all the incident radiation, and a perfect, diffuse emitter
  - Space is usually assumed to be a blackbody
- Black bodies form the standard by which real bodies can be compared

Real bodies



For a given surface: $\alpha + \rho + \tau = 1$

Radiative Heat Transfer – Gray Bodies

- **Emissive power of a "real world" material is:**

$$Q = \varepsilon Q_{blackbody} = \varepsilon(\sigma A T^4)$$

for one dimensional radiative heat flow from a gray body surface where ε is the emissivity of the surface and $0 < \varepsilon < 1.0$

Where:

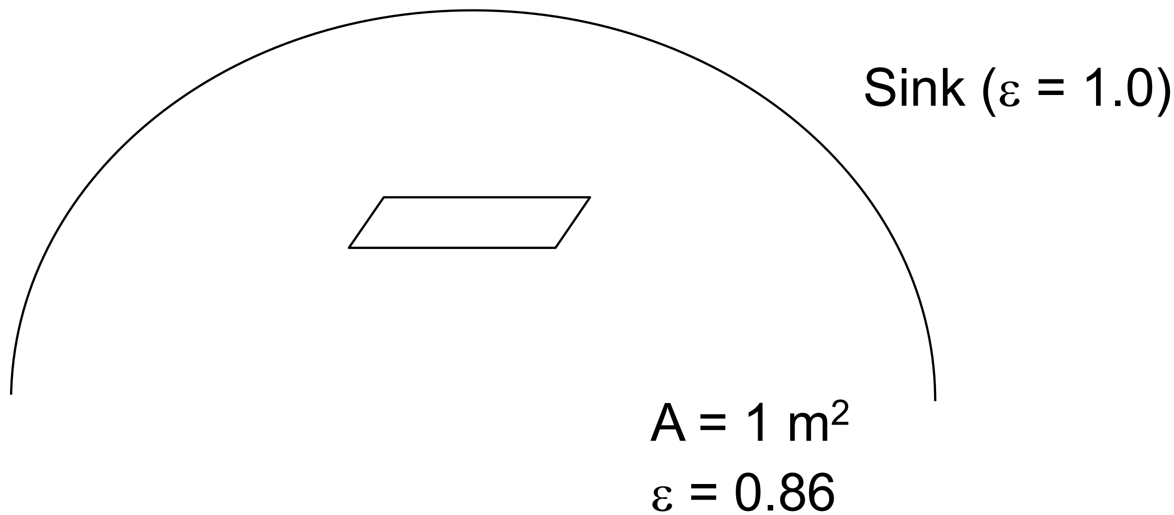
$Q(T)$	**emissive power from a surface, or from one surface to space**
σ	**Stefan-Boltzmann constant, 5.67×10^{-8} W/(m^{2}K^4)**
ε	**emissivity, or emittance, of the surface material ($\varepsilon < 1.0$)**
A	**area of surface radiating**

Note: For most spacecraft, the material emissivity is constant since they are near-ambient. However, for temperature excursions > 50K from ambient, temperature dependency should be used.

Radiative Heat Transfer - Example

- **How much heat can be radiated from a 1m² black plate (ε=0.86) at 30°C to a perfect sink at 0°C?**

Note: All radiation calculations must be completed on an absolute temperature scale! (Kelvin or Rankine)



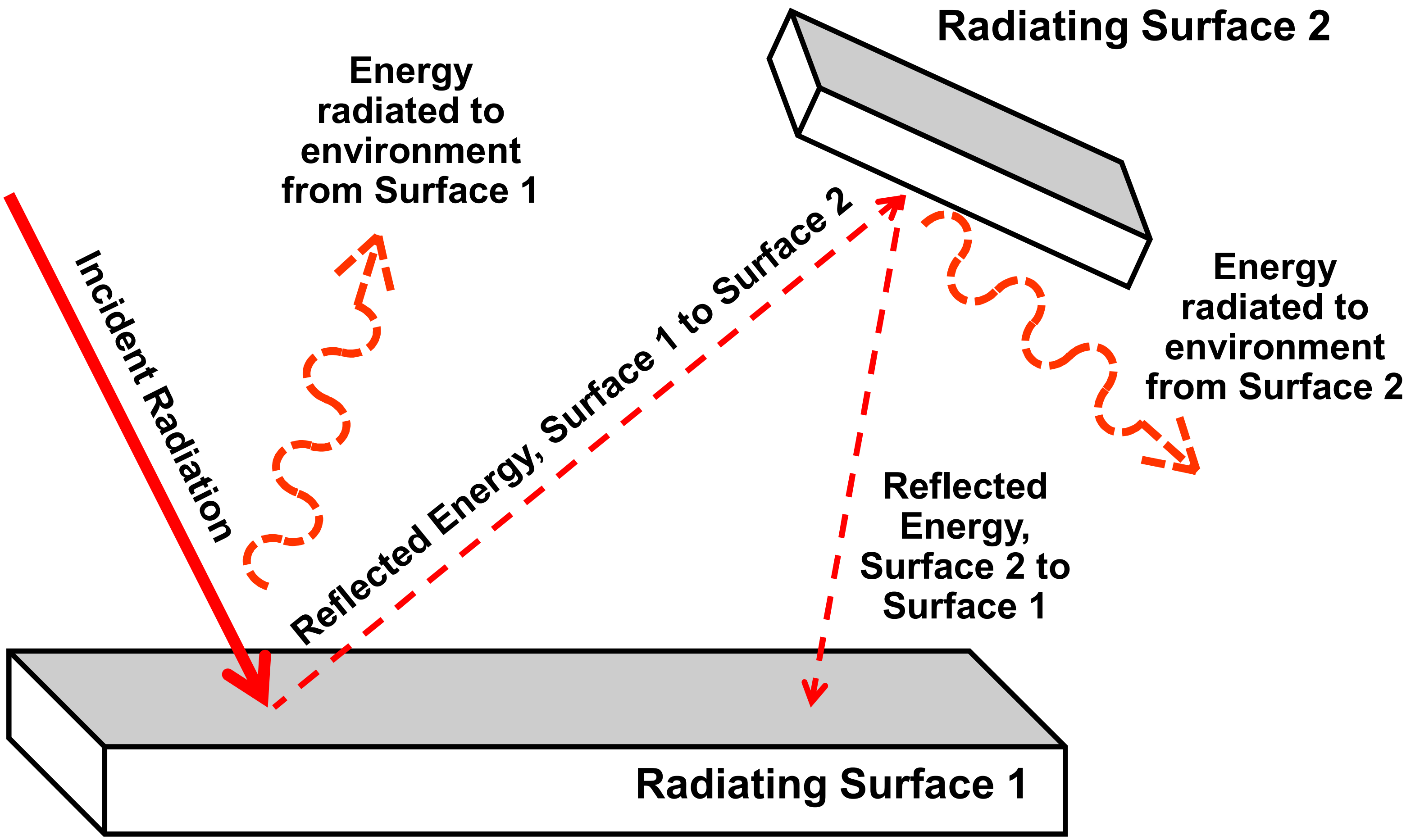
$$Q = \sigma \varepsilon A (T_{rad}^4 - T_{sink}^4)$$

Solution: $$Q = (5.67 x 10^{-8}\ W/m^2K^4) * 0.86 * 1\,m^2 * [(303K)^4 - (273K)^4]$$

$$Q = 140W$$

Radiative Heat Transfer - Example

- Now what about radiation between two or more surfaces?



- To determine the energy exchange in this system, we must first know the optical properties of the surfaces, and their views to each other

Radiation Heat Transfer - View Factor

- The radiation interchange between surfaces is determined by the geometry of the surfaces studied and their orientation with respect to each other
  - View factors between these surfaces must be determined (this is needed in both heat transfer and applied optics)
  - Although some materials have very directional or spectral properties of reflection, we'll only be dealing with diffuse surfaces
  - A diffuse configuration factor is the fraction of uniform diffuse energy leaving a surface that reaches another surface
- There are several methods for calculating these view factors:
  - Nusselt unit sphere
  - Hemicube
  - Direct integration
  - Ray tracing
- Some of these calculations have been tabulated in several references on heat transfer (e.g. Holman, 1986) or the NACA handbook
- These calculations are possible for simple geometries, but the typical satellite has many surfaces that interact "radiatively"
- Thermal software platforms are used to generate radiation views to space and between the many satellite surfaces. A typical satellite geometry model has 500-2000 surfaces with calculated radiation view factors numbering in the 10's or 100's of thousands
- Handy Fact: The view factor from any one side to any other in a cube has a closed form solution of 0.2

Radiative Heat Transfer – Summary

- **Using all these additional considerations from the basic "blackbody" emissive power equation, we get:**

$$Q_{1\text{-}2} = \sigma\,\varepsilon_1 A_1\,\mathcal{F}\,(T_1^{\,4} - T_2^{\,4})$$

for one dimensional radiative heat flow between <u>gray body</u> surfaces. (Note: 1 is always radiator, 2 is always sink)

Where:

$Q(T)$	emissive power from a surface, or from one surface to space
σ	Stefan-Boltzmann constant
ε	emissivity, or emittance, of the surface material
A	area of surface radiating
$\mathcal{F}$	"script F" or $\varepsilon_2 F_{1\text{-}2}$
$F_{1\text{-}2}$	View factor between surface 1 and surface 2

A good example of backloading is the energy from the warm solar arrays onto the side(s) of the spacecraft.

Note the Reciprocity Rule: $A_1F_{1\text{-}2} = A_2F_{2\text{-}1}$ $(F_{1\text{-}2} \neq F_{2\text{-}1}!)$

Summary of Equations

- **Conduction:** $Q(T) = \left[k(T)\frac{A}{\Delta x}\right]\Delta T$
- **Convection:** $Q(T) = [h(T) * A]\Delta T$
- **Radiation:** $Q_{1-2}(T) = \varepsilon\sigma_1 A_1 \mathcal{F}_{1-2}({T_1}^4 - {T_2}^4)$

View Factors: $A_1 F_{1-2} = A_2 F_{2-1}$ **(Reciprocity Rule)**

$\mathcal{F}_{1-2} = \varepsilon_2 F_{1-2}$

$\mathrm{VF}_{1-2} = F_{1-2} = \frac{Area}{Area_{HEMISPHERE}} = \frac{Area}{2\pi r^2}$

Radiative Energy Balance with the Sun:

$$A_P Q_S \alpha = A\varepsilon\sigma T^4$$

$$A_P = A\cos(\beta)$$

BASICS OF THERMAL DESIGN

Parameters Needed for Thermal Design

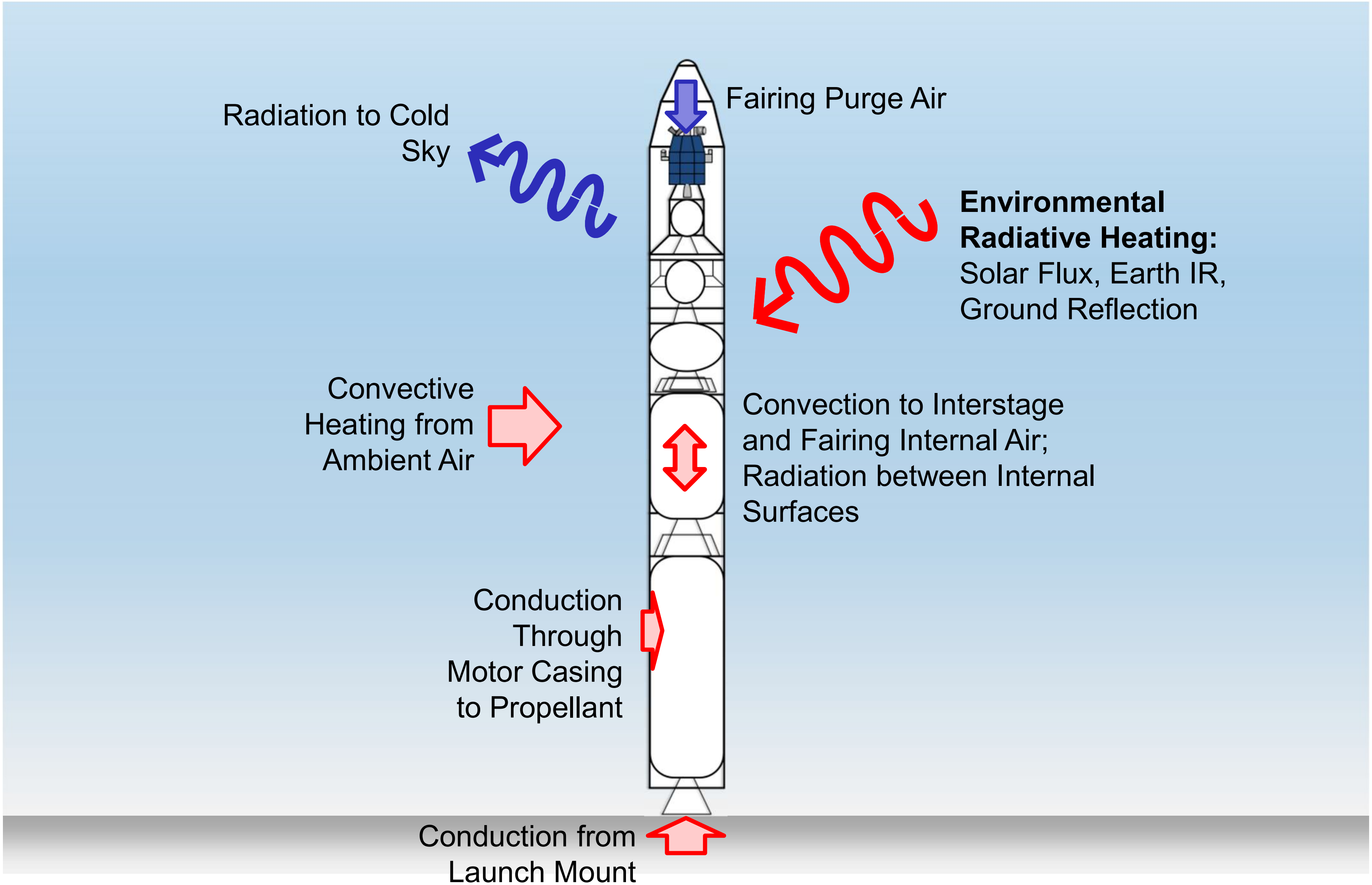
- **To establish a basic thermal design, we need to know:**
  - What are the sources of environmental heat around our spacecraft?
  - What is the spacecraft orbit (Altitude, Inclination, Equatorial crossing time at ascending node if LEO) and orientation (operational, safehold/survival, sun avoidance angle, pitch/roll/yaw angle, launch configurations)?
  - How much heat is being generated within the spacecraft (i.e. what are the power dissipations from all components)?
  - What are the mechanical interfaces for: Instruments, Propulsion, GN&C, Power, Structures, Communications, C&DH, and other subsystems?
  - Where are the components on the spacecraft and what path does the heat have to escape the spacecraft?
  - What are the mass/power/location allocations for thermal hardware?
  - What are the thermal requirements/temperature limits (operational and survival) of each component?
  - What thermal hardware do we have in our arsenal to control temperatures? What area do we have to place it on?
  - How do we verify that our thermal design operates as intended?

Thermal Design Considerations

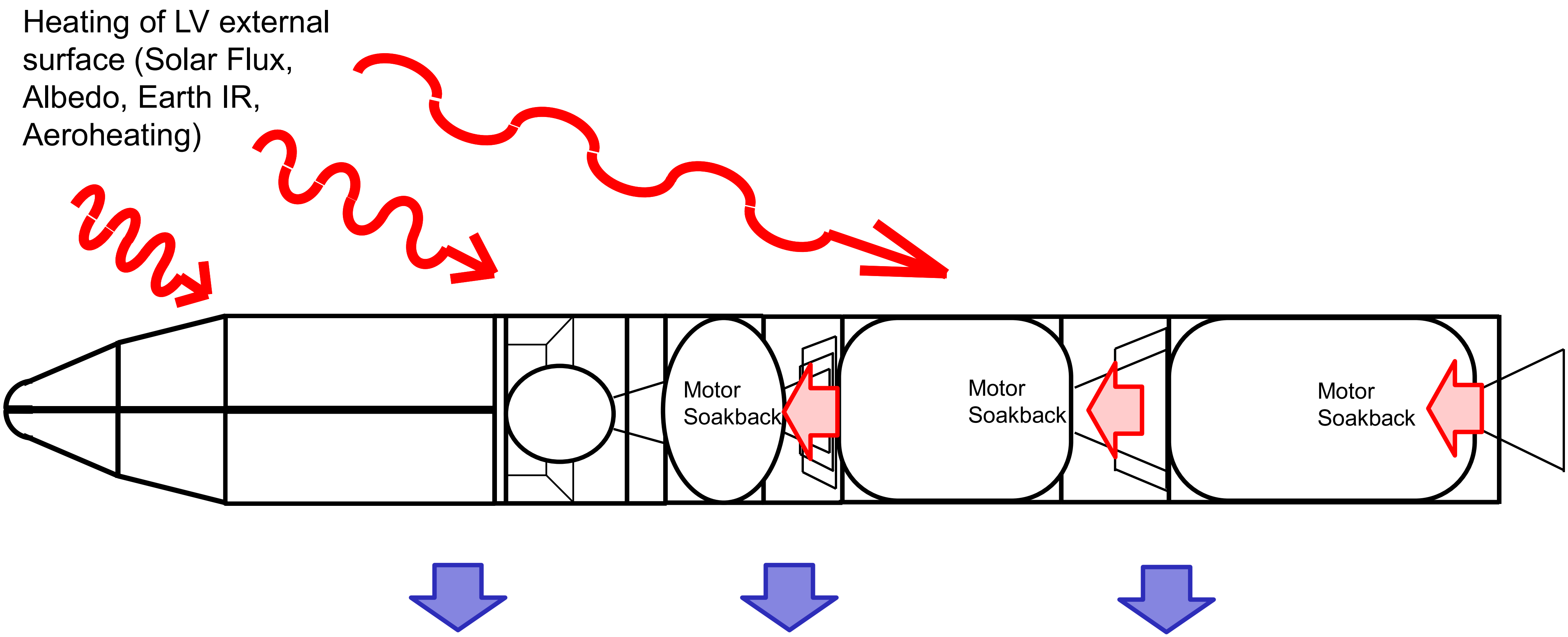
- ***The thermal engineer is responsible for all phases of the mission***
  - Thermal analysis requires consideration of all worst hot and cold cases the spacecraft will encounter during the mission lifetime
- **Thermal design should focus on *reliability* and *simplicity (when feasible)***
  - Bound and scope the problem with worst-case thermal scenarios
  - Leave **significant** margin in design for growth potential
  - Conduct a basic, simple thermal analysis to verify that the **basic design concept** works before commitment to execution (i.e. does the thermal design make sense?)
    - When a concept is finalized, **then** detailed analytical models can be generated to predict temperatures and heat flows
  - **It is critical to iterate with other subsystems** throughout the development phases to get the best system, keeping compromise and testing in mind

An overview of mission phases and associated thermal loads are provided in the following slides

Mission Phases: Ground Operations

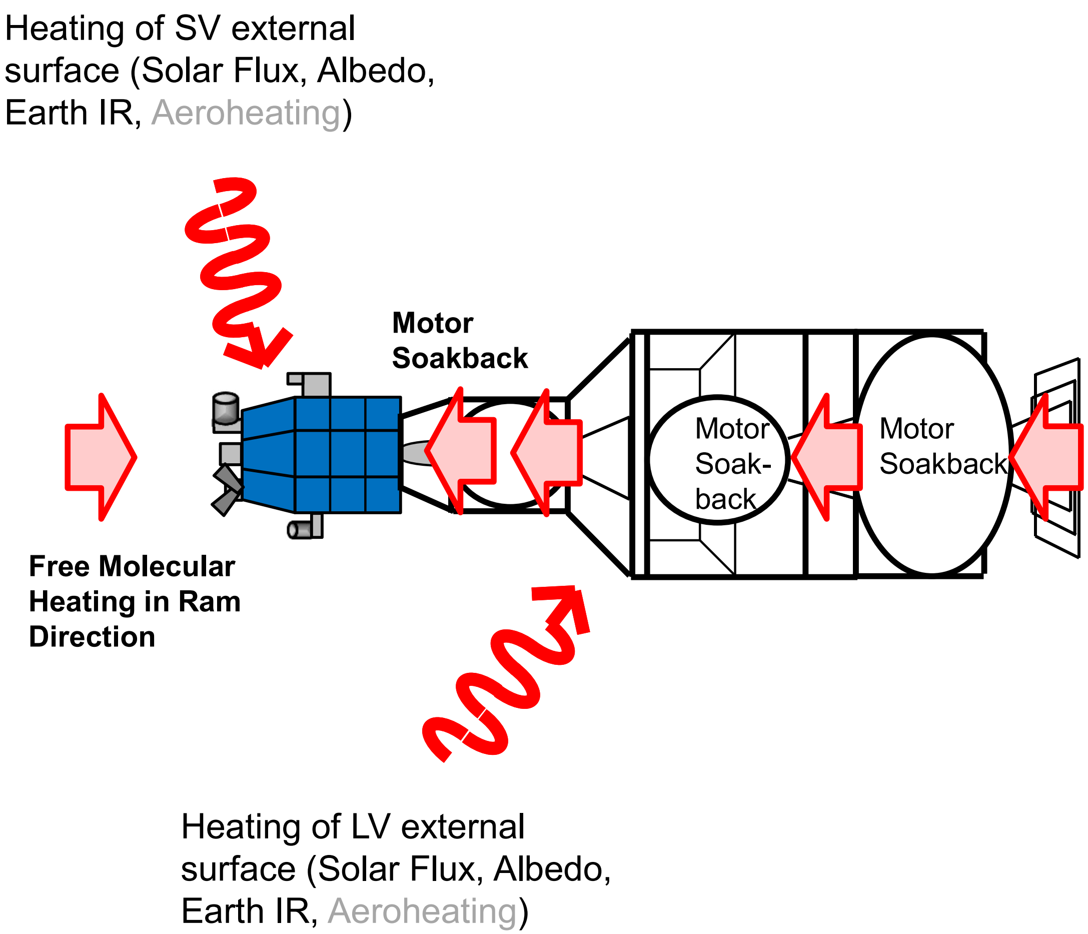


Mission Phases: Launch to Spacecraft Commissioning

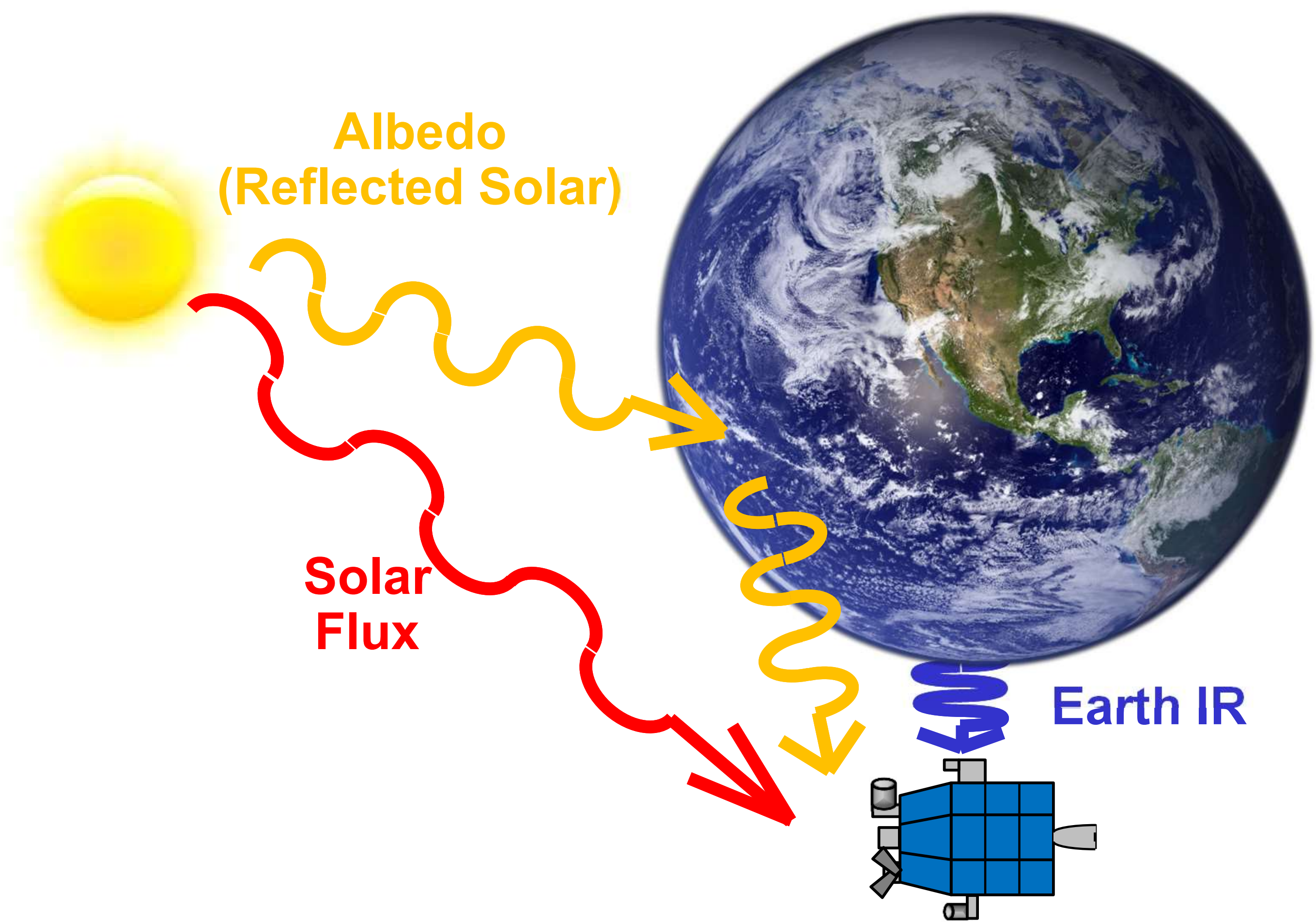


Convection to stagnant air (diminishes with loss of atmospheric pressure)

Mission Phases: Launch to Spacecraft Commissioning

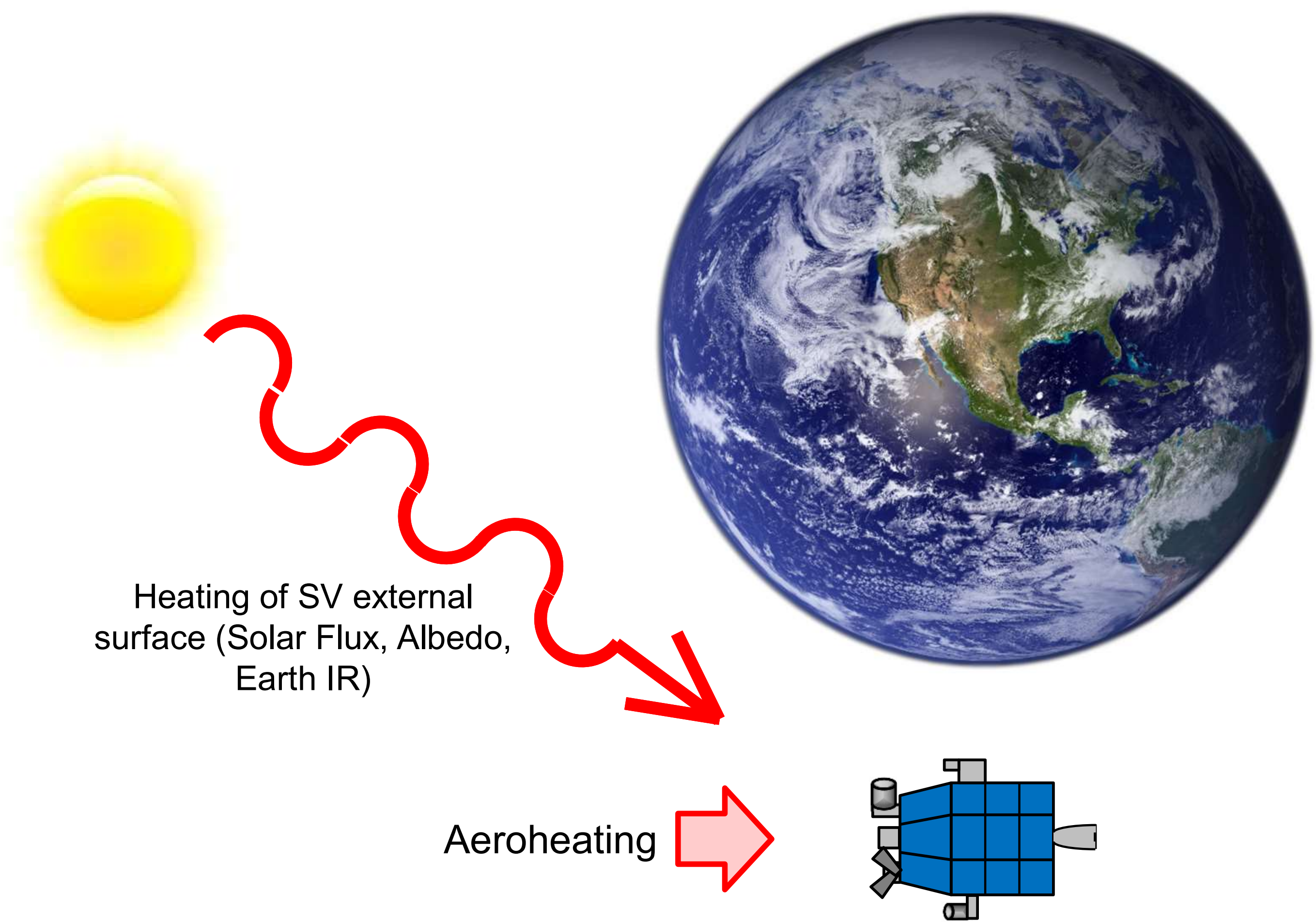


Mission Phases: Spacecraft Operations



Earth Orbit

Mission Phases: Decommissioning



Decommissioning

YK1

Thermal Design Development Process

Inputs	Thermal Design Step	Outputs	Key Issues
All subsystems: provide component thermal requirements	**1. Identify thermal requirements and constraints**	• System-level thermal requirements • Specialized requirements for specific equipment	• Identify payload thermal requirements and major elements that may present thermal challenges (see step 3)
ACS: orbit/attitude history **Mech:** S/C size/shape **All subsystems:** provide component thermal requirements	**2. Determine thermal environment**	• Total energy input into the spacecraft • Profile of energy input vs. time	• Max. and Min. distances to the Sun, Earth, or other central body • Chemical or nuclear internal heat sources
Thermal requirements from above **ACS:** attitude history **Mech:** equipment placement	**3. Identify thermal challenges or problem areas**	• List of specific thermal problem areas or problem times/events (hot, cold, or stability)	• Identify major elements that: - Generate large amounts of heat - Need cryogenic operating temps - Have boiling/freezing problems - Require a narrow temp range • Identify extraordinary thermal events/actions

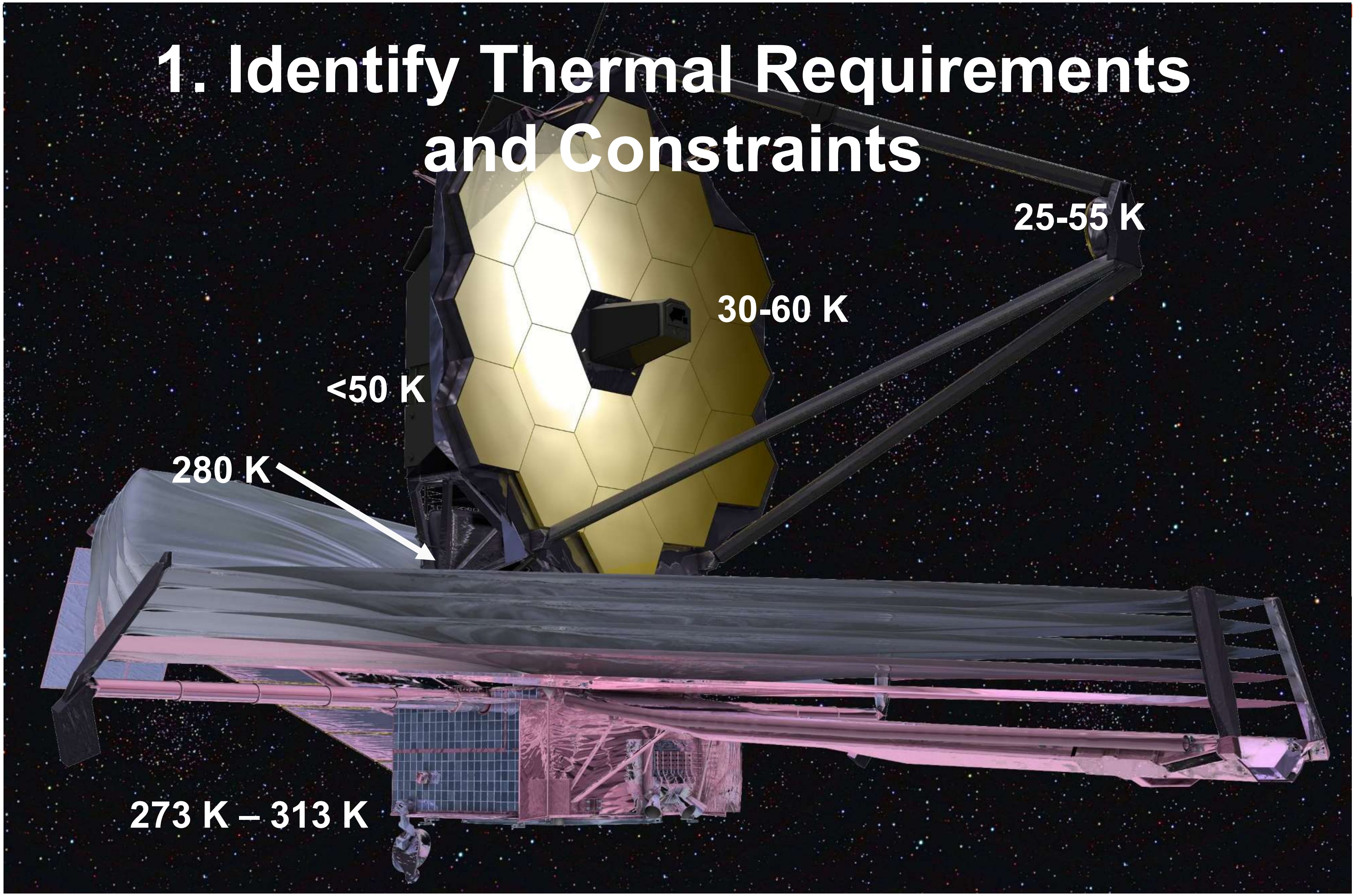
Slide 30

YK1 So with all of those mission phases in mind, we can now talk about process for designing your thermal system so that it can be protect the hardware from the thermal loads seen in every phase.
Yang, Kan, 2/28/2017

Thermal Design Development Process

Inputs	Thermal Design Step	Outputs	Key Issues
Thermal requirements and energy profile from above All subsystems: Additional constraints	4. Identify applicable thermal control techniques	• Preliminary list of thermal control mechanisms for mission duration and principal spacecraft components, areas, or times	• Prefer passive over active means • Component placement often key • Pay particular attention to problem areas / severe thermal constraints
List of thermal environments and events; Thermal control approach; component temp limits	5. Determine radiator and heater requirements	• Radiator sizes and temperatures to match hot cases with margin • Heater power for cold case thermal control	Take into account: - Degradation of thermal surfaces over mission life - Longest eclipse furthest from warm central body - Components that require a narrow temp range - Extraordinary thermal events/circumstances
List of TCS methods and components	6. Estimate TCS mass and power	• TCS Mass • TCS Power	• Typically 2-10% of total dry mass • May impact mass/power of other subsystems
	Document and Iterate	Thermal robustness can be key to system design, flexibility, and reducing operations cost	

Thermal Design Process



Thermal Requirements

- **All satellite components have thermal requirements**
  Thermal requirements come from the individual subsystem engineer, systems engineer, or scientist
  (contrary to popular belief, thermal doesn't make the thermal requirements!)
- **Three types of thermal design requirements:**
  - **Temperature Limits:** the acceptable range of temperatures for various modes and environments
    - Operational Temperature Limits: Exceeding can provide out-of-tolerance performance on components
    - Survival Temperature Limits: Exceeding can result in permanent equipment damage
  - **Stability:** the acceptable temperature or heater power change as a function of time (can be millisecond or multiple day basis)
  - **Gradient:** the acceptable spatial temperature difference

Thermal Requirements and Constraints

- **Typical temperature ranges for satellite components**

Component	Typical Temperature Ranges (°C)			
	Operational		Survival	
	Min	Max	Min	Max
Batteries	0	15	-10	25
Power Distribution/Power Box Baseplates	-10	50	-20	60
Reaction/Momentum Wheels	-10	40	-20	50
Gyroscopes/IMUs/IRUs	0	40	-10	50
Star Trackers	0	30	-10	40
C&DH Boxes	-20	60	-40	75
Hydrazine Tanks/Lines/Valves	15	40	5	50
Antenna Gimbals / External Mechanisms	-40	80	-50	90
Antennas	-100	100	-120	120
Solar Panels	-150	110	-200	130

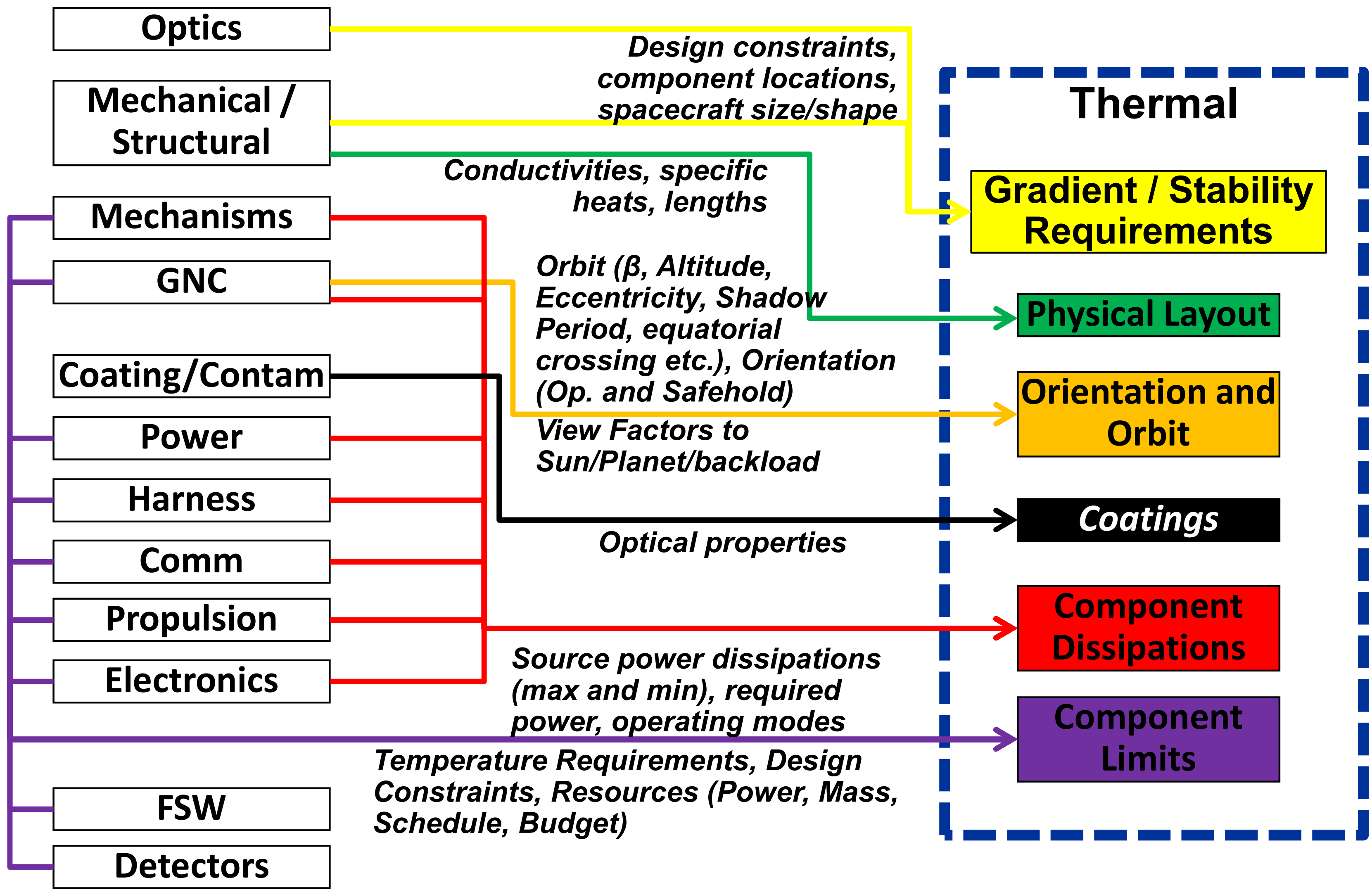
Source: Space Mission Analysis and Design, 3rd Edition

- **Structural/Stress Analysis may also dictate limits on structures**
  - **CTE mismatches at epoxied joints are a big source of thermal constraints on structure**
- **You must design your satellite so that all components fall within this range in the worst case hot and cold scenarios**

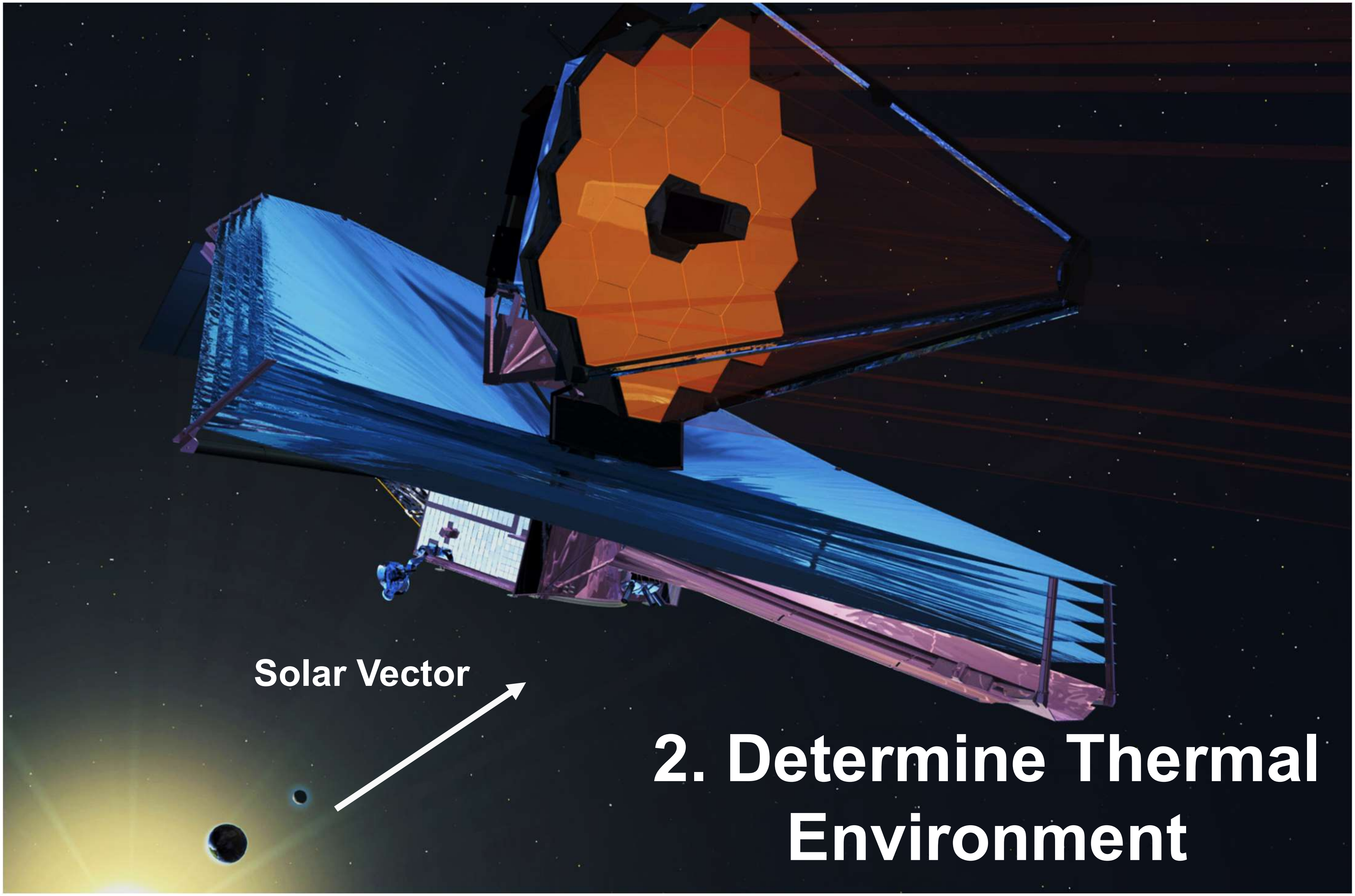
GEVS and Gold Rules

- **Each NASA center has its own design standards/requirements**
- **Gold Rules:** The Goddard Open Learning Design (GOLD) Rules specify sound engineering principles and practices, which have evolved in the Goddard community over its long and successful flight history. They are intended to describe foundational principles that "work," without being overly prescriptive of an implementation "philosophy." The GOLD Rules are a select list of requirements, which warrant special attention due either to their historical significance, or their new and rapidly evolving nature
  - 4.25 Thermal Design Margins
    - 5K margin from predicts to limits or <70% duty cycle on heaters
  - 4.27 Test Temperature Margins
    - Should test beyond limits (electronics)
  - 4.28 Thermal Design Verification
    - Must perform a thermal balance test
  - 4.29 Thermal-Vacuum Cycling
    - Components must have 8 thermal cycles before installation onto spacecraft
- **GEVS**: General Environmental Verification Standard (GEVS) For GSFC Flight Programs and Projects
  - This standard provides requirements and guidelines for environmental verification programs for GSFC payloads, subsystems and components and describes methods for implementing those requirements.

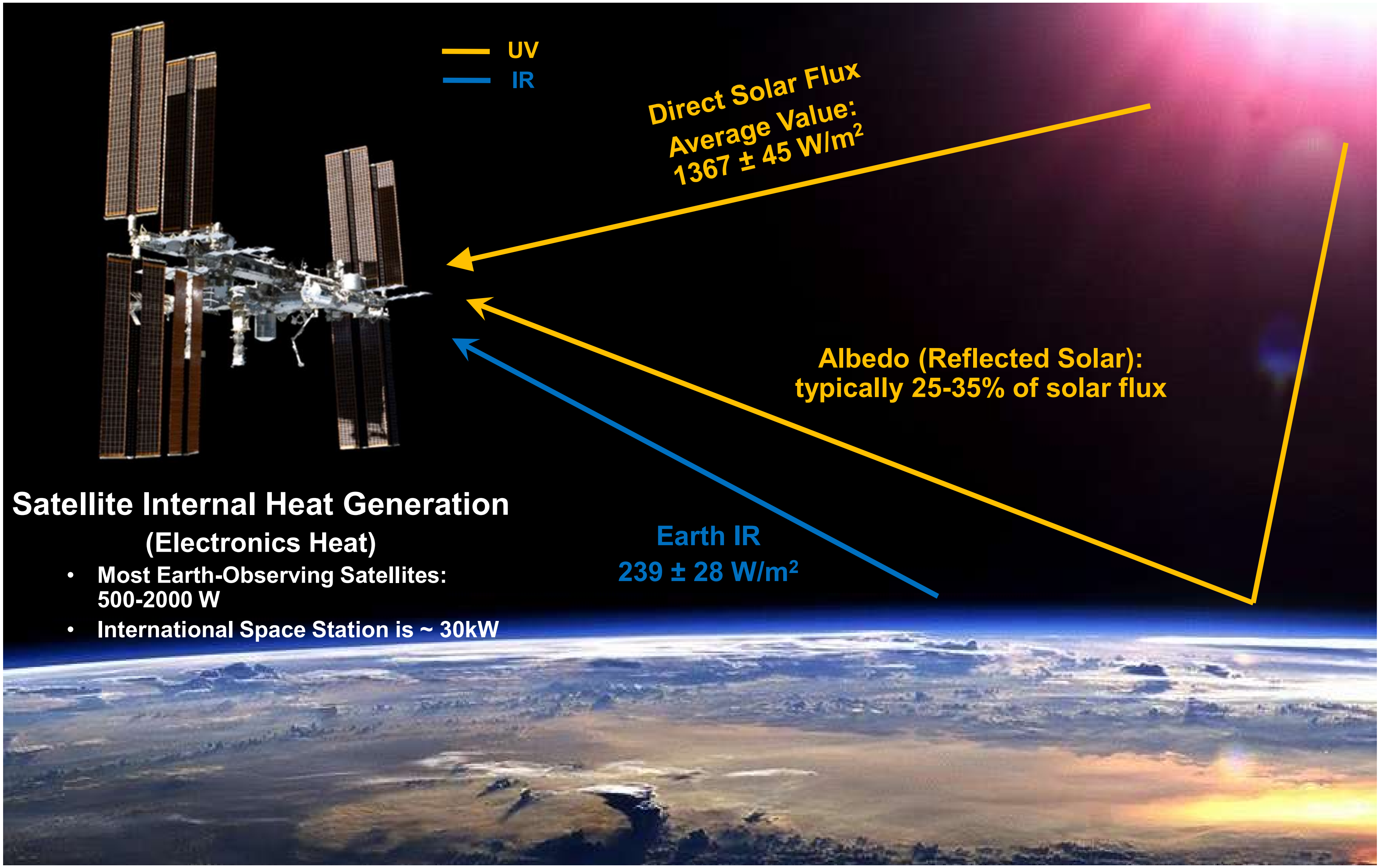
Key Information from Other Subsystems



THERMAL DESIGN PROCESS



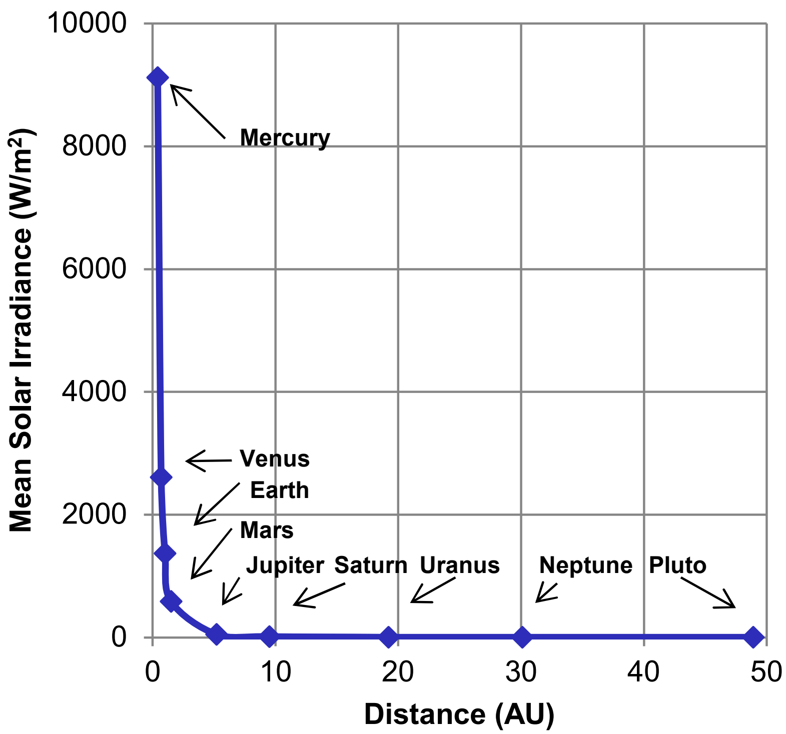
SPACE IS A HARSH ENVIRONMENT



Environment - Solar Irradiance

- Solar irradiance is the primary heating source to be considered in most satellites. The intensity varies with distance from the sun; inversely proportional to the square of the distance from the sun (energy from an expanding sphere)
  - At 1 AU, Average SOL = 1367 W/m^2
- Mission to outer planets may require something other than solar power (like radioisotope generators)

Planet	Solar Irradiance, W*m^{-2}		
	Mean	Perihelion	Aphelion
Mercury	9116.4	14447.5	6271.1
Venus	2611.0	2646.4	2575.7
Earth	1366.1	1412.5	1321.7
Mars	588.6	715.9	491.7
Jupiter	50.5	55.7	45.9
Saturn	15.04	16.76	13.53
Uranus	3.72	4.11	3.37
Neptune	1.510	1.515	1.507
Pluto	0.878	1.571	0.560

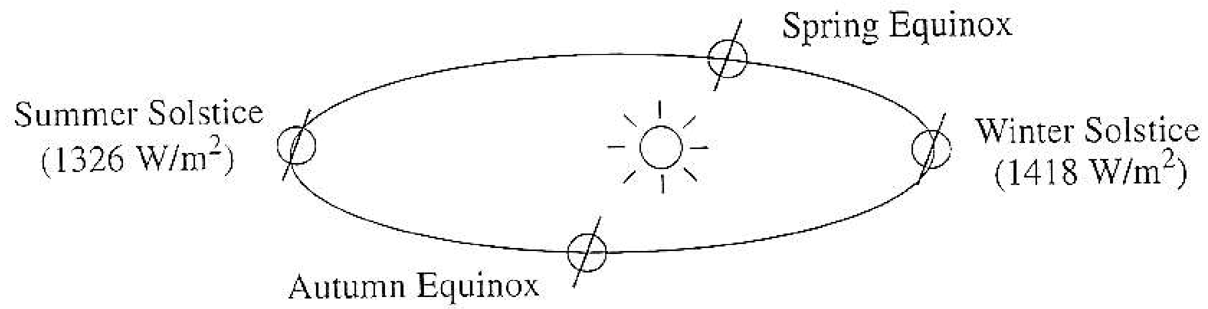


Related Fact

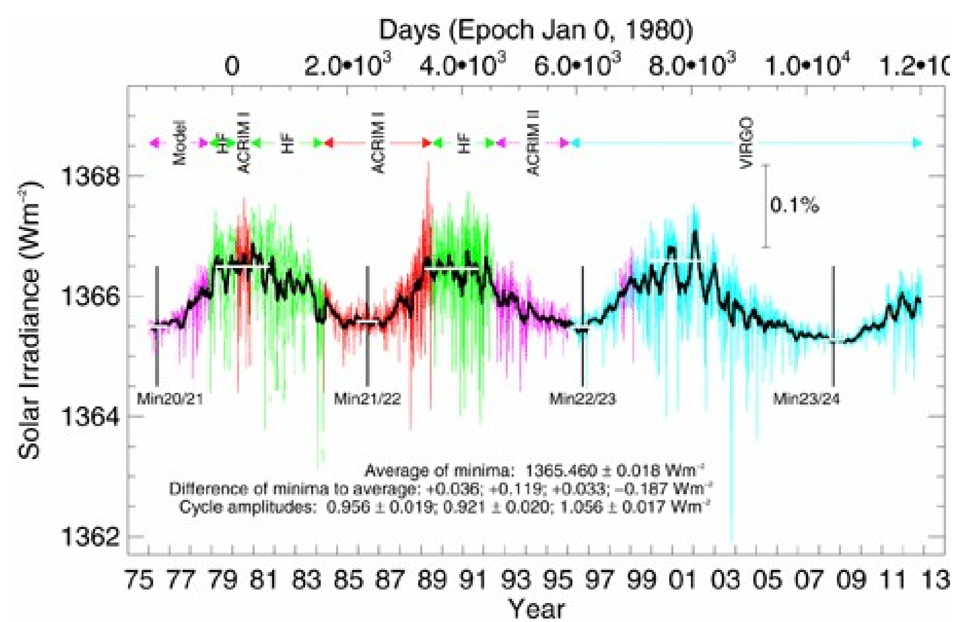
The MESSENGER Spacecraft, which studies Mercury, is protected from solar intensities as large as 14,000 W/m^2 using a ceramic sunshade. While temperatures in front of the shade can reach as high as 370°C, behind it the spacecraft can operate around room temperature (~20°C)

Environment - Solar Irradiance At The Earth

- **Earth's orbit is elliptical, the intensity of sunlight reaching Earth varies approximately ± 3.5%, depending on Earth's distance from the Sun**



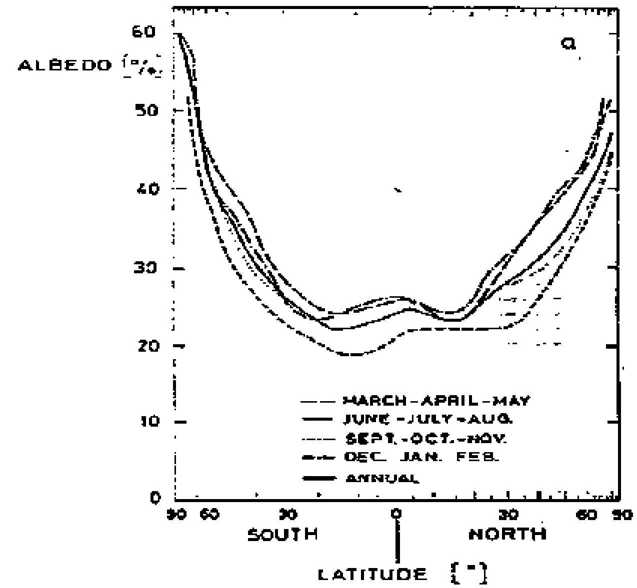
- **Also, the sun's intensity varies slightly (<0.5%) over an 11 year "solar cycle"**
- **Typical range of total solar irradiance variation:**
  1322 < SOL < 1412 W/m²

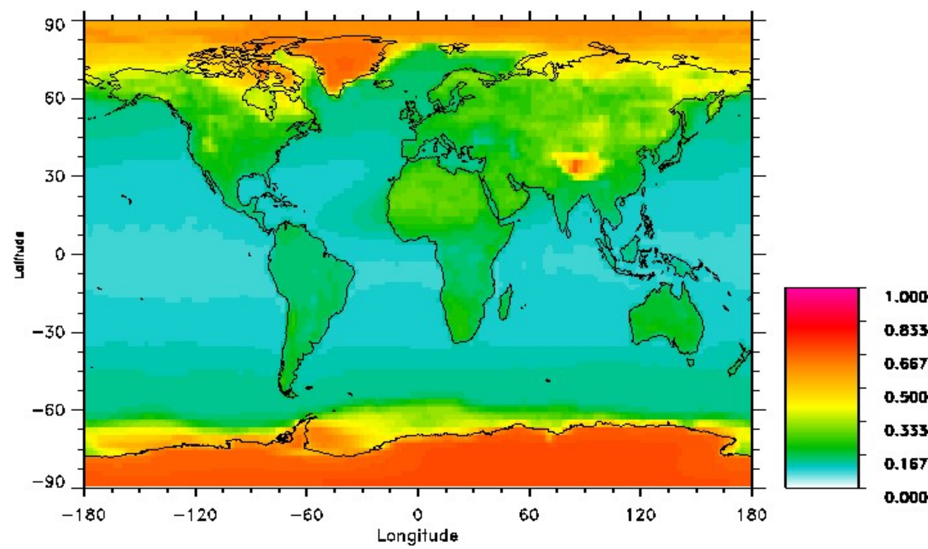


http://www.pmodwrc.ch/pmod.php?topic=tsi/composite/SolarConstant

Environment - Albedo

- **Albedo is sunlight reflected off a planet / moon / other body**
  - Mathematically, this is the ratio of UV reflected off a planet to the light received from a source, expressed as a number between zero (total absorption) and one (total reflectance)
- **Albedo is highly variable**
  - Usually, reflectivity is greater over land as compared with oceans and generally increases with decreasing local solar-elevation angles and increasing cloud coverage
  - Because of greater snow and ice coverage, decreasing solar elevation angle, and increasing cloud coverage, albedo also tends to increase with latitude
- **Typical: 0.25 < ALB < 0.35 (orbit average)**

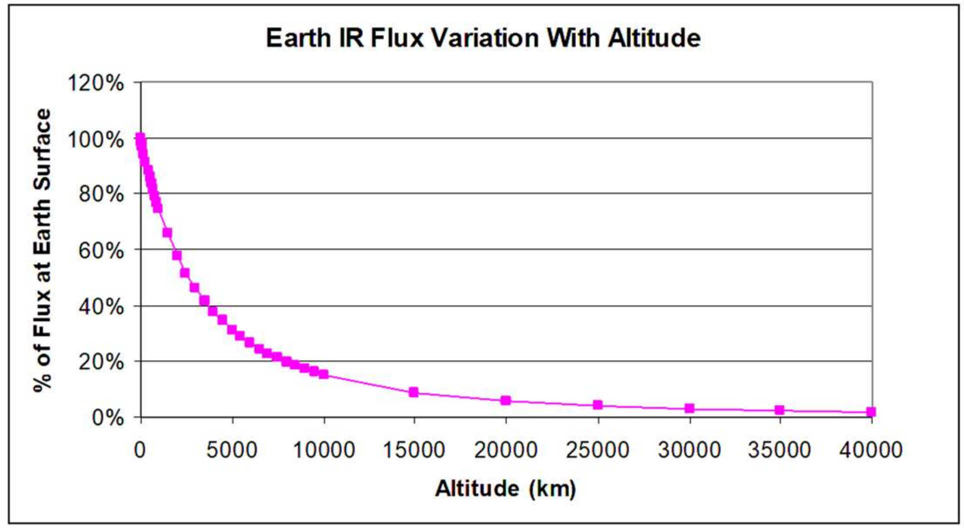




For More Information on Albedo:
Spacecraft Thermal Control Handbook, David Gilmore, et al, Aerospace Corp
Vol 1, Chapter 2

http://folk.uio.no/bjornhs/cicero/aerocom/A2/gallery/index.html

Environment - Planetary IR



- **Sunlight incident on the Earth not reflected as albedo is absorbed and eventually re-emitted as IR energy or blackbody radiation**
  - **The planet is normally characterized as a blackbody emitting radiation at a uniform temperature**
  - **Energy emitted at Earth's surface reduces with distance from the Earth:**
    R_{EARTH}^2 / r^2 where $r = R_{EARTH} + h$
  - **While this balance is maintained fairly well on a global annual average basis, the intensity of IR energy emitted (flux) at any given time from a particular point on Earth can vary considerably depending on:**
    - temperature of the Earth's surface (i.e. diurnal and seasonal variation, e.g. Up to 20% over deserts)
    - Amount of cloud cover absorbing shine
  - **Nevertheless, Earth IR variations are less severe than for albedo**
  - **Diurnal variations for bodies like the Moon (85K to 400K) or Mars (130K to 308K) are much more significant ➔ large fluctuations in IR over surface**
- **This results in EIR ~239 W/m^2, with approximately +/-28 W/m^2 variation. This correlates to a nominal Earth temperature of -18°C (254K)**
- **Depending on orbit altitude, EIR / Albedo may be negligible**

Orbital Mechanics: A Thermal Perspective

So when and how much do these orbital effects come into play during the course of the mission?

… use orbital mechanics to find out!

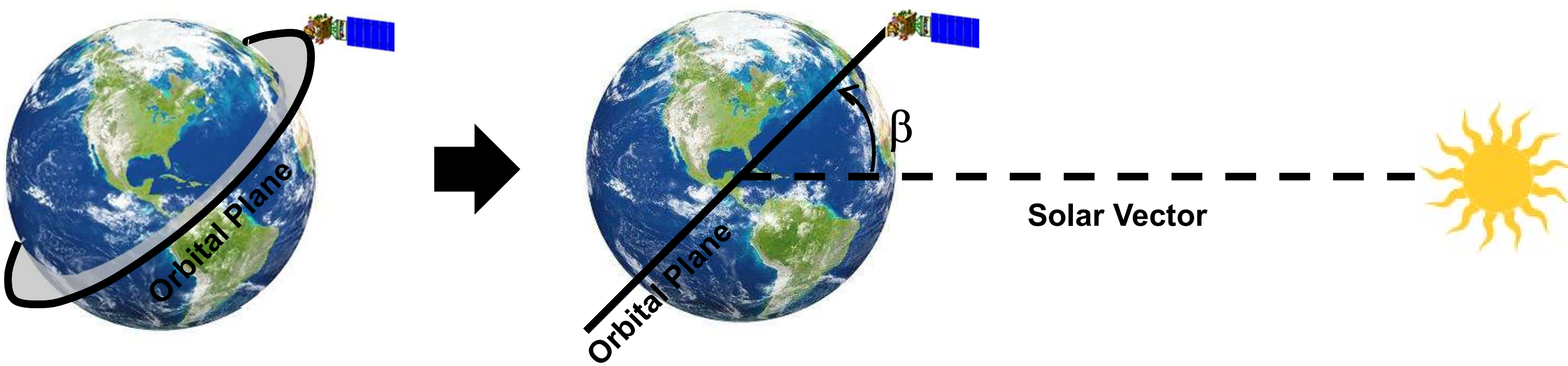
- **Orbital mechanics allows understanding of the heat inputs / other variations from the external environment (Solar, Albedo, Earth IR) during the mission lifetime:**

  ➔ Since the most important parameter in the thermal control of a satellite is the external environment …

  … understanding the relationship of the satellite to the environment throughout the mission is key in establishing the analytical hot & cold bounding cases
- **For satellites orbiting Earth (or any moon/planet), the nodal crossing, altitude and orbit inclination result in a range of "beta" angles**
  - The term beta (β) angle is defined as the minimum angle between the solar vector and the orbit plane: $-90° < \beta < +90°$
  - Some missions are designed so that the orbit regression rate equals the rate of change of the sun's right ascension, resulting in a "sun-synchronous" orbit with a small beta angle range

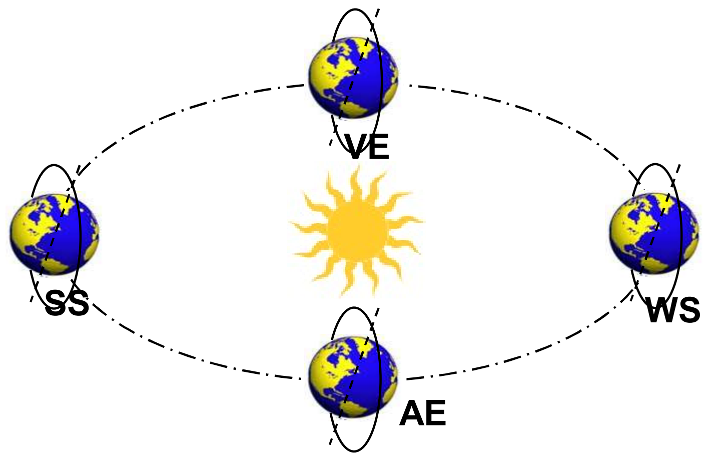
Orbital Mechanics: A Thermal Perspective

- **Think about beta angle like this:**

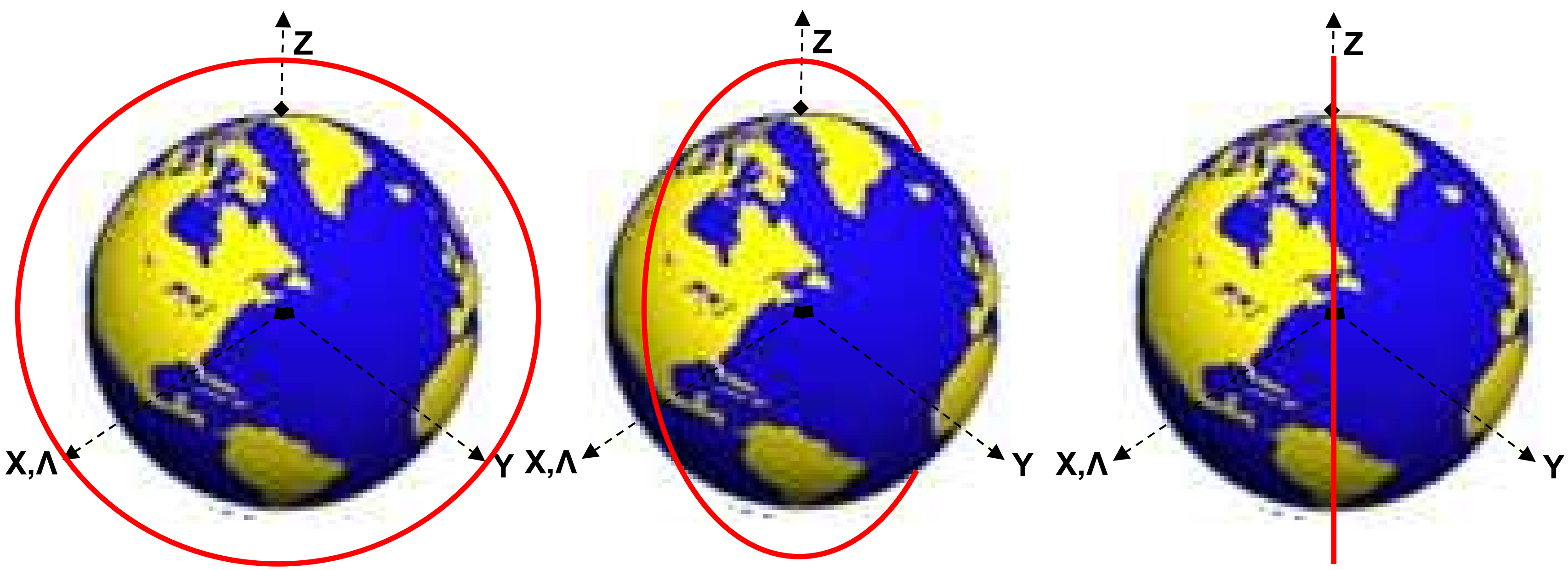


- **Determining the range of beta angles for your mission is key to understanding the solar input, for both thermal and power reasons**
- **Also, the orientation of your satellite in the orbit may change the sun angles on the satellite surfaces**

Beta Angle – Non-synchronous

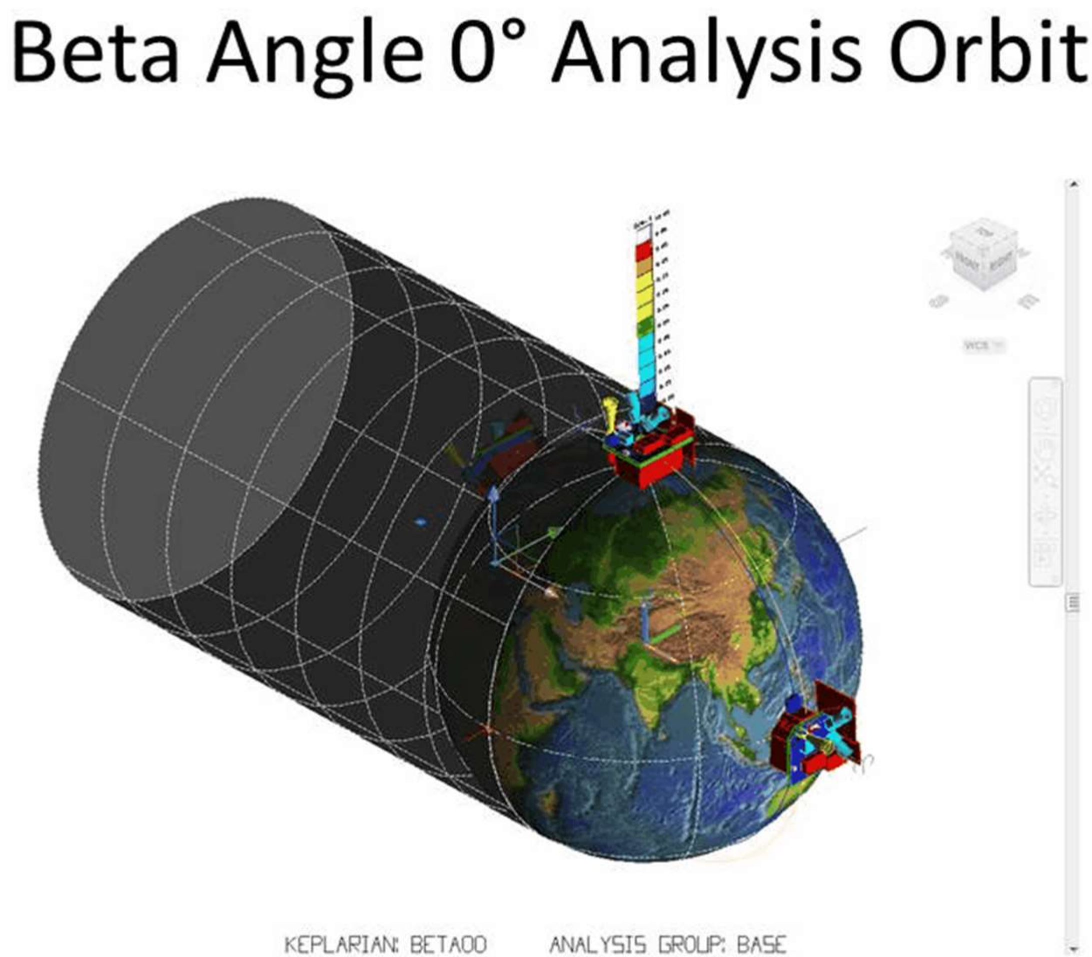


View from the sun:



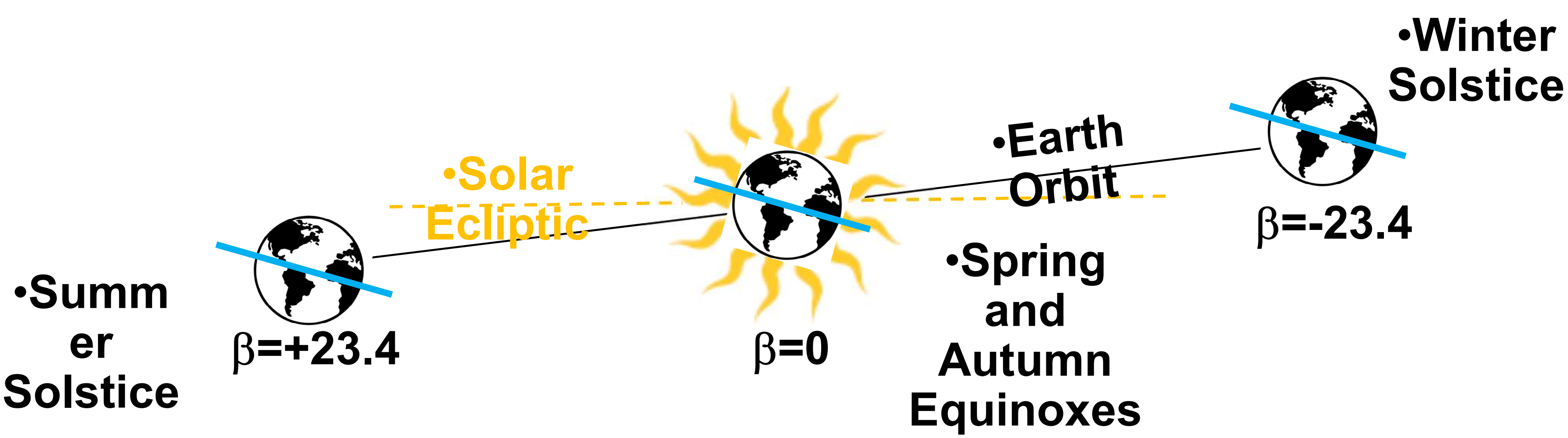
$\beta = 90°$ $\quad 0° < \beta < 90°$ $\quad \beta = 0°$

Beta Angle Animation



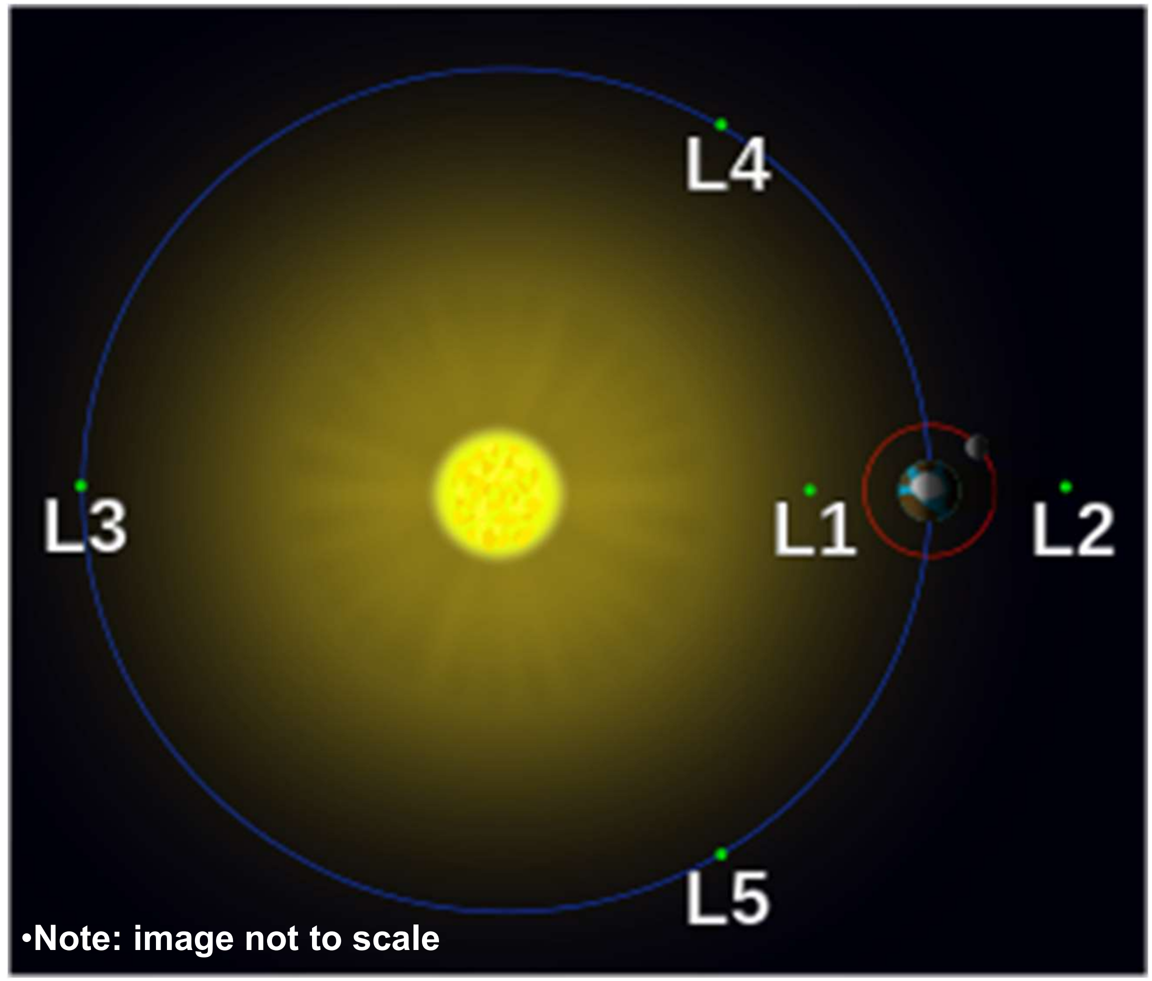
Geostationary Orbit

- **Many missions orbit in a Geostationary orbit, where the period is exactly equal to one Earth day and the orbit is in the equatorial plane.**
- **These orbits have the advantage of observing the same ground location at all times as the satellite rotates with the Earth to remain fixed above the ground, relatively speaking**
- **Due to the tilt of the Earth's rotational axis relative to the normal to its orbit plane, the range of Beta Angles for Geostationary orbits is -23.4° to +23.4°**
- **In general, the only consequential environmental source for Geostationary orbits is Solar. <u>Albedo and Planet IR are usually negligible</u>**



Lagrange Points

- **Lagrange points are special locations in the solar system where the gravitational and body forces balance. These points are desirable for spacecraft since they require considerably less fuel to maintain their position**
- **Besides being optimum for fuel usage, the Lagrange points are also good thermal points as they present very stable and consistent environments**
- **L1 is at about 0.99 AU. Note that spacecraft here will see full sun**
- **L2 is at about 1.01 AU. Spacecraft here need to orbit outside of the shadow cone of Earth to allow for illumination of the Solar Arrays**
- **<u>Lagrange points also can generally neglect albedo and Earth IR environmental sources</u>**



Environmental Heat Transfer to Satellite

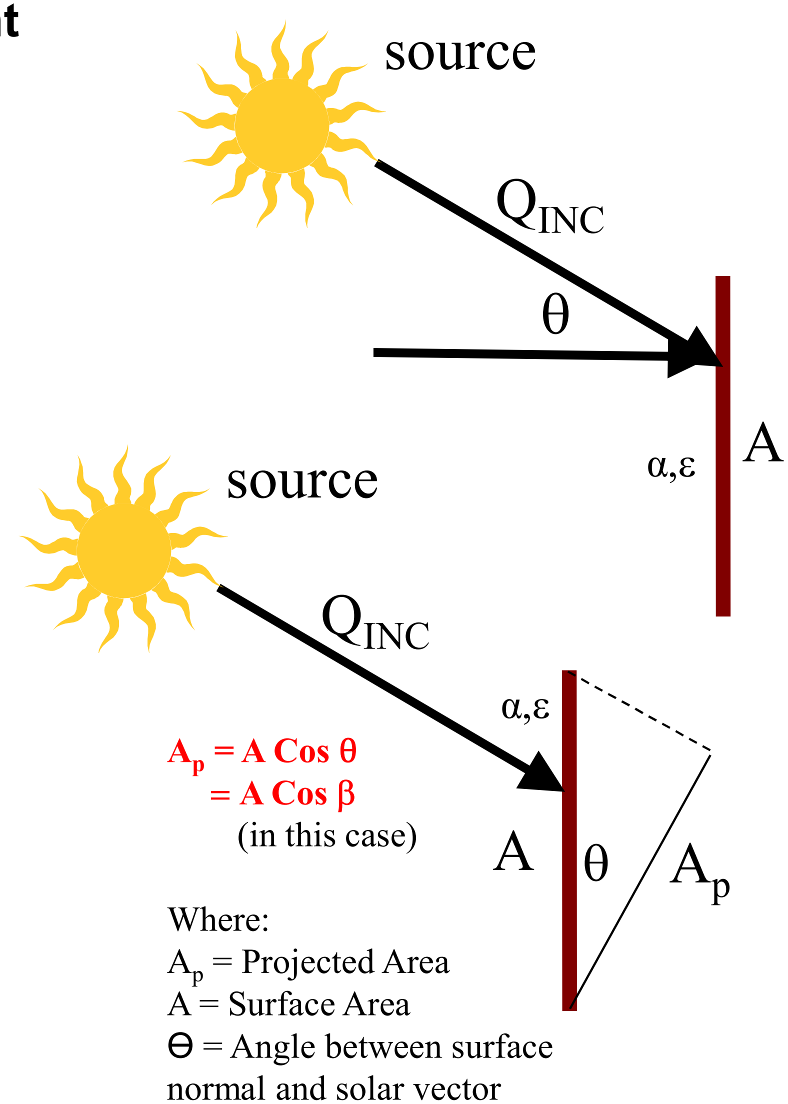
- **The amount of heat transfer from the environment to a satellite surface is determined by the orientation and optical properties (α, ε) of that surface**

Solar Irradiance = $\mathbf{A_{P,\ Solar}\ Q_{solar}}\ \alpha_{\mathbf{rad}}$

Albedo = **(Albedo Fraction)** $\mathbf{A_{P,\ Albedo}\ Q_{solar}}\ \alpha_{\mathbf{rad}}$

Earth IR = $\mathbf{A_{P,\ Earth\ IR}\ Q_{Earth\ IR}}\ \varepsilon_{\mathbf{rad}}$

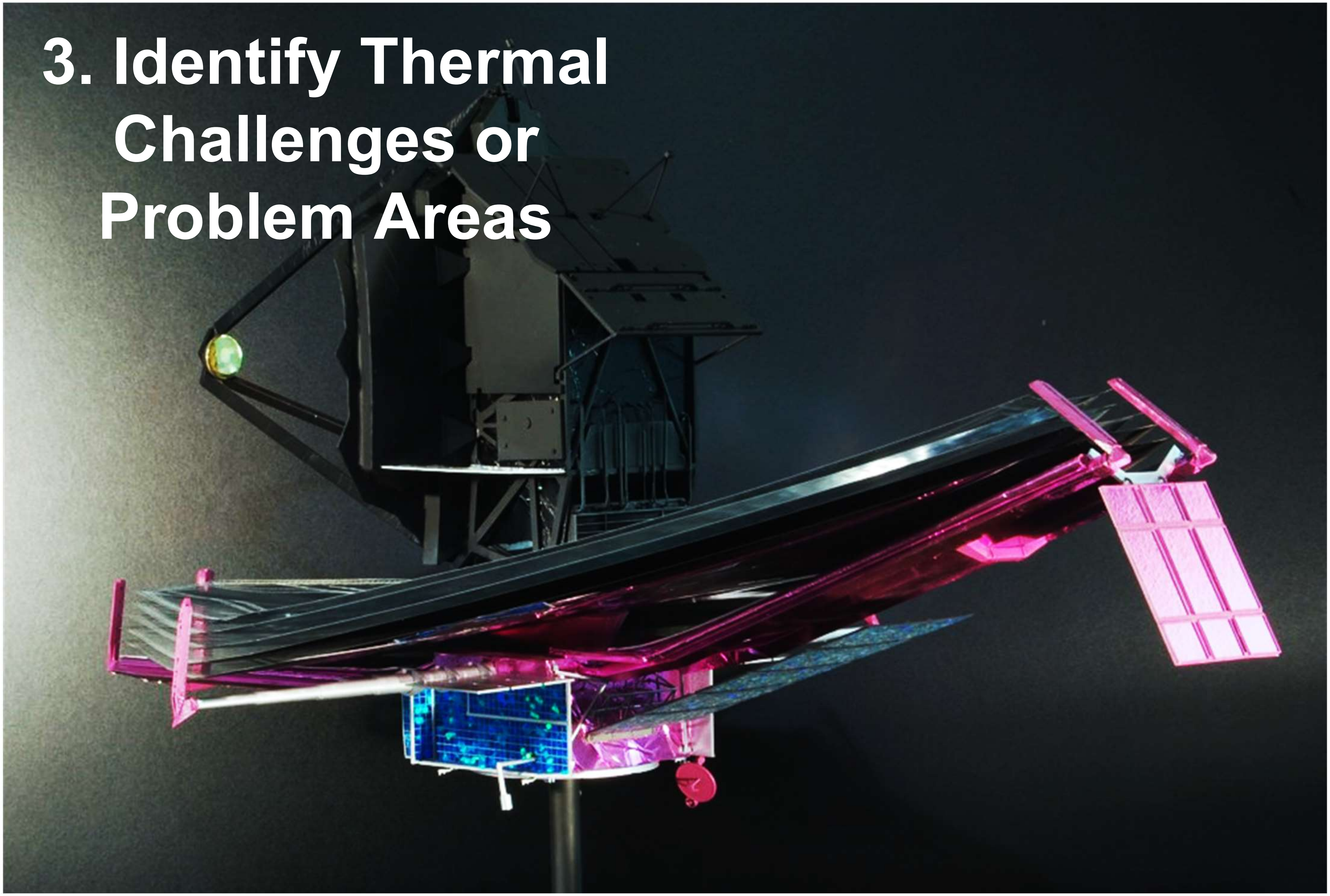
Where:	<u>SI</u>
Q - Heat Flow	Watts
T - Absolute Temperature	K
A - Surface Area	m^2
A_P - Projected Area	m^2
Albedo Fraction	0.25 – 0.35 (dimensionless)
Q_{solar} - Solar Constant	1367±45 W/m^2
Q_{solar} – Typical Earth IR Flux	239±28 W/m^2
α - Solar Absorptance	$0 \le \alpha \le 1$ (dimensionless)
ε – IR Emittance	$0 \le \varepsilon \le 1$ (dimensionless)



Total Absorbed Heat:

$$Q_{ABS} = Q_{SOL}\alpha A\cos\theta_{INC,\,SOL} + (\text{Albedo Fraction})\,Q_{SOL}\,\alpha A\cos\theta_{INC,\,ALB} + Q_{Earth\ IR}\,\varepsilon A\cos\theta_{INC,\,Earth\ IR}$$

THERMAL DESIGN PROCESS

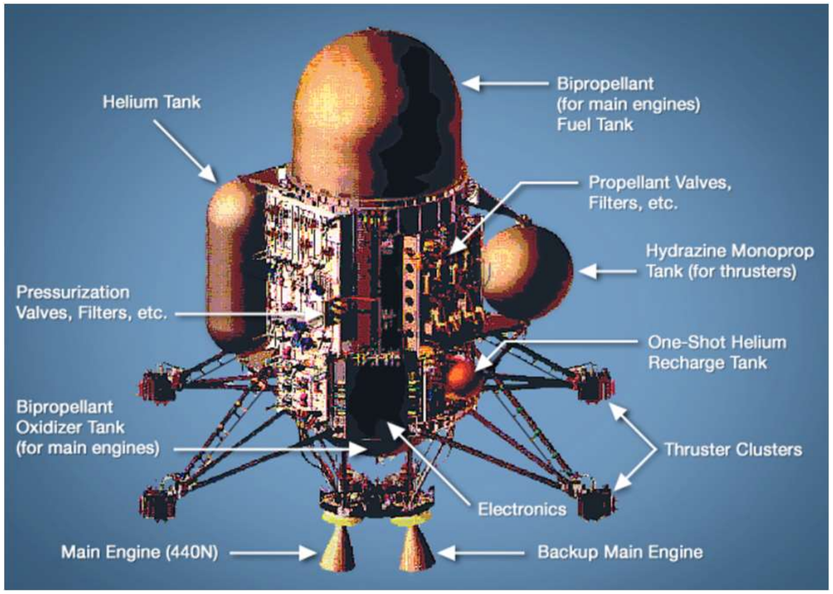


Thermal Design Challenges

- **Additional thermal requirements are flowed down from higher level requirements**
- **Identify major elements that:**
  - Generate large amounts of heat
  - Need cryogenic operating temps
  - Have boiling/freezing problems
  - Require a narrow temp range
  - Have tight thermal stability requirements
  - Require high temperatures
- **Identify extraordinary thermal events/actions**
  - Post launch orientations
  - Long cruises
  - Orbit insertion thruster burns

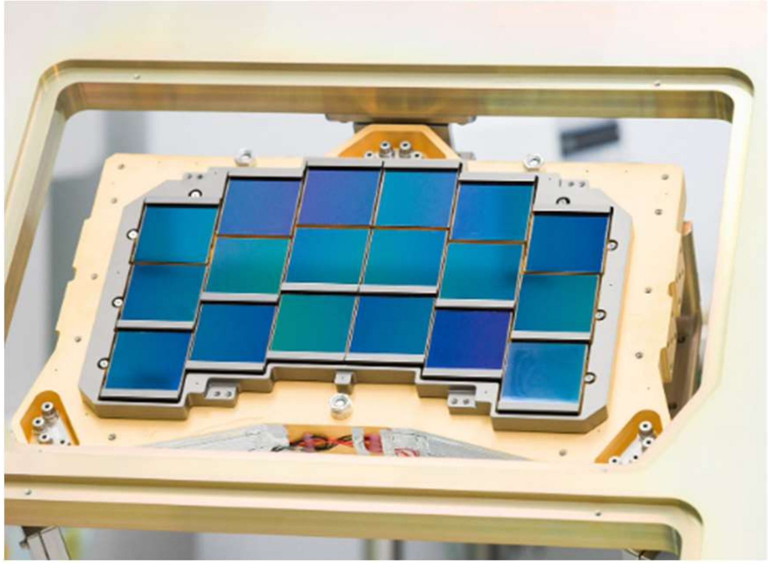
Narrow Temperature Range / Thermal Stability

- Propulsion Systems must be kept in a narrow temperature range
- Detectors, depending on the type, have to be kept cold
- Optical supports need to be kept thermally stable or focus could be affected



Cassini Propulsion System: https://solarsystem.nasa.gov/basics/chapter11-4/

Roman Space Telescope Optical Telescope Assembly: https://roman.gsfc.nasa.gov/science/



Roman Space Telescope Wide Field Focal Plane Detector Array: https://roman.gsfc.nasa.gov/science/WFI_technical.html

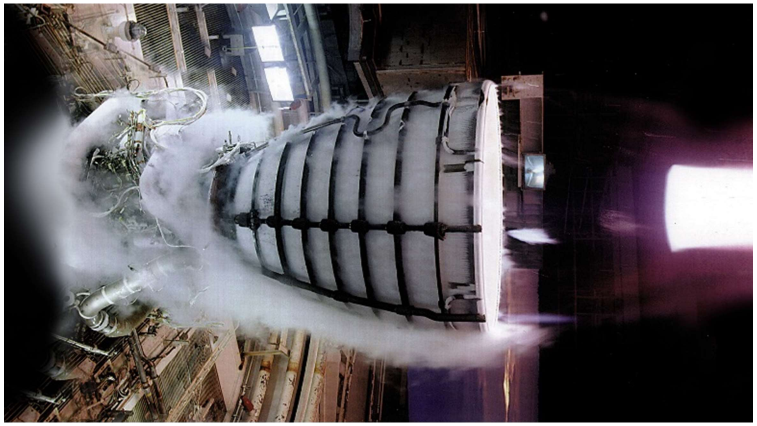
Sources of Heat Dissipation

- **Almost all satellite components generate heat**
  - **What are the largest sources of heat generation?**

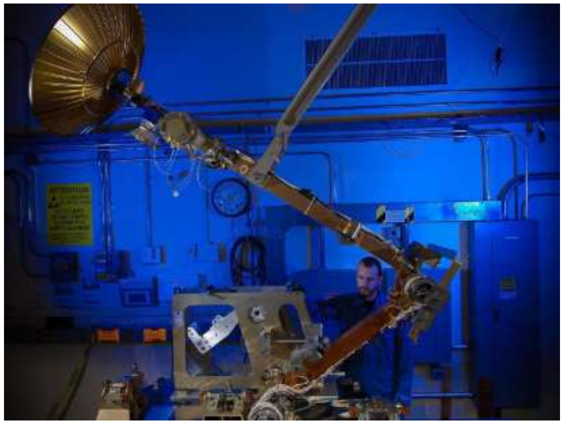
Batteries

Electronics

ACS Components



Propulsion

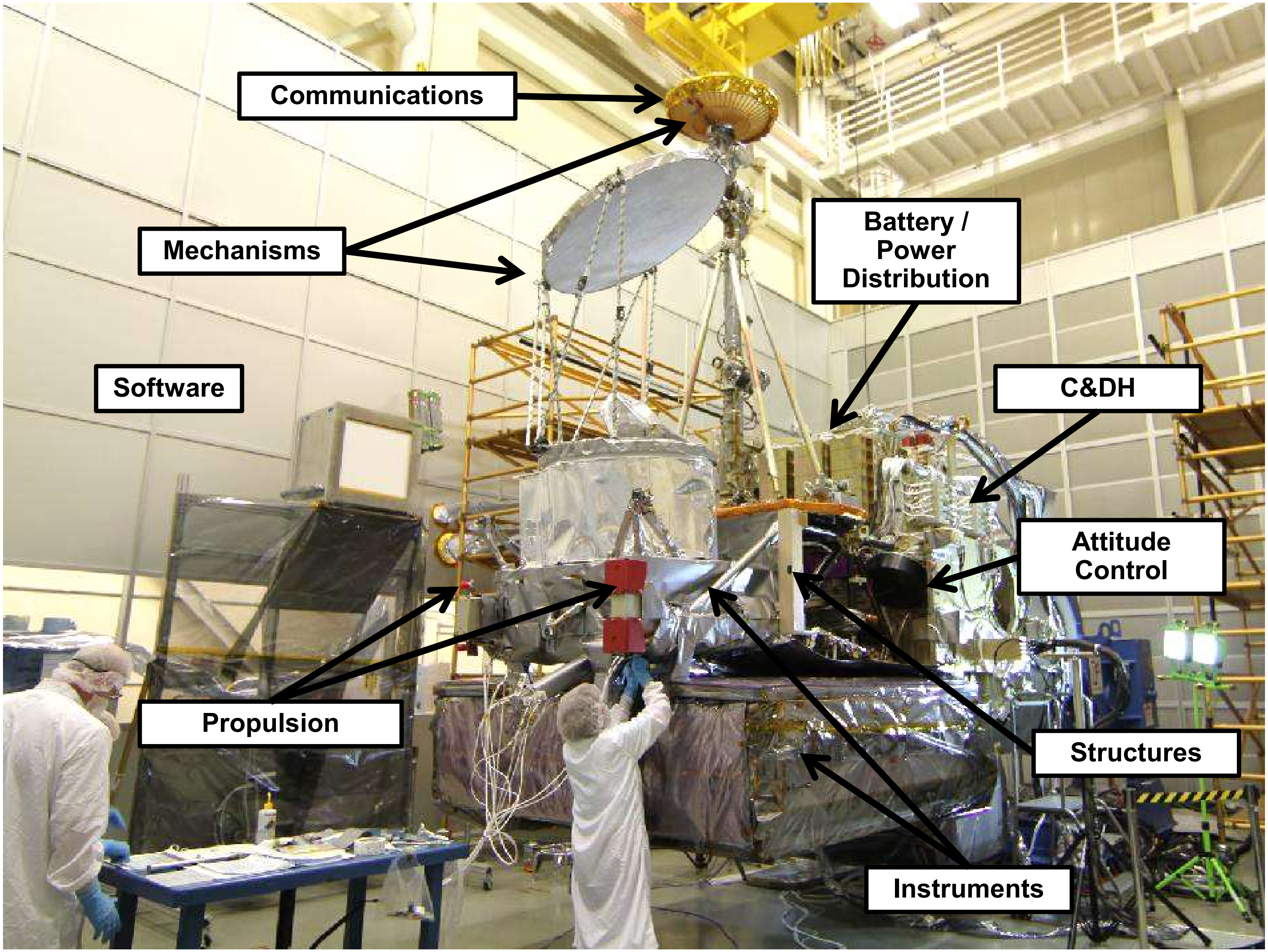


Mechanisms

Instruments

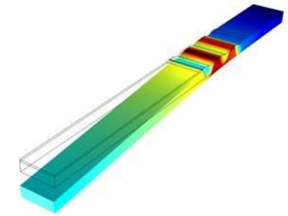
- **How do we reject the heat from the satellite into space so that it doesn't overheat internal temperatures? Use heat transfer principles!**

What other subsystems does thermal affect?



Affected Subsystems

Structures:



- Most primary/secondary structure is bonded or bolted panels with adhesive-limited temperature range
- CTE effects will limit range and transients for pointing error

Command & Data Handling (C&DH):

- Computer, Electronics Boards, Telemetry I/F do not work consistently at non-optimal temp. ranges

Battery / Power Distribution:

- Batteries have very strict operational temperature requirements (usually these are the most narrow allowable temperature ranges on spacecraft)

Communications:

- Antennas and RF transponders have limited operational temp. range
- Antennas especially can get very cold when pointed towards deep space

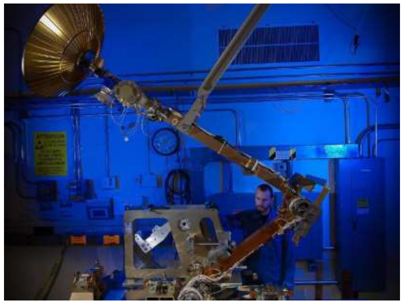
Attitude Control:

- Inertial Measurement Units (IMU), Star Trackers (ST), Magnetic Torquer Bars (MTBs), and Reaction Wheel Assemblies (RWAs) all have stringent temperature requirements
- Reaction wheels can get very hot when spun up

Affected Subsystems

Propulsion:

- Thrusters and other associated propulsion components must handle very high temperatures, yet be isolated from rest of the spacecraft
- Tanks / valves / lines must be kept in tight temperature range due to fuel constraints

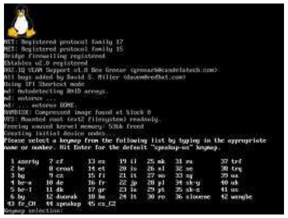


Mechanisms:

- Most actuate near ambient
- Some used only in Early Operations (e.g. actuators)
- Other mechanisms others must work throughout the mission lifetime (e.g. gimbals)

Software:

- Algorithms for control of thermal components (heaters, sensors, etc.) need to be programmed into the flight software
- Flight software manages temperature limit alarms



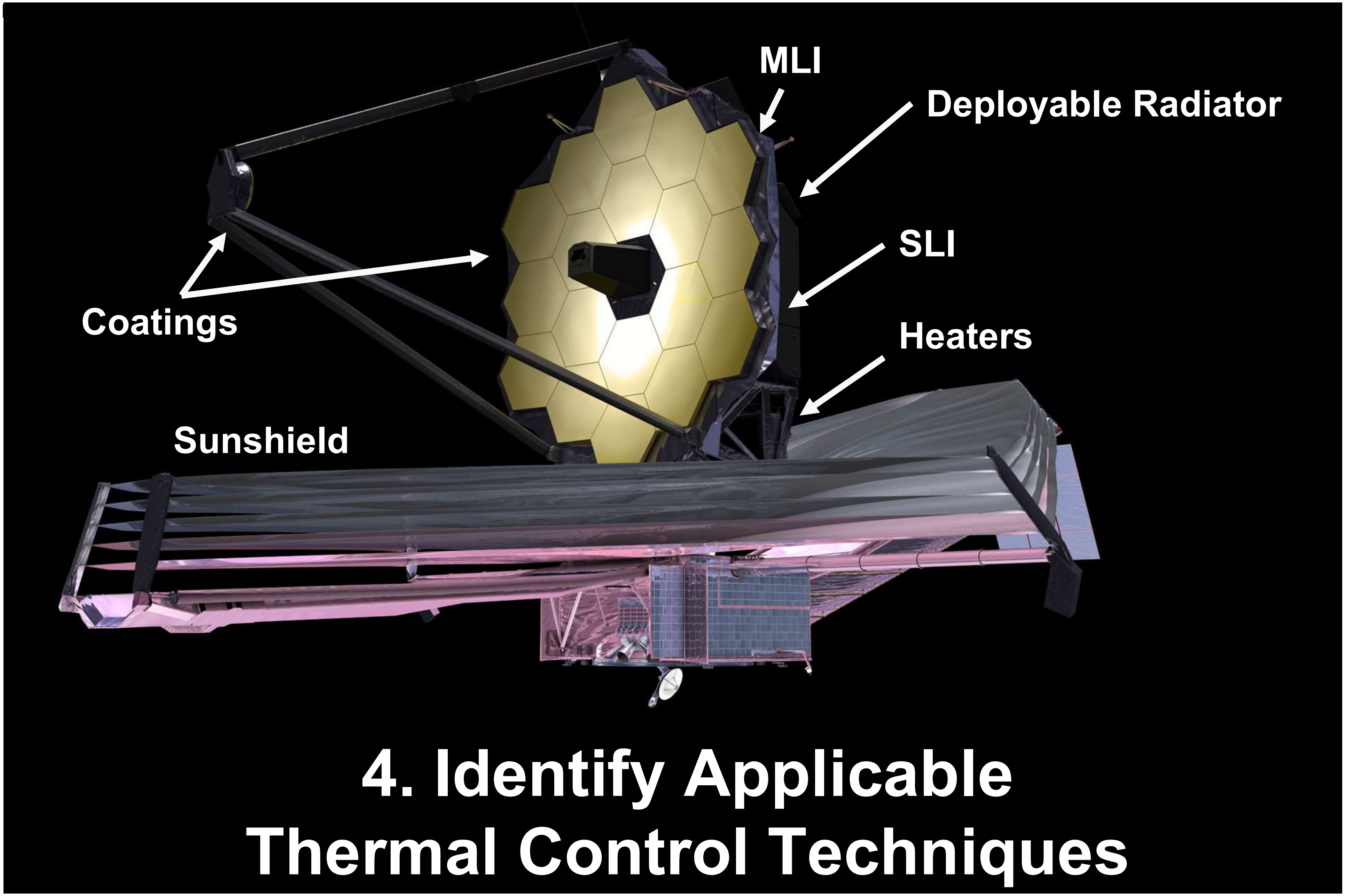
Instruments:

- Typically instruments have their own separate thermal control system
- Interfaces from spacecraft bus to instruments must be kept in tight temperature range ➔ no heat leaks between spacecraft and instrument

Identifying Problem Areas

- **Simple hand calculations will determine the steady-state temperatures of all the spacecraft components at stacked worst case hot and cold conditions**
  - This allows you to identify the "problem areas" that may require additional heat (from a heater) or require additional cooling (from a radiator) to maintain a component within limits
- **Sometimes, specialized thermal hardware (such as active thermal hardware) is needed because passive components cannot meet the thermal requirements**
  - Typical examples include:
    - Cryogenics for optical instruments
    - Insulation for high-temperature applications (such as thruster firing or solar-observing instruments)
    - Active thermal control for instruments / electronics with narrow operating temperature ranges
    - Transport of heat from an area of large dissipation to a cold sink
    - Spreading of heat from a "hot spot" to a uniform surface
  - These will be discussed in the following section

THERMAL DESIGN PROCESS

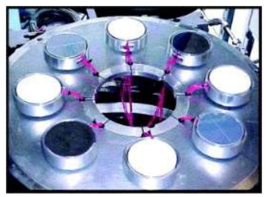


Passive Systems

- Basis of all thermal control
- Requires no control functionality to perform
- Based on the thermal behavior of the S/C surfaces
- Low mass, volume and cost requirements
- High reliability
- Lifetime limited only by degradation of thermo-optical properties
- Examples:

RADIATIVE

Multi-Layer Insulation (MLI) Blankets



Thermal Coatings

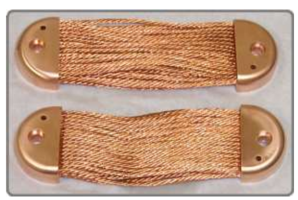
Radiators / Radiative Coolers

HEATER CONTROL

Thermostat / bi-metallic control

CONDUCTIVE

Doublers: plates to distribute heat more evenly over an area



Heat Straps (to transport heat from one area to another)

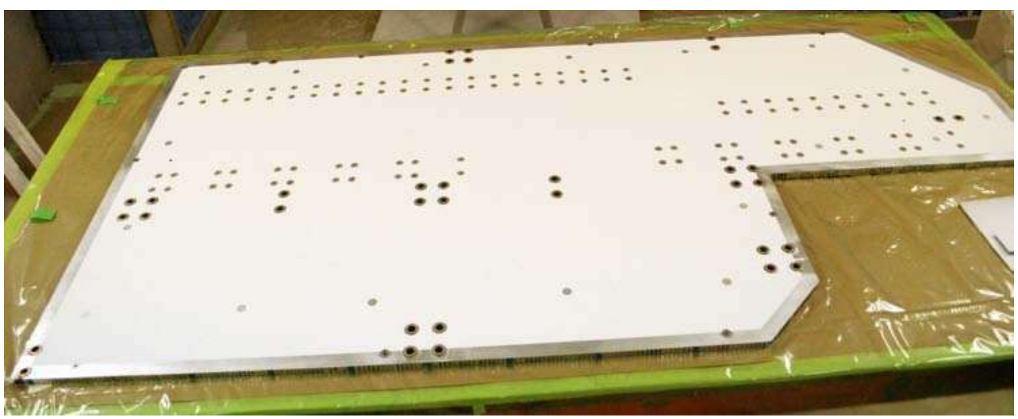
Isolators: minimize heat transfer to/from components, i.e. batteries, prop lines.

Phase change materials can also limit heat absorption

Constant Conductance Heat Pipes

Passive Example: Radiators and Thermal Control Coatings

- **Radiator – a dedicated structure whose purpose is the rejection of waste heat to deep space**
  - Coated with high emissivity coating to maximize heat rejection potential
  - May be coated with high or low solar absorptivity coating depending on view to solar sources
  - If not existing structure, then supports are needed
- **Coatings – films, tapes, paints, etc. applied to surfaces to obtain the desired thermo-optical properties for thermal control**
  - Thermo-optical properties are intrinsic to the material itself
  - a – Solar Absorptivity – percentage of sun energy (Direct Solar, Albedo) absorbed
  - e – IR Emissivity – percentage of planet energy (Planetshine) absorbed
    - Emissivity is also a measure of emissive capability of a surface to reject heat via IR radiation



Surface Finish	α (Beginning of Life)	ε
Optical Solar Reflectors		
8 mil Quartz Mirrors	0.05 to 0.08	0.80
2 mil Silvered Teflon	0.05 to 0.09	0.66
5 mil Silvered Teflon	0.05 to 0.09	0.78
2 mil Aluminized Teflon	0.10 to 0.16	0.66
5 mil Aluminized Teflon	0.10 to 0.16	0.78
White Paints		
S13G-LO	0.20 to 0.25	0.85
Z93	0.17 to 0.20	0.92
ZOT	0.18 to 0.20	0.91
Chemglaze A276	0.22 to 0.28	0.88
Black Paints		
Chemglaze Z306	0.92 to 0.98	0.89
3M Black Velvet	~0.97	0.84
Aluminized Kapton		
1/2 mil	0.34	0.55
1 mil	0.38	0.67
2 mil	0.41	0.75
5 mil	0.46	0.86
Metallic		
Vapor Deposited Aluminum (VDA)	0.08 to 0.17	0.04
Bare Aluminum	0.09 to 0.17	0.03 to 0.10
Vaporized Deposited Gold	0.19 to 0.30	0.03
Anodized Aluminum	0.25 to 0.86*	0.04 to 0.88*
Miscellaneous		
1/4 mil Aluminized Mylar, Mylar Side	(Material degrades in sunlight)	0.34
Beta Cloth	0.32	0.86
Astro Quartz	~0.22	0.80
MAXORB	0.9	0.1

* Anodizing and similar surface treatments must be carefully controlled in order to produce repeatable optical properties.

Source: Space Mission Analysis and Design, 3rd Edition

Passive Example: Multi-Layer Insulation

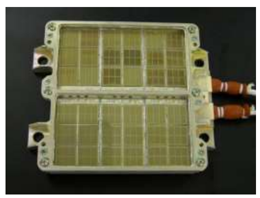
- **Sensitive electronics and other thermally-restrictive components must be protected from environmental extremes**
- **Thermal blankets (MLI) used to insulate against excessive heat loss (or gain)**
  - Most common thermal control elements on satellites, covering most of the external surface with cut-outs for radiator areas to reject the internal heat load
  - Heat transfer through blanket can be represented with blanket effective emissivity, e*
  - Kapton/mylar layers are highly specular to reflect radiative heat from the environment; dacron mesh separators prevent conduction through the blanket
  - In LEO, outer layer material must be sized to withstand Atomic Oxygen (AO) erosion: degradation of blanket over mission lifetime affects its optical properties and ability to insulate
  - MLI must be grounded via ground straps and conductive outer layer material
  - Sometimes, Kevlar is included in blanket layup for Micrometeoroid protection
  - Mechanically supported by buttons, velcro, tie-downs, or double sided transfer adhesives (e.g. tapes)
  - If it is not a radiator, aperture, or solar array, thermal probably wants a blanket there…

Active Systems

- Complement passive systems
- Requires control interface to implement
- Requiring power input and/or mechanical moving parts
- Able to cope with large heat loads and variations in power dissipation
- Disadvantages: High mass, volume, power and cost requirements; reliability and lifetime issues
- Examples:

RADIATIVE

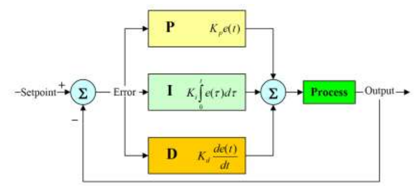
Louvers with feedback control



Variable Emittance Coatings

HEATER CONTROL

Electronic / software controlled heater



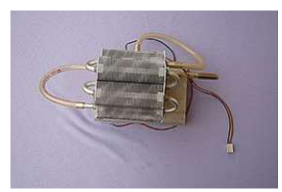
Proportional Controllers (for stability, instead of thermostats)

CONDUCTIVE

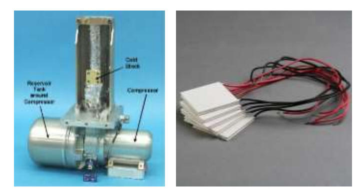
Dewars (for cold temp operations)

Variable Conductance Heat Pipes (VCHP)

Loop Heat Pipes (LHP)



Capillary Pumped Loop Systems (CPL)



Mechanical Coolers / Thermoelectric Coolers (TECs)

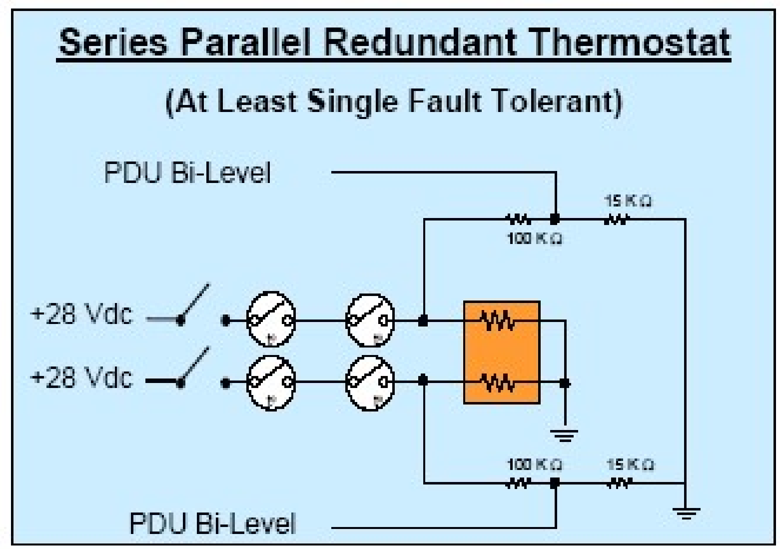
Active Example: Heaters

- **Kapton film heaters with internal resistance element are commonly used**
- **Typical bus voltage range of 27V – 35V**
  - Some applications may require more complex heater designs, e.g. propulsion lines
  - Lower voltages possible for survival modes
- **Operational heaters may employ electronic controllers providing on/off, proportional, integral, or differential control; redundancy not always required**
- **Survival heaters are usually controlled passively by mechanical thermostats**
  - These are redundant circuits that prevent a component from falling below its survival limit and causing permanent equipment damage

Kapton Heater

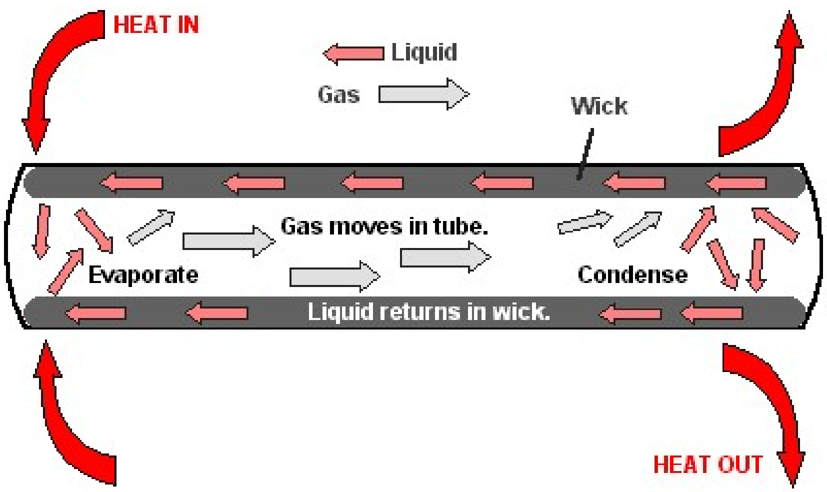
Dale-Ohm Heater

Cartridge Heater



Heat Transport: Two-Phase Systems

- **Efficient heat transfer – evaporation of the working fluid to absorb heat at the source and transfer it to cooler areas, ultimately the radiator, with small temperature drops**
- **Used often when structural heat paths do not have sufficient capacity to transport internal waste heat to radiator**
- **Ammonia (NH_3) is the typical working fluid, but others have been used (propylene, water, etc.)**
- **Types of heat pipes:**
  - **Constant Conductance Heat Pipes**
  - **Variable Conductance Heat Pipes**
  - **Loop Heat Pipes**
  - **Capillary Pump Loops**

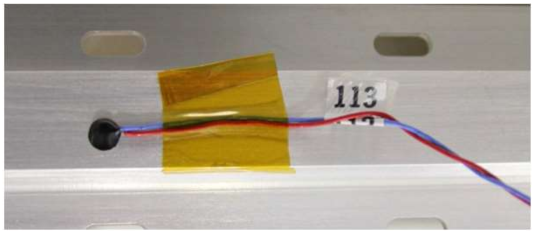


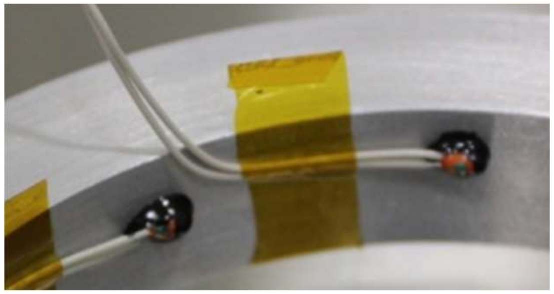
Related Fact

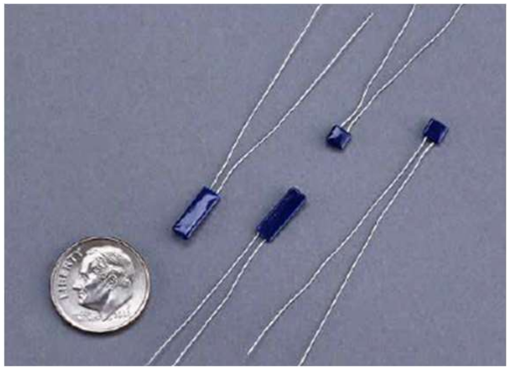
Hubble's Wide Field Camera 3 used eight Variable Conductance Heat Pipes to transport the heat generated from its CCD camera (black trapezoidal box) to its radiator (white curved surface)

Temperature Sensors

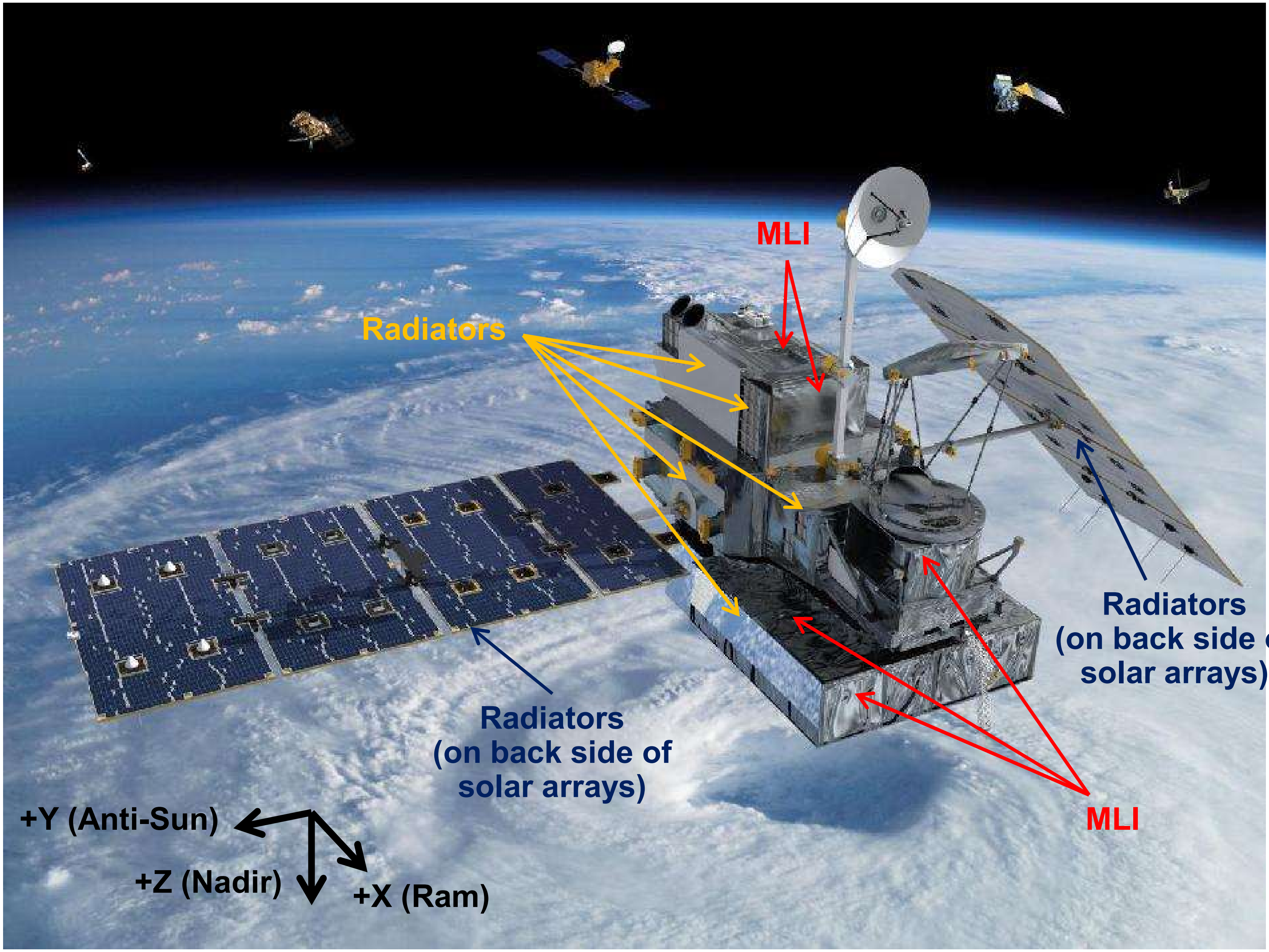
- **Most common:**
  - **Thermocouples:**
    - **Used for ground testing**
    - **Two dissimilar metal wires are joined to form a junction**
    - **When the junction gets warmer or colder a voltage is generated in the wires which corresponds to a temperature**
  - **Thermistors:**
    - **Often used in flight monitoring**
    - **Resistor that changes resistance with temperature**
  - **Platinum Resistance Thermometers (PRTs):**
    - **Used in flight temperature monitoring at high and low temperatures**
    - **Length of platinum wire that changes resistance with temperature**
- **Other temperature sensors:**
  - **Silicon Diodes**
  - **Cernox Sensors**
- **Lake Shore Cryotronics has a temperature selection guide online which is handy for choosing an appropriate temperature sensor**







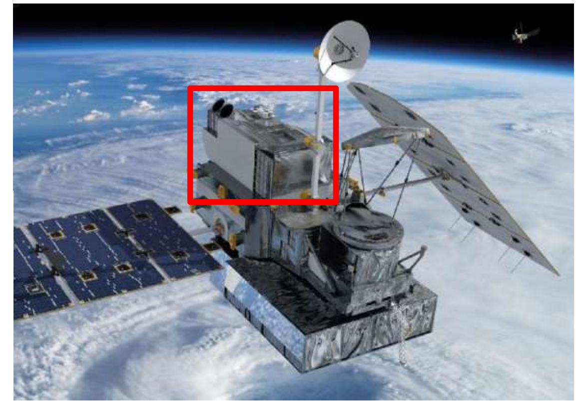
And where does this thermal hardware go?



And where does this thermal hardware go?

Thermistors (Temperature Sensors)

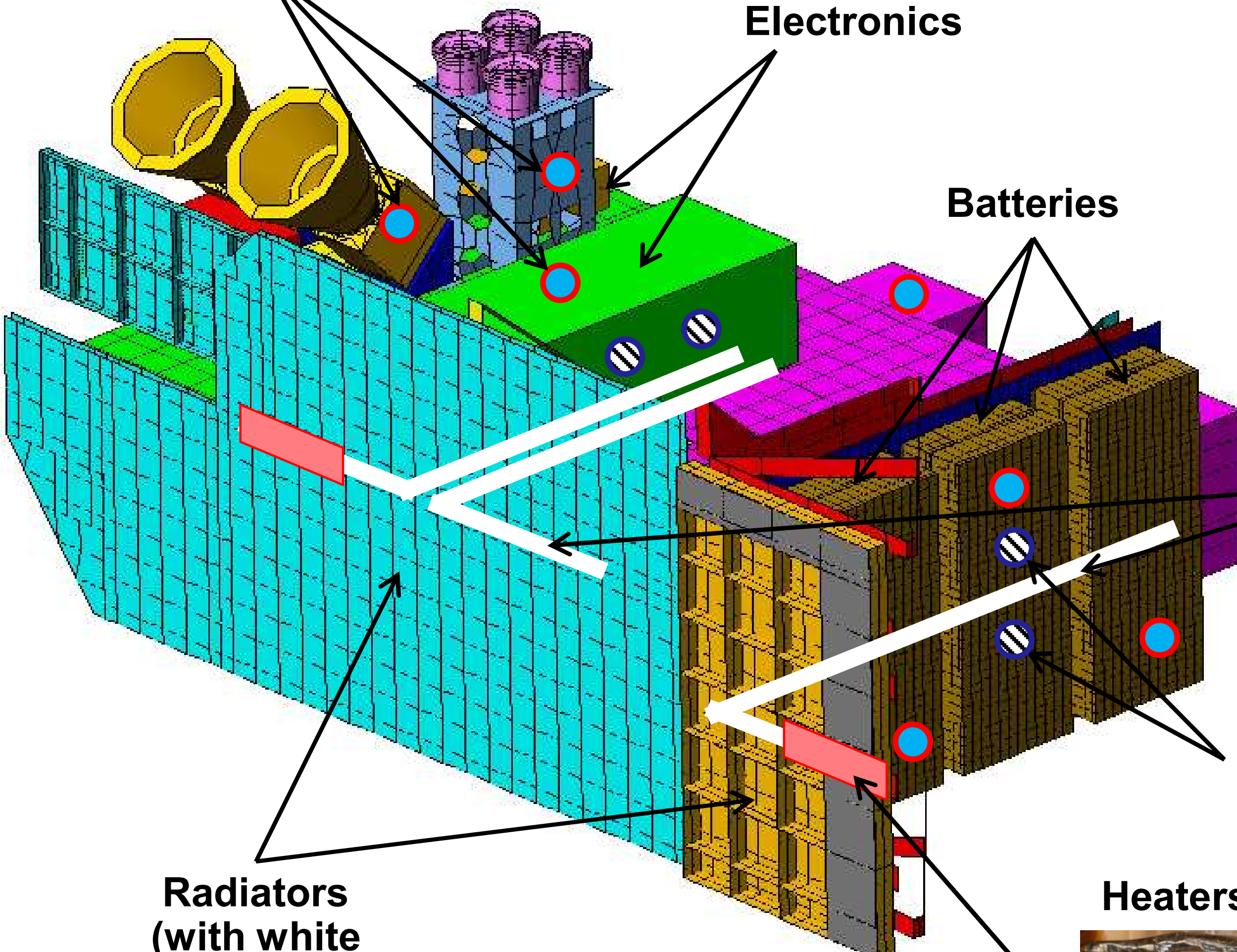
Note: most thermal components are embedded inside the structure



Relative Position

Electronics

Batteries

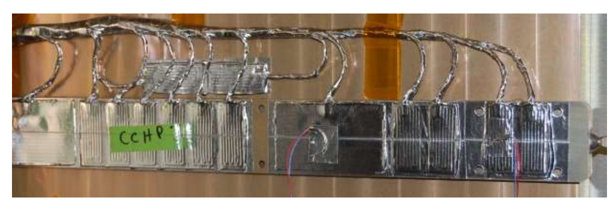


Heat Pipes

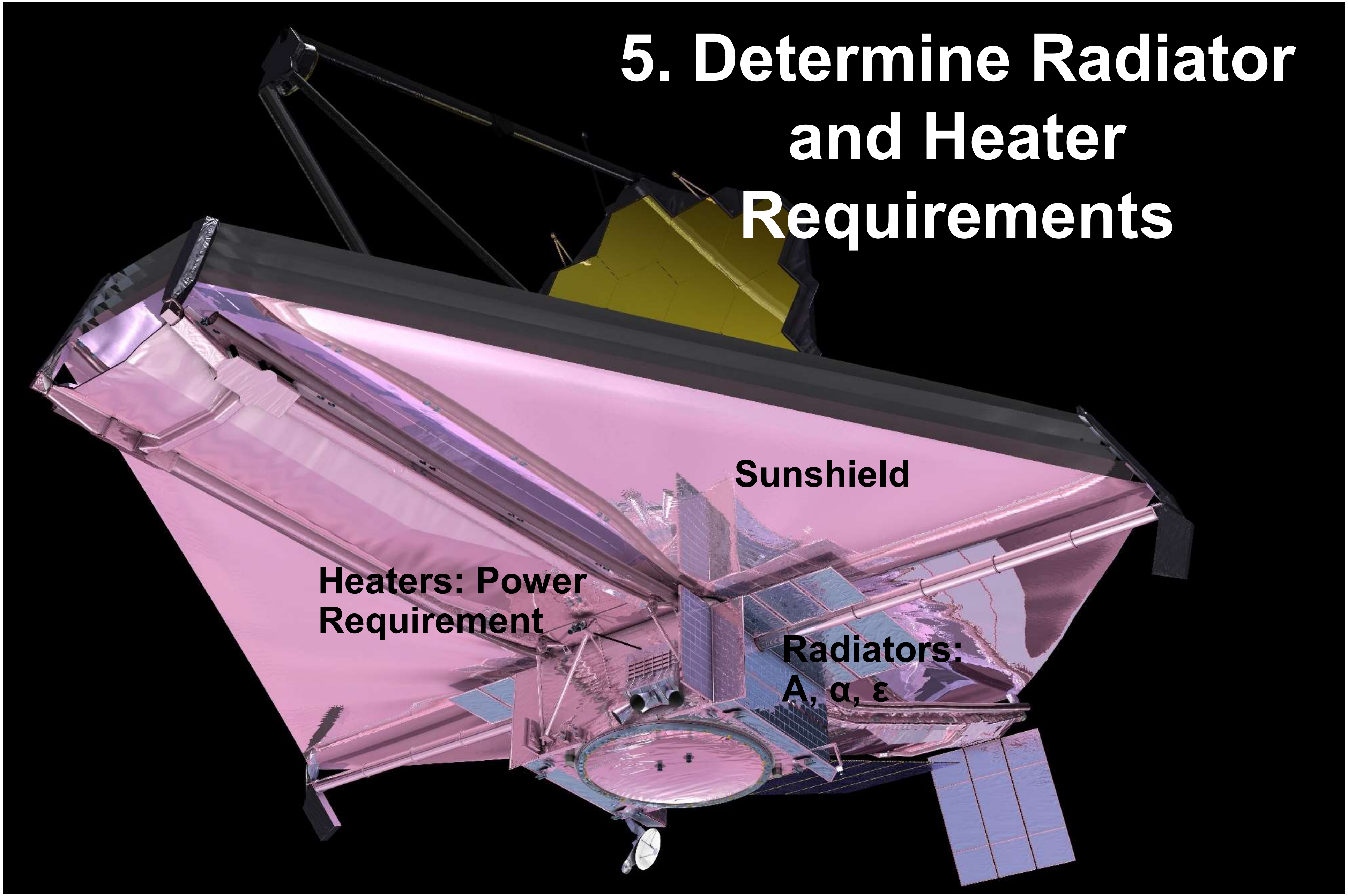
Mechanical (Bi-metallic) Thermostats

Radiators (with white paint coating)

Heaters



THERMAL DESIGN PROCESS



Worst Case Analyses for Spacecraft

Finally, we can use all of the information we just gained to size radiators and heaters for the worst-case hot and cold conditions on a spacecraft

- **It is impossible to model every surface, heat path, or environmental condition for a spacecraft, so there is an inherent "uncertainty" in the thermal analysis results for a mission**
  - We try to find to bounding thermal cases, assuming that if a spacecraft is designed for worst-case scenarios, all other scenarios during normal operations are accounted for
  - We must consider degradation of thermal control coatings over the course of a mission
  - Heat loads from most components also must be estimated in the design phase, then measured later using engineering development models / flight hardware
- **It is critical that worst case hot and cold parameters be identified *early* during the spacecraft design / development process and used to bound the range of possible permutations**
  - Typically, there are singular hot and cold cases, but depending on the on-orbit orientations, there may be many cases to bound the problem
  - Values for hot/cold cases will be refined during development

Bounding the Problem

- **Worst case stacked parameters**

	Hot case	Cold case
Solar Constant	High	Low
Albedo	Greater percentage of reflected sunlight	Lower percentage of reflected sunlight
Earth IR	High	Low
Radiator coating	End-of-life properties (higher α, lower ε)	Beginning-of-life properties (lower α, higher ε)
MLI Blanketing	Less effective emissivity on cold side	More effective emissivity on cold side
Power Dissipation	Maximum	Minimum

  – **Use heat loads w/contingency from electronics for hot case, best estimate (no contingency) for cold operational case**
  – **There must be a reduced operational configuration for survival mode (no data processing, no guidance (inertial only), only S-band comm., etc.) that should be assessed as well to make sure sufficient heater circuits and power is available**

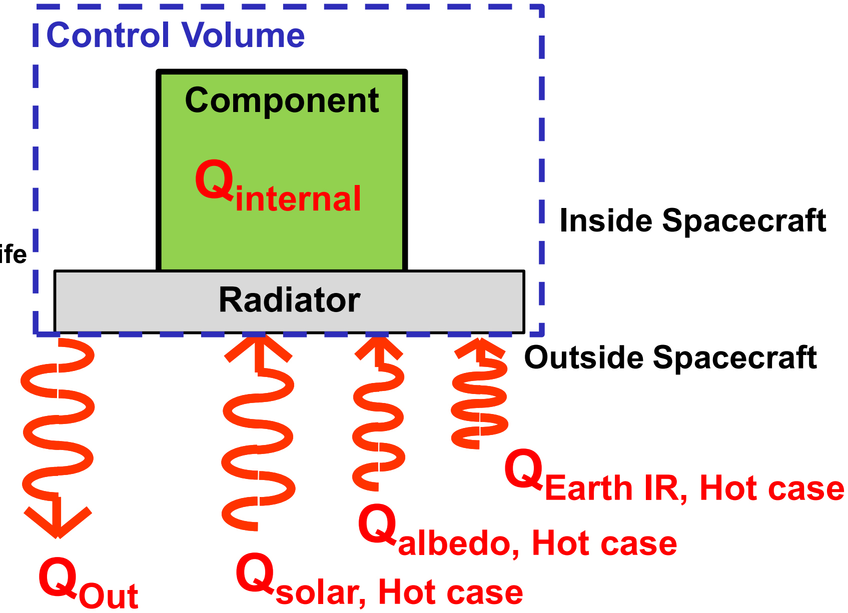
How do you calculate radiator size?

In LEO:

$\Sigma q_{internal,\ Max}$

* EOL = End-of-Life

α_{EOL} , ε_{EOL}



$Q_{in} = Q_{out}$

$$A_{P,\ Solar}\, Q_{Solar,\ Hot}\, \alpha_{EOL} + (ALB)\, A_{P,\ Albedo}\, Q_{Solar,\ Hot}\, \alpha_{EOL} + A_{P,\ Earth\ IR}\, Q_{Earth\ IR,\ Hot}\, \varepsilon_{EOL} + Q_{internal,\ Max} = \mathcal{F}\, \sigma\, A_{rad}\, \varepsilon_{EOL}\, (T_{rad}^4 - T_{space}^4)$$

$$A_{rad} \cos\beta_{Solar}\, Q_{Solar,\ Hot}\, \alpha_{EOL} + (\approx 0.35)\, A_{rad} \cos\beta_{Albedo}\, Q_{Solar,\ Hot}\, \alpha_{EOL} + A_{P,\ Earth\ IR}\, Q_{Earth\ IR,\ Hot}\, \varepsilon_{EOL} + Q_{internal,\ Max} = \mathcal{F}\, \sigma\, A_{rad}\, \varepsilon_{EOL}\, (\underbrace{T_{rad}^4} - T_{space}^4)$$

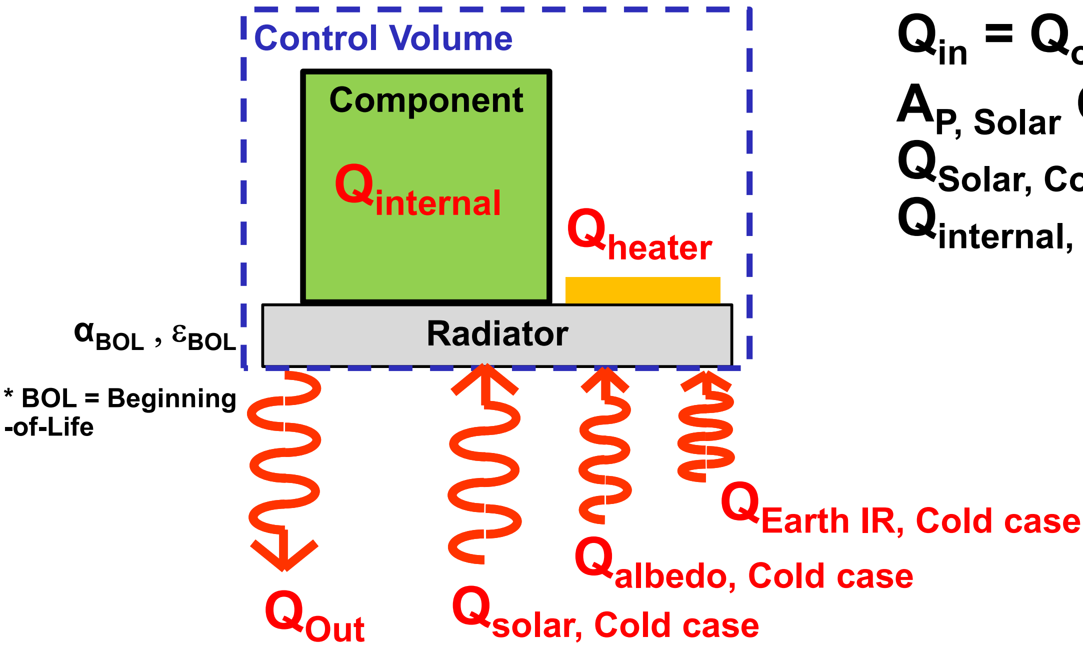
Solve for A_{rad}

Very often determined by temperature limits of internal components and their conduction path to the radiator; in this case, you would want to use the highest acceptable temperature limit for your radiator

- Would we want high or low (α/ε) coatings for external surface of radiators?
- Would we typically want high or low ε coatings internally?
- Why do we typically avoid low ε coatings externally?

How do you size heaters?

- Using the radiator area sized in the Hot Biased Environment, size your heater for the Cold Biased Environment



$$Q_{in} = Q_{out}$$

$$A_{P,\,Solar}\, Q_{Solar,\,Cold}\, \alpha_{BOL} + (ALB)\, A_{P,\,Albedo}\, Q_{Solar,\,Cold}\, \alpha_{BOL} + A_{P,\,Earth\ IR}\, Q_{Earth\ IR,\,Cold}\, \varepsilon_{BOL} + Q_{internal,\,Min} + Q_{Heater} =$$

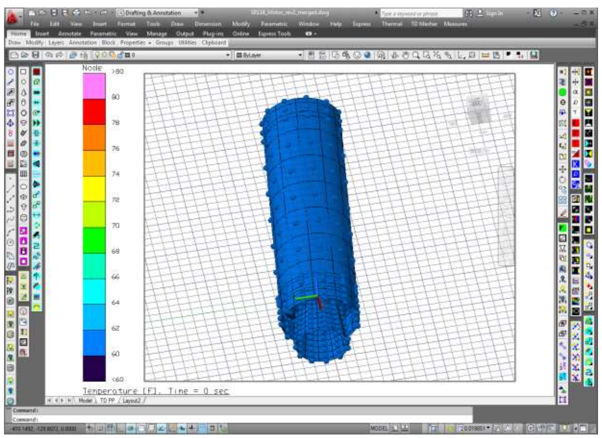
$$\mathcal{F}\, \sigma\, A_{rad}\, \varepsilon_{BOL}\, (T_{min,\,required}^{4} - T_{space}^{4})$$

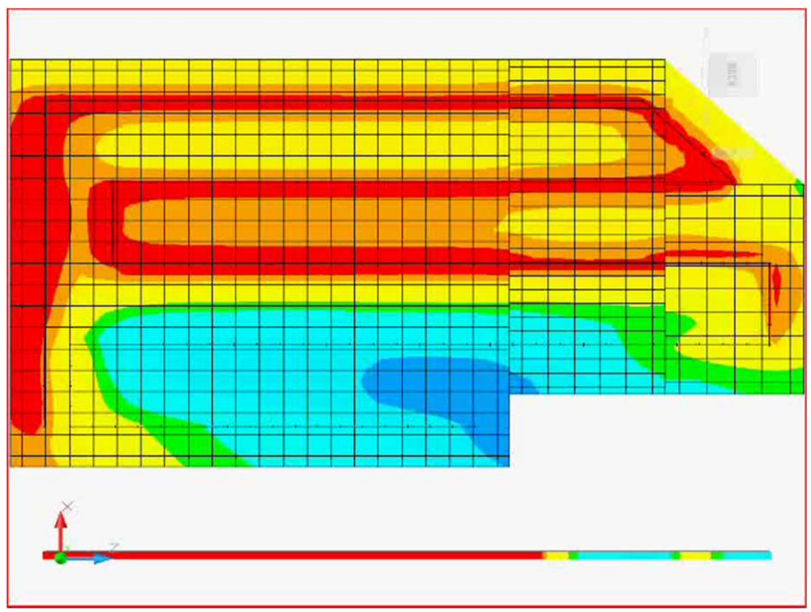
If $Q_{in} < Q_{out}$, then Q_{heater} is the amount of heater power required to balance the equation

- Consider redundant circuits (survival circuits) for crucial components, such as the battery, instrument, or electronics boxes

Thermal Models

- Due to the large-scale complexity of instruments and spacecraft, thermal engineers usually employ thermal models to determine resultant temperatures and heat flows on a spacecraft
- Radiation Exchange (GMM): specialized programs can calculate radiation "couplings" for all spacecraft surfaces using ray trace algorithms
  - Typically 500-10,000 surfaces in geometry model, generating 10,000 – 1,000,000's of couplings
  - Also used to calculate solar, planetary heat loads on external surfaces, including eclipses
  - Can simulate "moving" surfaces through the orbit (solar arrays, deployables, etc.) or change S/C attitudes (yaw maneuvers, etc.)
  - Post-process temperature plots, maps, colorized pictures, etc.

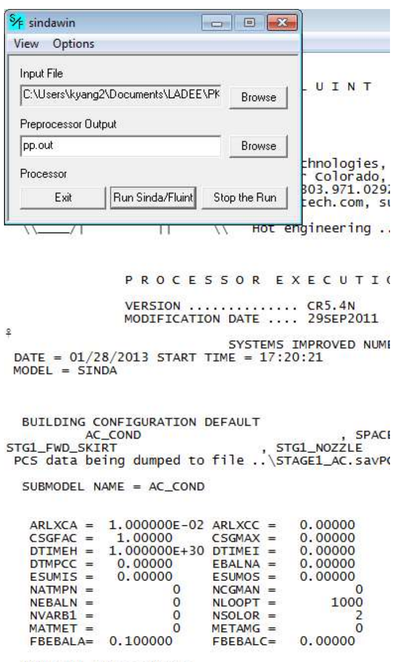


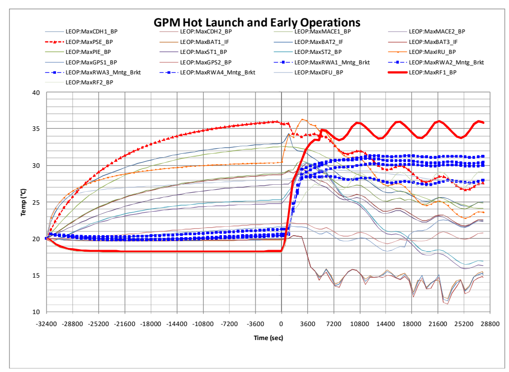


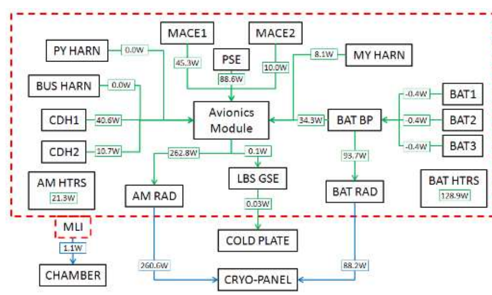
Thermal Desktop/ RADCAD is a popular AutoCAD-based Geometric Math Modeling (GMM) tool used to calculate radiation couplings and other inputs needed for a thermal solver

Thermal Models

- **Thermal Analyzers (TMM): define nodal network with conductive (and convective, if applicable) couplings, import radiation couplings, define heat loads from electronics, heater routines, etc.**
  - **Calculates temperatures, heater power using steady state and transient solution capabilities, goal seeking capabilities, optimization capabilities (multiple variables), time and temperature-varying properties, user-determined solution techniques, solution sequences, accuracy levels, and outputs spreadsheet-like expressions and user variables**





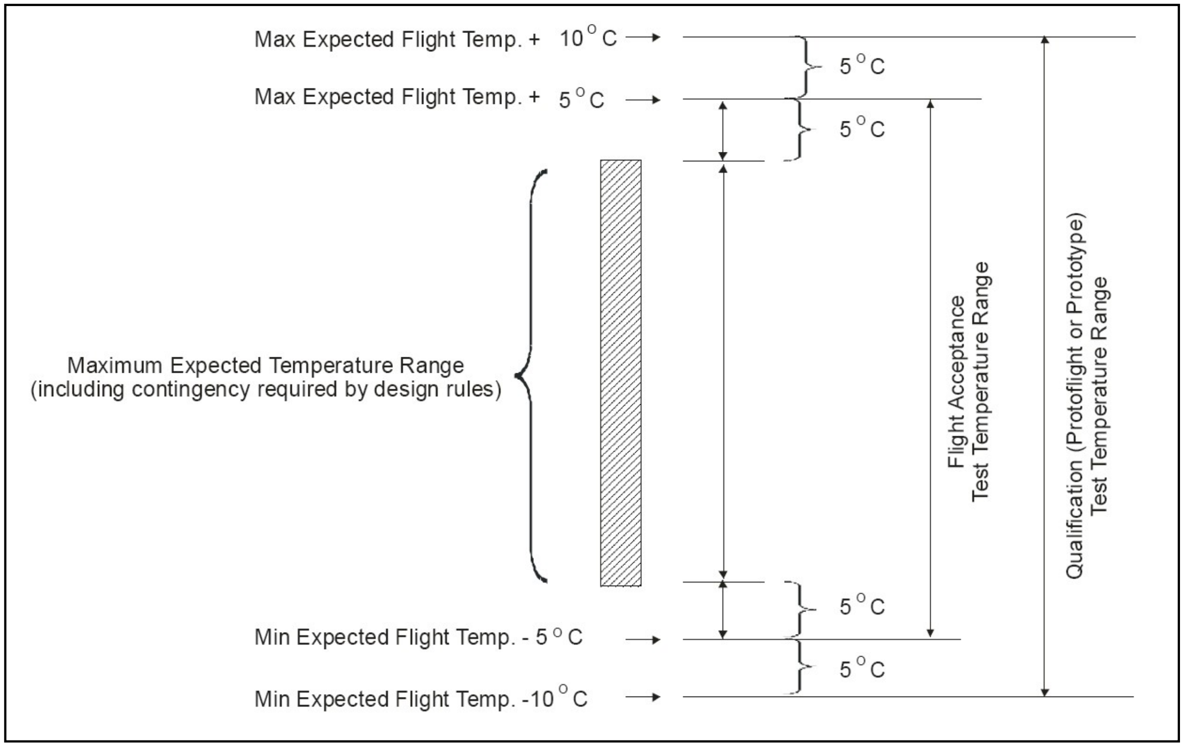


SINDA/FLUINT is a popular thermal math model (TMM) analyzer/ solver which outputs temperatures, fluxes, capacitances, and other data which can be visualized with a post-processor

THERMAL TESTING

How do you verify that they work as intended?

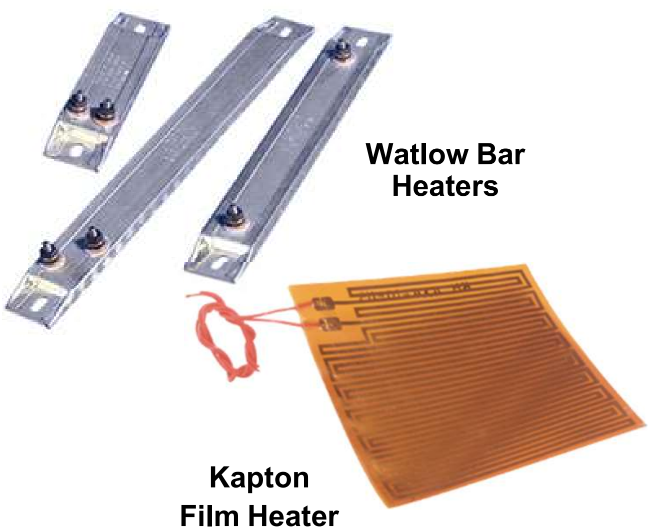
- **Thermal testing is done at *margined* temperature extremes and is designed to verify workmanship, demonstrate performance, and collect data to be used in correlating thermal models**
  ➔ We want to verify thermal model and ensure satellite can meet performance requirements at temperatures beyond the allowable flight temperatures
- **Margins: Qualification: 10°C margin (hot and cold)**
  Acceptance: 5°C
- **All testing is done in a vacuum chamber. Exceptions can be made at unit level, but never at instrument or S/C level**



Typical Thermal Test Ground Support Equipment

Cryopanels

- Aluminum plate with tubing on back side for GN_2 or LN_2 flow at desired temperature
- Isolated from support structure using thermal isolators
- Coated on side facing spacecraft
- Used when component heat flows to environment are large and fast ramp rates are required for heating and cooling

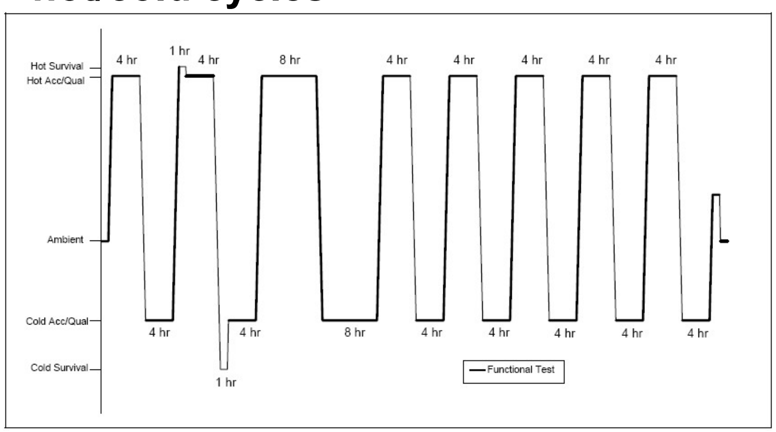


Heater Panels

- Aluminum plate with bar or film heaters attached to achieve desired temperature
- Coated on both sides and isolated from support structure
- Cooling is achieved solely through radiation to the test chamber (typically at GN_2 or LN_2)
- Used when component heat flows to environment are relatively minimal and a slower cooling rate is acceptable

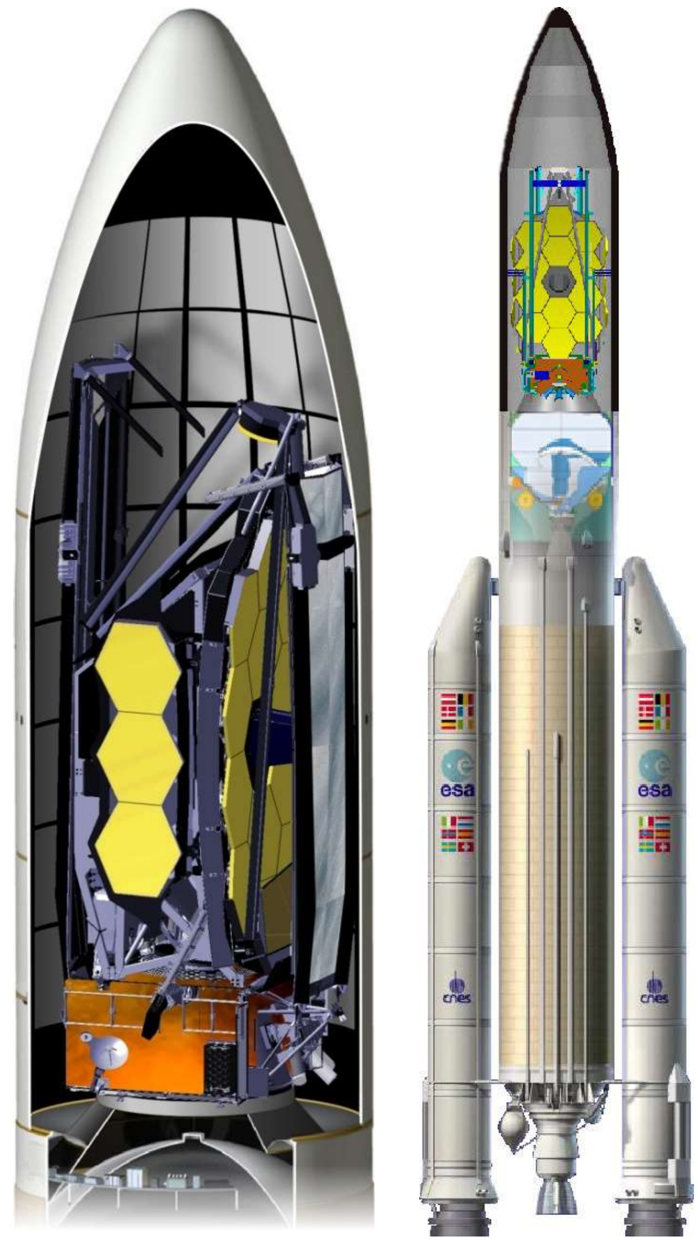
Thermal Balance / Thermal Vacuum Testing

- **Thermal balance plateaus**
  - Thermal environment is set, and spacecraft must achieve energy balance with environment. Balance criteria met from achieving temp. rate-of-change requirement on components
  - Thermal data collected is used to verify predictive accuracy of thermal models
- **Thermal vacuum cycles**
  - Quality assurance test to take hardware beyond its operational temperatures and ensure it will survive temperature extremes: used to verify workmanship on components
  - NASA Goddard typically requires 12 hot/cold cycles



James Webb Space Telescope OTIS CV Test

6. Estimate TCS Mass and Power



Thermal Hardware Mass and Power Estimates

	Mass	Power	Comments
Multi-Layer Insulation	0.73 kg/m^2	0 W	Based on 15 layers
Kapton Heaters	0.36 kg/m^2	Various, based on heater power requirements	Based on 10-mil thick Kapton heaters
Thermostats	6 grams each	0 W	
Thermistors / Thermal Sensors	1-3 grams each	~0 W	
Heat Pipes (Ammonia)	0.15 kg/m	0 W for Constant Conductance Heat Pipes ~10 W for Variable Conductance Heat Pipe (VCHP) Control	Mass per unit length Add 1-3 kg each for VCHP reservoirs
Loop Heat Pipe Evaporator	2-5 kg	10-30 W Control Power	
Radiator Panels	3.3 kg/m^2	0 W	Mass based on Aluminum Honeycomb radiator Add heat pipe mass if embedded
Electronic Controllers	0.2 kg	1-3 W each	

Sources: Space Mission Analysis and Design, 3rd Edition; NASA GSFC Thermal Engineering Branch Standards

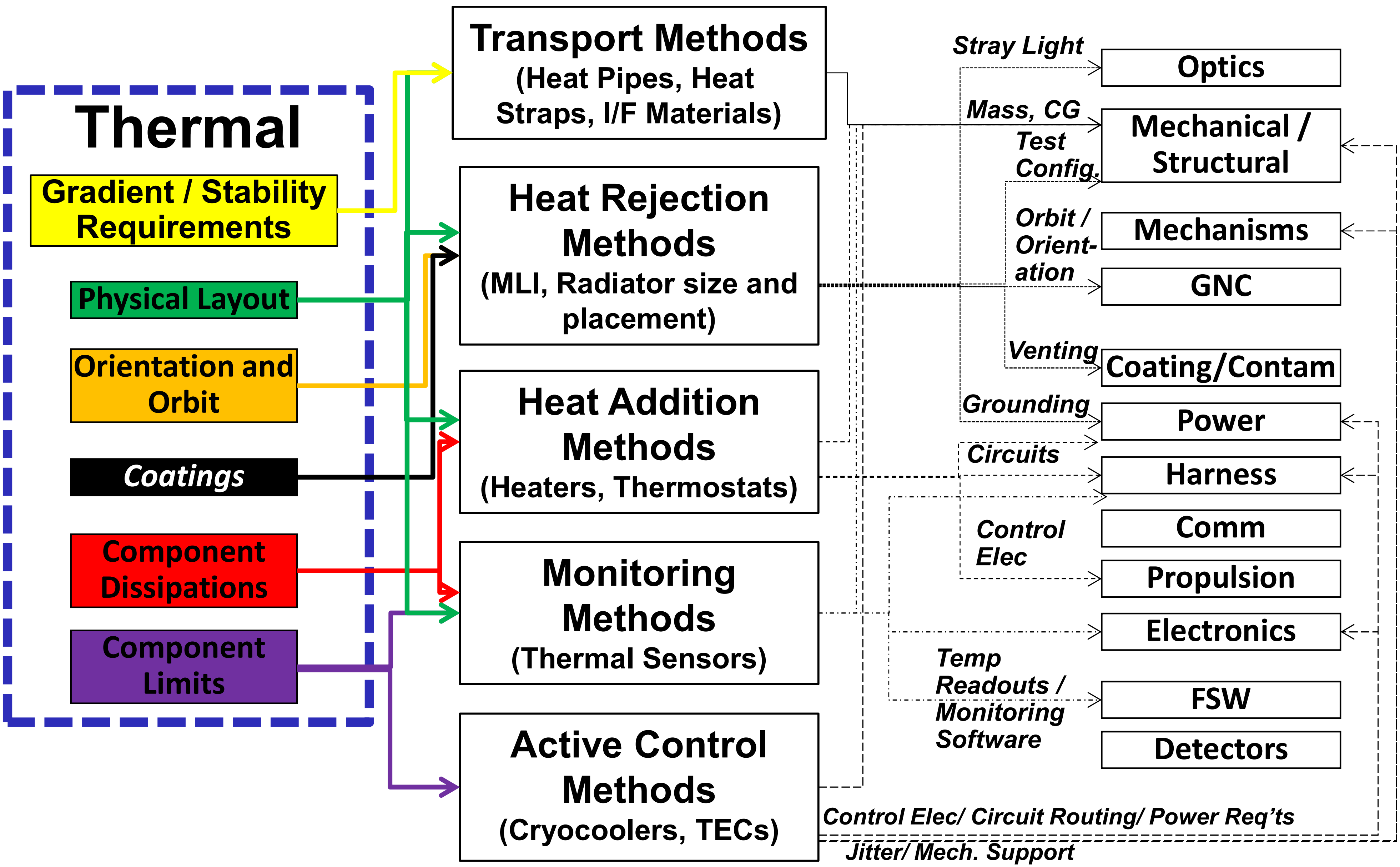
Notes about Thermal Mass and Power Estimates

- **Historically, thermal subsystem mass is about 2-10% of spacecraft/instrument dry mass**
  - **Purely passive thermal design closer to 2% range**
  - **Active thermal design closer to 10% range**
  - **Bulk of mass is MLI and any specialized thermal components**

- **Thermal control power estimates normally only consist of heater power, unless electronic controllers are used**
- **Power estimates are divided in two groups:**
  - **Operational:** use coldest operational scenario with mechanical thermostats and electronic controllers with PID control (sometimes handled through flight software)
  - **Survival:** use coldest survival scenario. Circuits should be controlled in most reliable manner, usually with mechanical thermostats and not flight software; only circuits with critical component limits have survival heating applied

- **However, Thermal does not just impact mass and power, but all spacecraft subsystems (shown on the following slide)**
  - **The thermal design must be iterated with other subsystems until a satisfactory compromise is achieved**

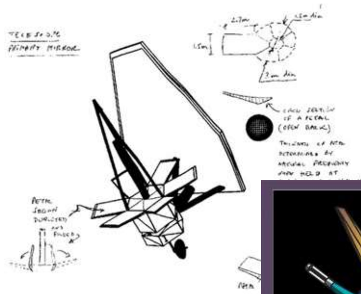
Thermal's Impact on Other Subsystems

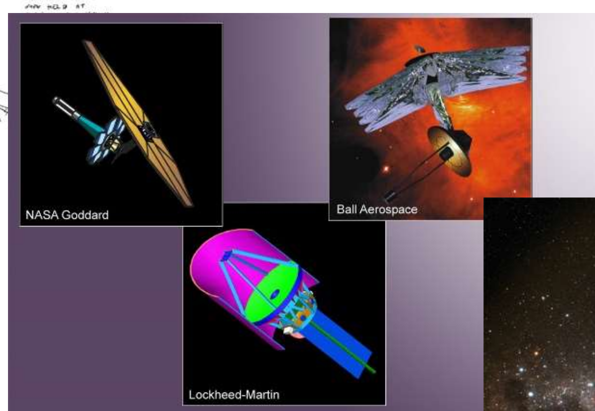


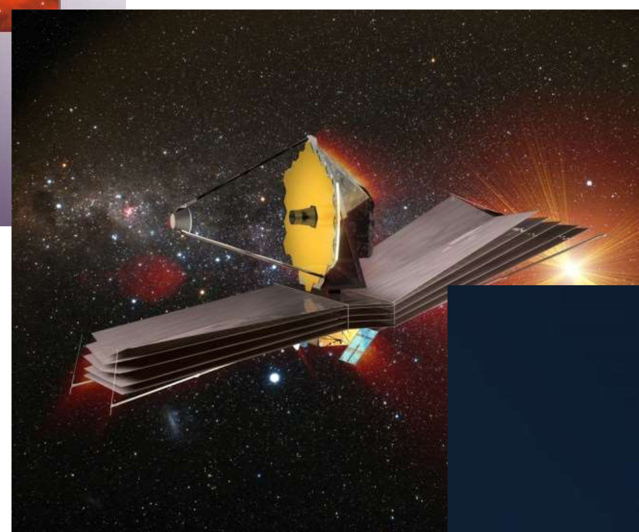
Considerations for an effective design process

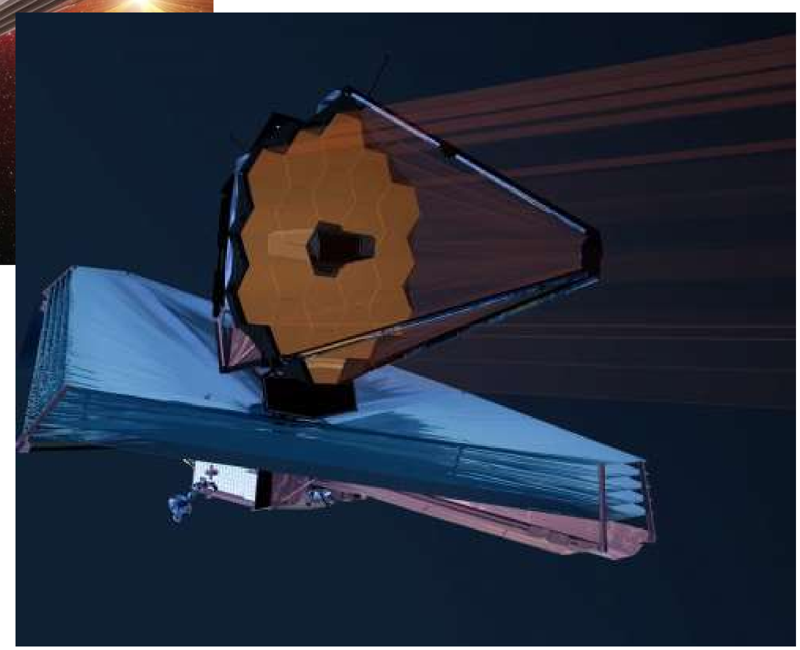
- **Technical exchanges required/ parameters to be reiterated with input from other subsystems throughout the course of the study:**
  - **Radiator size/location/material/coating:** dependent upon orbit (GNC), available space (structures)
  - **Heater power / size:** dependent on component temperature constraints (all subsystems), available power (electrical/power)
  - **Component dissipations (electrical/power)**
  - **Need for thermal transport systems:** dependent on component temperature/gradient/rate constraints (all subsystems), placement and available mass (mechanical/structural)
  - **Need for isolation between components/from structure:** dependent on component temperature/gradient/rate constraints (all subsystems)

7. Document and Iterate









JWST Design Evolution

Recap: Factors Affecting Thermal Control

- **External Thermal Environment:**
  - Our sun is the biggest source of external heating (for the most part); but other sources are substantial as well, such as albedo and Earth IR
  - Missions away from Earth must consider the varying solar irradiance as you travel away from our planet. This also affects power generation
  - For missions orbiting a planet/moon, the spacecraft can cycle through from full sun to shadow (eclipse) many times in one day, accumulating tens of thousands of cycles over the mission lifetime

- **Component Requirements**
  - Heat loads / internal power: heat loads from various components typically used on satellites varies greatly
  - Allowable Temperatures: Typically, most electronics must be kept “near ambient” temperature, or roughly -10° to +50°C, although there are several exceptions
  - Science instruments typically have extremely challenging temperature, stability, and gradient requirements

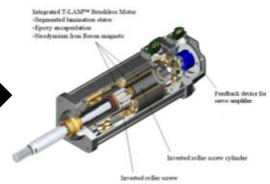
- **Your final thermal design must take all of these factors into consideration to meet thermal limits & constraints while minimizing mass and power requirements**
  - If a particular thermal design works for 99.9% of the mission lifetime but fails during 0.1%, then it is a failed thermal design
  - Thermal design needs to be built for margined extremes in environment and on-orbit operations

The Thermal Design Process: Takeaways

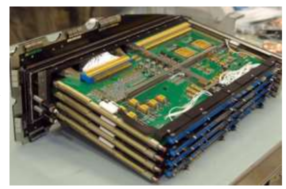
Since thermal design is integral with the entire observatory and affects all subsystems…

➔ Thermal must be involved in all levels of design…

Circuit Boards



Mechanisms



Electronics Boxes

Sub-Assemblies (like instruments)

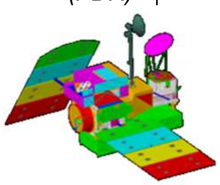
Satellite / Observatory

… and all phases of design

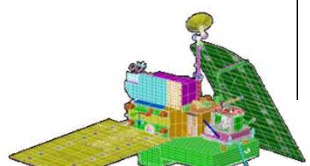
Initial Design	Final Design / Component Fab	System Assembly, Integration & Testing (I&T), Launch	Mission Ops

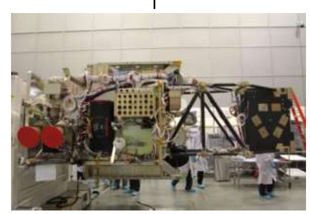
Mission Concept Review (MCR)

Preliminary Design Review (PDR)



Critical Design Review (CDR)





Hardware integration

Pre-Environmental Review (PER)

Environmental Testing (Vibrational, Acoustic, EMI, Thermal Vacuum)

Pre-Ship Review (PSR)

Launch & Mission Ops

The greater extent that Thermal is involved in the spacecraft design, the greater the chance for all spacecraft components to be thermally safe in flight ➔ Mission Success!

Class Project

1. Determine the orbit, orientation, external and internal geometry (including placement of spacecraft components), and temperature limits of your satellite.

2. You are welcome to give your orbit, orientation, and external geometry to me to run through an orbital ray trace / heat rate program and give you fluxes for each side of your spacecraft at the hottest and coldest points in the orbit. Otherwise, you are welcome to calculate bounding heat fluxes on your own as well.

3. Using these heat fluxes, choose coatings/insulation for each side of your spacecraft and radiator placement. Determine the hottest and coldest temperatures that your spacecraft components will see given their current placement. Change your coatings/insulation and internal placements as necessary to prevent limit violations. Try to work out a passive solution.

4. If a passive solution does not work, determine thermal problem areas that need to be resolved with an active solution. Determine any active cooling, transport, or heating needs as necessary.

5. Iterate your design until it satisfies all of your temperature limits for the worst hot and cold cases. Please feel free to contact me at any time if you need guidance or additional information.

Your Project Report

- **Report outline:**
  - **7.0** Thermal
  - **7.1** Driving requirements and assumptions
  - **7.2** Options considered
  - **7.3** Thermal Analysis
  - **7.4** Thermal system baseline design and rationale

- **Understand your thermal environment**
  - **Orbit:** Altitude? Sun synchronous? Beta angle range?
  - Get orbit information from GNC

- **Summarize Unit Requirements**
  - Unit heat loads: Current Best Estimate (CBE) for cold and hot cases
  - Allowable temperatures (operational/survival) for all components using info from this lecture, or subsystem specific info from other lectures, or another reputable source
  - Determine how components are arranged internally; they can't be all on the same panel

What Do You Need for Your Project?

- **Energy balance for each radiator panel:**
  - Use environments you've determined
  - Any IR backloads from major appendages (e.g. solar arrays), using view factors that you can calculate
  - Calculate size (area) using "stacked" parameters in hot case, keeping 5°C uncertainty margin from max allowable
  - Calculate temperatures for "stacked" cold case parameters. If heater power is needed to maintain minimum temperatures in cold case (or hot case), show those values too (must be included in power budget as well!)

- **List thermal control materials & hardware needed for your design, including heaters, thermostats, MLI, etc. used for mass estimates. If you are moving heat from one side of the S/C to another: how are you doing that? What radiator coating(s) are you using?**

- **Summarize all units (Watts, Kelvin or Celsius, meters) and their allowable temperatures along with hot and cold case calculated temperatures and any heater power needed. You could group per panel, if needed.**

Recap for Class Project

- The top things I want to see are:
  - Radiator Size (based on stacked worst hot case)
  - Heater powers/placement (based on stacked worst cold case)
  - Where are your components in your satellite? What side are the radiators placed?
  - How do these components transfer their heat to the radiators? What conductive heat paths do they take?
  - What are the final worst-case hot and cold temperatures for each satellite component? Do they meet your component allowable temperature ranges?

References

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- "General Environmental Verification Specification (GEVS) for STS and ELV Payloads, Subsystems, and Components", GSFC-STD-7000, April 2005
  - http://arioch.gsfc.nasa.gov/302/gevs-se/toc.htm
- "Test Requirements for Launch, Upper-Stage, and Space Vehicles", MIL-STD-1540

Acknowledgements

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Charlie Wildermann for organizing the ENAE 691 Satellite Design Course

Thank You

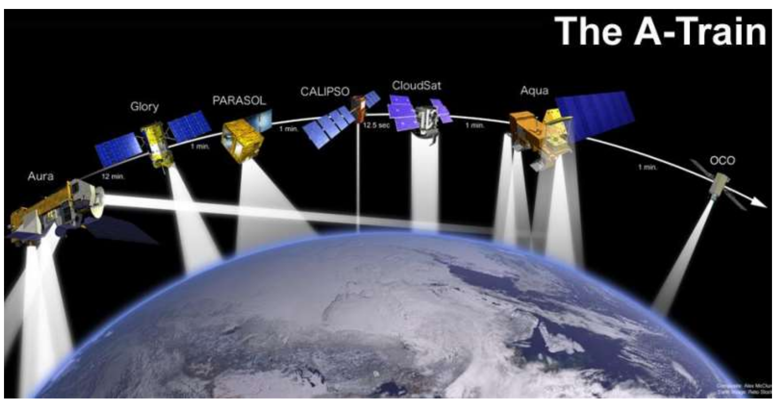
Questions?

Backup Slides

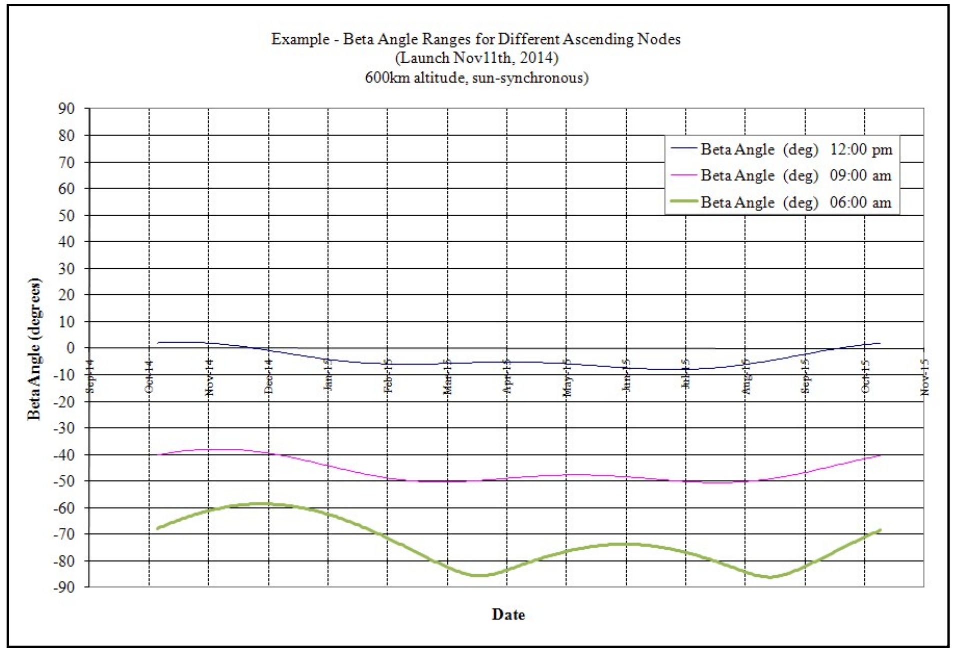
Beta Angles - Examples

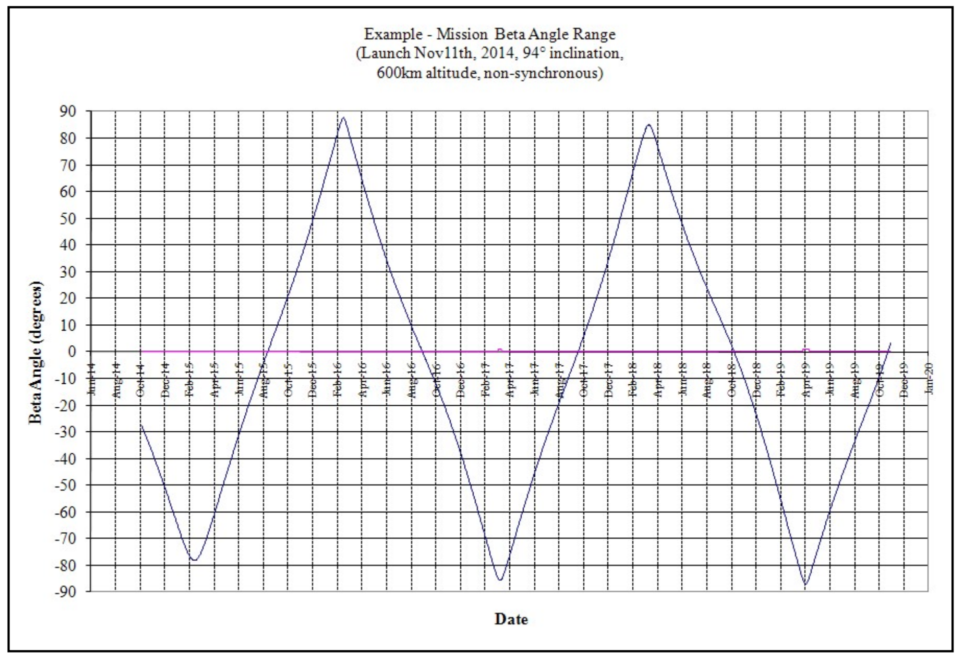
- For sun-synchronous orbits, the beta angle is fairly constrained
- Typical launch window is “centered” on the nominal time, so this can widen the range of beta angles
- Non-synchronous missions pretty much cover the full range of beta angles (depending on inclination): -90° < β < +90°

Related Fact



Most Earth Observation satellites, such as the those in the A-Train Constellation, are in sun-synchronous polar orbits

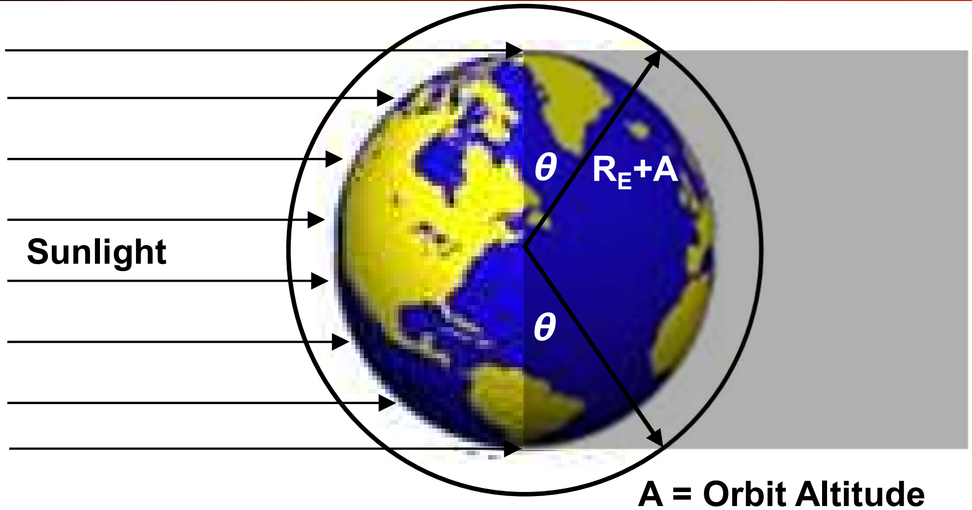




Things to Consider

- **Many sun-synchronous earth observing missions:**
  - have “mid-morning” equatorial crossing
  - have limited range of beta angles
  - keep nadir surface pointed to the earth
  - have one solar wing; obviously on “sun-side” of satellite
  - have multiple possible radiator locations
    - “anti-sun” side is best radiator, but can’t put everything here
    - “anti-earth” side can be used as radiator (large solar load near ascending node)
    - Other two sides (“ram” and “wake”) can be used; also have large solar load near poles
- **Most changes in design will have an impact on Thermal ➔ Thermal must consistently work with other subsystems to establish robust design**
- **Radiator sizes / heater powers are a byproduct of physics and the input powers/limits provided to thermal**
- **Thermal designs are driven by requirements provided to Thermal and not by Thermal itself**
- **Thermal provides design details, and <u>bounding case</u> temperature predictions and heater power estimates**

Additional Orbital Information: Eclipses



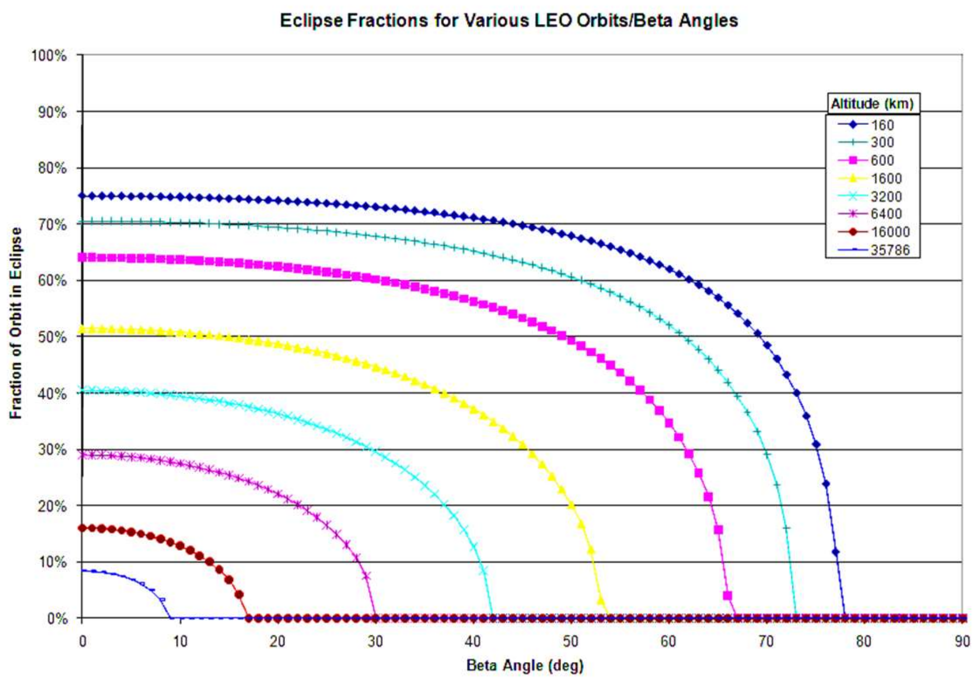
A = Orbit Altitude

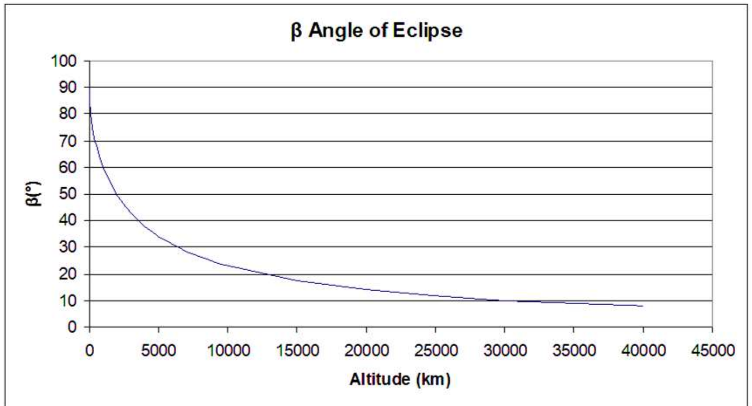
R_E = radius of the Earth

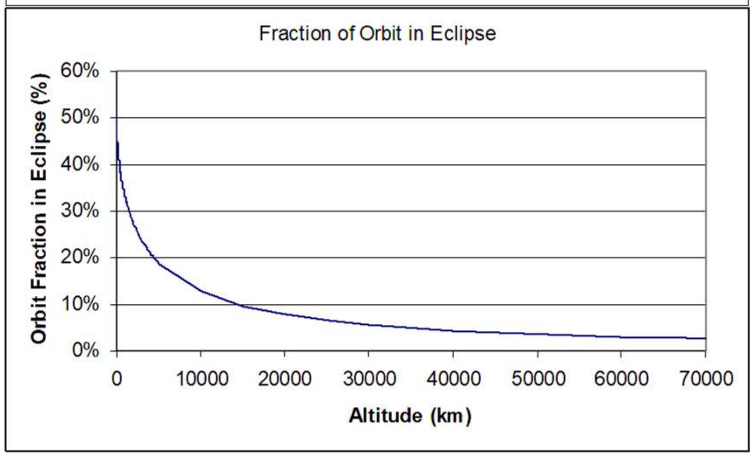
$$\theta = \cos^{-1}\frac{R_E}{R_E + A}$$

$$\text{Fraction of time in sunlight} = \frac{180 + 2\theta}{360}$$

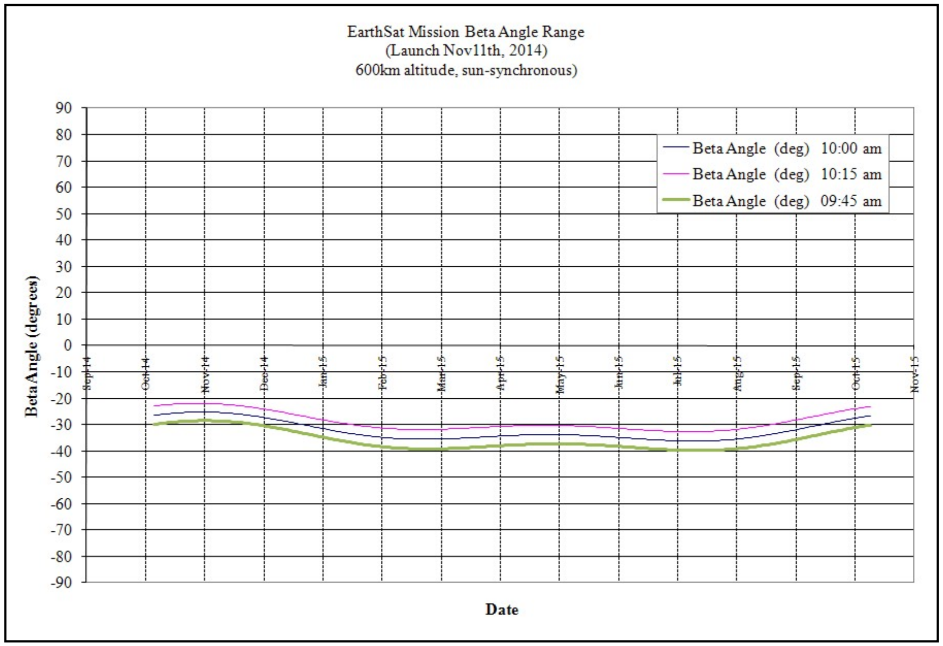
$$\text{Fraction of time in eclipse} = \frac{180 - 2\theta}{360}$$

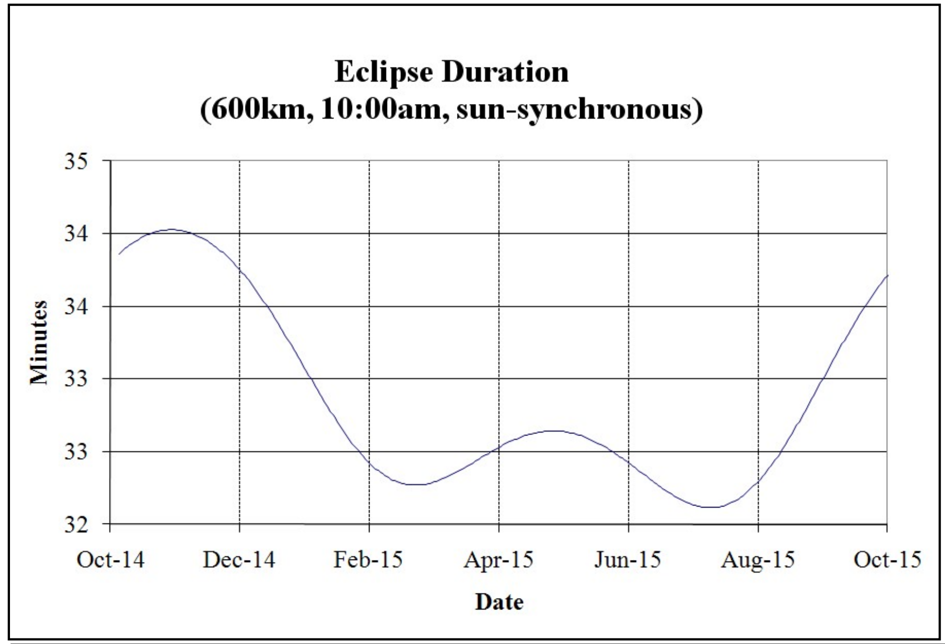


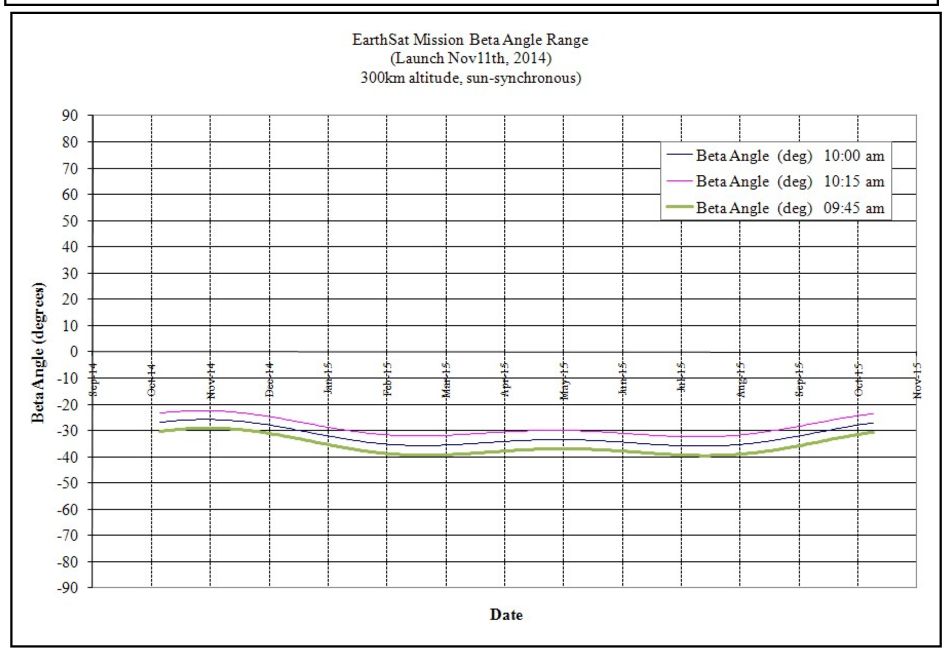


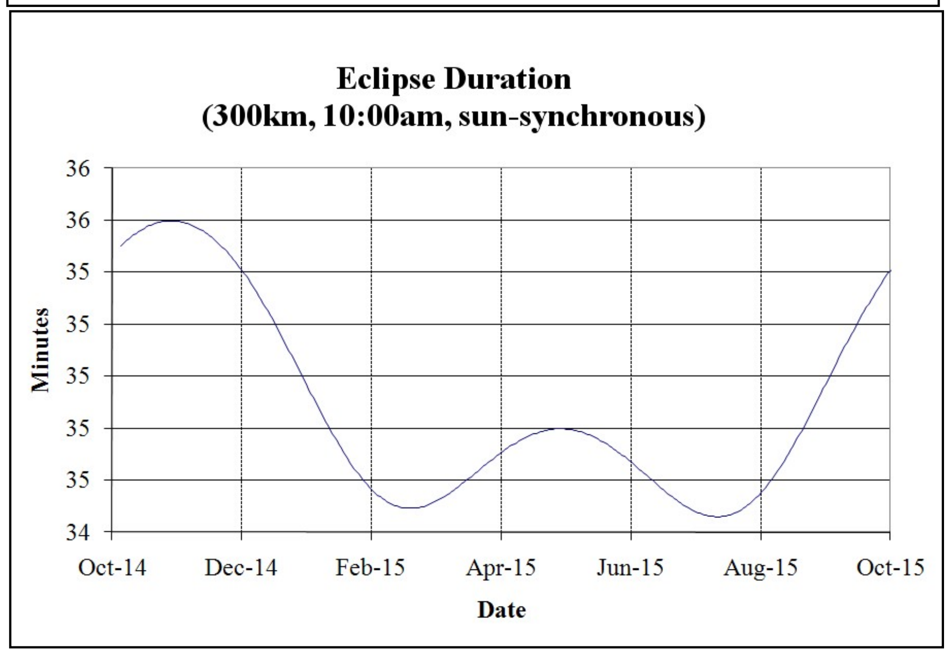


Additional Orbital Information: Beta Angle and Eclipse Examples



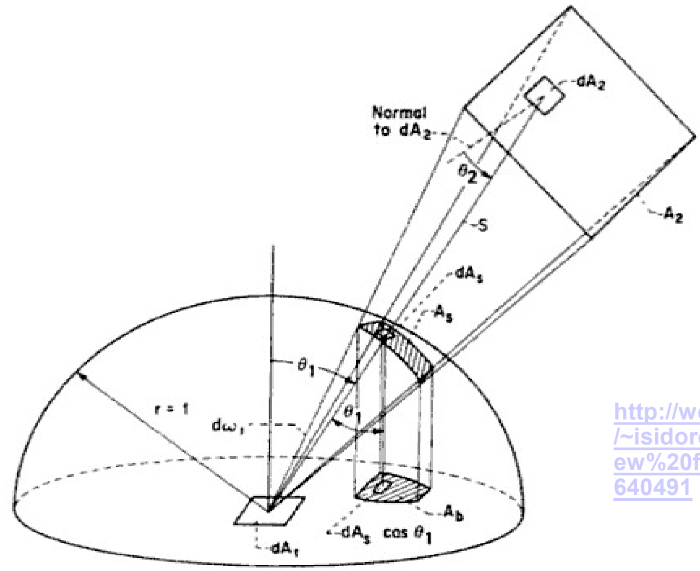






Radiation Heat Transfer –View Factor

- For those of you that love doing integrals, you can derive formulas for radiation view factors for some simple geometries.
- Some of these calculations have been tabulated in several references on heat transfer (e.g. Holman, 1986) or the NACA handbook. They range from ~zero (e.g. two small bodies spaced very far apart) to 1 (e.g. one body is enclosed by the other)
- These calculations are possible for simple geometries, but the typical satellite has many surfaces that interact “radiatively”
- Thermal software platforms are used to generate radiation views to space and between the many satellite surfaces. A typical satellite geometry model has 500-2000 surfaces with calculated radiation view factors numbering in the 10’s or 100’s of thousands



http://webserver.dmt.upm.es/~isidoro/tc3/Radiation%20View%20factors.htm#_Toc271640491

Related Fact

View factors for slightly more intricate geometries and orientations can become complex very quickly: shown below is the view factor from a rectangle to another rectangle in a parallel plane

$$F_{1-2} = \frac{1}{(x_2 - x_1)(y_2 - y_1)} \sum_{l=1}^{2}\sum_{k=1}^{2}\sum_{j=1}^{2}\sum_{i=1}^{2} (-1)^{(i+j+k+l)} G(x_i, y_j, \eta_k, \xi_l)$$

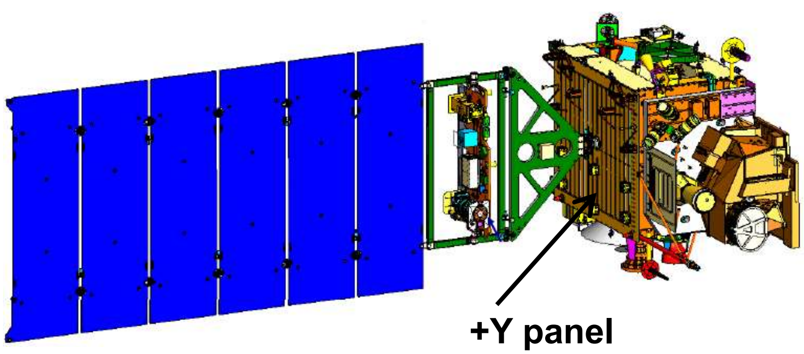
$$G = \frac{1}{2\pi}\left\{ \begin{array}{l} (y-\eta)\left[(x-\xi)^2 + z^2\right]^{1/2} \tan^{-1}\left\{\dfrac{y-\eta}{\left[(x-\xi)^2+z^2\right]^{1/2}}\right\} \\ +(x-\xi)\left[(y-\eta)^2+z^2\right]^{1/2} \tan^{-1}\left\{\dfrac{x-\xi}{\left[(y-\eta)^2+z^2\right]^{1/2}}\right\} \\ -\dfrac{z^2}{2}\ln\left[(x-\xi)^2+(y-\eta)^2+z^2\right] \end{array} \right\}$$

http://www.engr.uky.edu/rtl/Catalog/index.html

Radiation Heat Transfer –View Factor

- A flat surface has a total possible "hemispherical" view, with surface area $2\pi(r^2)$.
  - A steradian is defined as a "square radian" of area, so a perfect view is a "2π steradian" or F=1.0 view
- If you think of the flat surface as a radiator looking to space, anything that is within its "hemispherical" view to space reduces its view factor to space, and therefore its ability to radiate heat

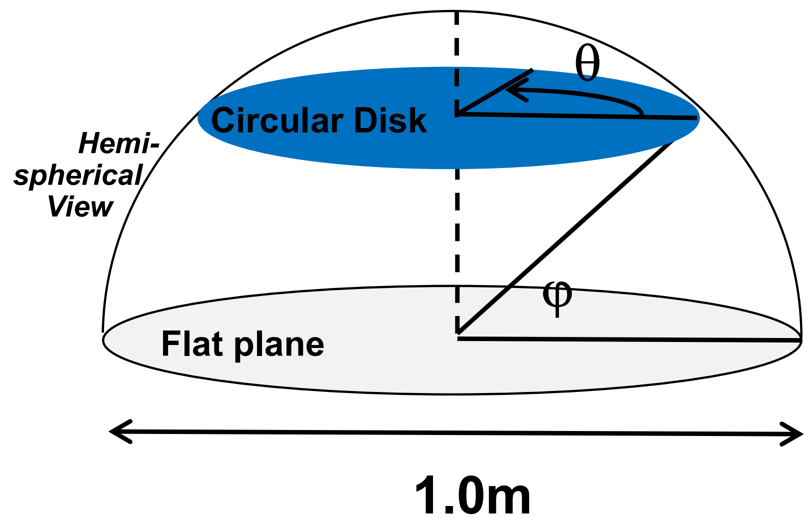
Related Fact



The solar array wing on the GOES-R (weather) satellites is in the view of the +Y panel radiator

- Example: View from a plane to a circular disk

$\varphi = 45° = \pi/4$



$$Area = \int_0^{2\pi} \int_{\pi/4}^{\pi/2} r^2 \sin\varphi \, d\varphi \, d\theta$$

$$Area = r^2 \int_{\pi/4}^{\pi/2} \sin\varphi \, d\varphi \int_0^{2\pi} d\theta$$

$$Area = r^2 [-\cos\varphi]_{\pi/4}^{\pi/2} [\theta]_0^{2\pi}$$

$$Area = r^2 [-(0) - (-0.7071)][2\pi - 0]$$

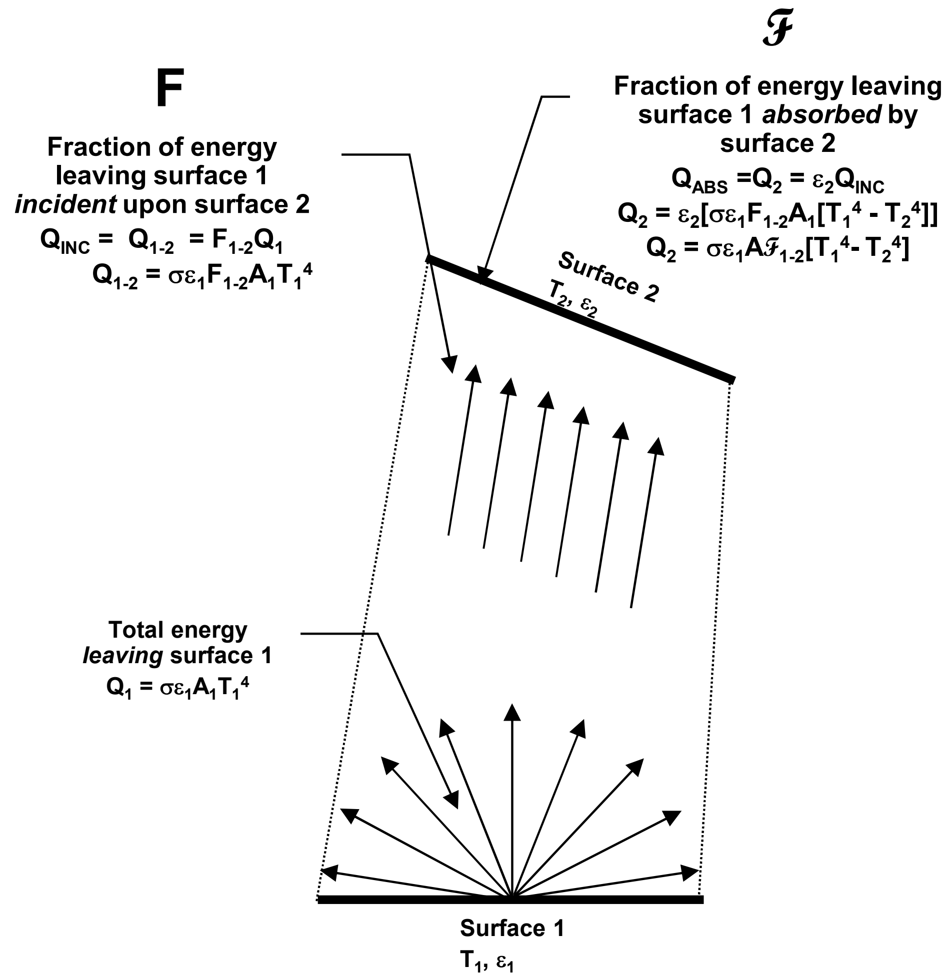
$$Area = r^2 [0.7071 * 2\pi]$$

$$Area = 1.4142\pi r^2$$

$$\boxed{VF = \frac{Area}{Area_{HEMISPHERE}} = \frac{1.4142\pi r^2}{2\pi r^2} = 0.7071}$$

Radiative Heat Transfer – Using All of This

- **Once you've determined view factors, the radiation heat transfer problem between diffuse gray surfaces can be set up**
- **Equation to calculate the total energy emitted by Surface 1**
- **Calculate view factor between Surface 1 and 2 ("VF_{1-2}" or simply "F_{1-2}")**
- **Use properties of Surface 2 to determine the heat absorbed**
- **"$\mathscr{F}$" or "*SF*" or "script F" is a function of both geometry and surface properties: $\mathscr{F}_{1-2} = \varepsilon_2 F_{1-2}$**
  - **$\mathscr{F}$: represents the fraction of energy leaving surface 1 that is *absorbed* by surface 2**
  - **Note: when radiating to space (ε_2=1.0), then F = $\mathscr{F}$**



Between 2 gray-bodies:

$$Q = \sigma * \varepsilon_1 * A_1 * (\varepsilon_2 * F_{1-2}) * (T_1^4 - T_2^4)$$

$$Q = \sigma * \varepsilon_1 * A_1 * (SF_{1-2}) * (T_1^4 - T_2^4)$$

How do we use all of this?

How do we take the heat transfer equations to determine what the temperatures, heat flows, and thermal problem areas on the spacecraft are?

➔ Set up Energy Balance for your level of analysis

- **Steady State conditions**

$$Q_{IN} = Q_{OUT}$$

- **Transient conditions**

$$Q_{IN} = Q_{OUT} + Q_{STORED}$$

$$Q_{IN} = Q_{OUT} + [M * C_p * dT / dt]$$

(Where M = mass, C_p = specific heat)

Energy Balance: Sun on a Flat Plate

- $Q_{in} = Q_{out}$ **(Solar only in this case)**

  Since $Q_{in} = A_P\, Q_{solar}\, \alpha$ **and** $Q_{out} = \sigma\, A\, \varepsilon\, T^4$
- **So:**

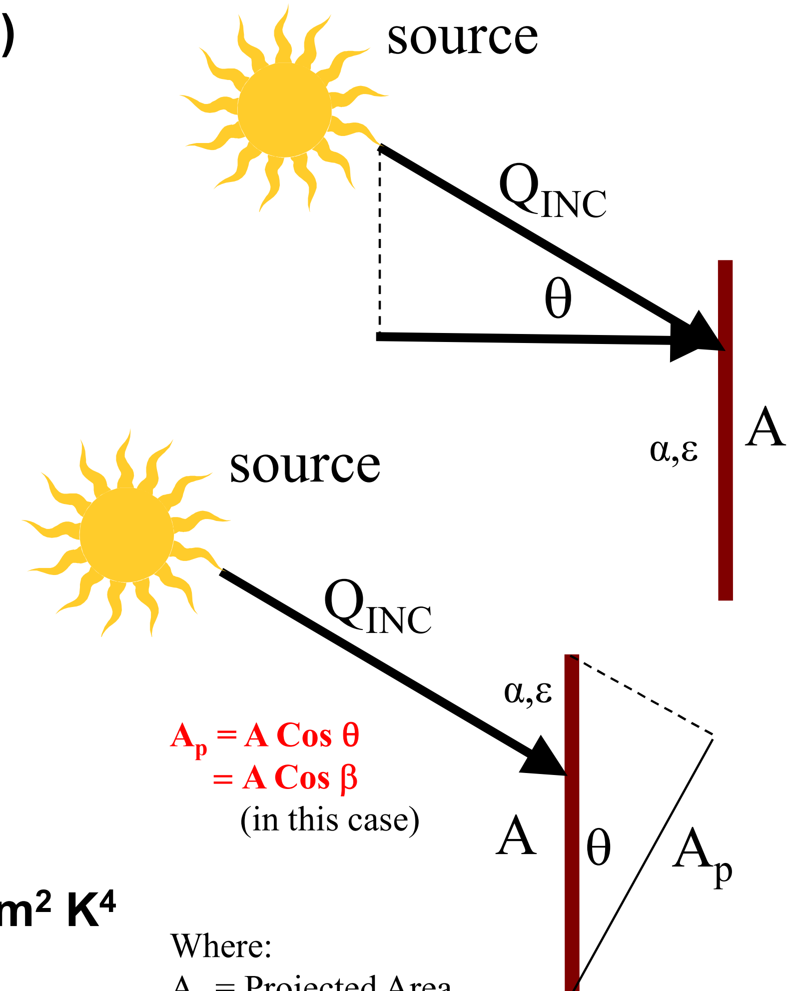
  $A_P\, Q_{solar}\, \alpha = \sigma\, A\, \varepsilon\, T^4$

  $T^4 = (A_P\, Q_{solar}\, \alpha) / (A\, \sigma\, \varepsilon)$

  $= (A_P / A)\, (Q_{solar} / \sigma)\, (\alpha / \varepsilon)$

- **Where:**

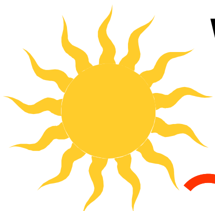
Where:	SI
Q - Heat Flow	Watts
T - Absolute Temperature	°K
A - Surface Area	m^2
A_P - Projected Area	m^2
Q_{solar} - Solar Constant	1367 W/m^2
σ - Stefan-Boltzmann Constant	$5.67 \times 10^{-8}\ W/m^2\ K^4$
α - Solar Absorptance	
ε – IR Emittance	



Where:
A_p = Projected Area
A = Surface Area
Θ = Angle between surface normal and solar vector

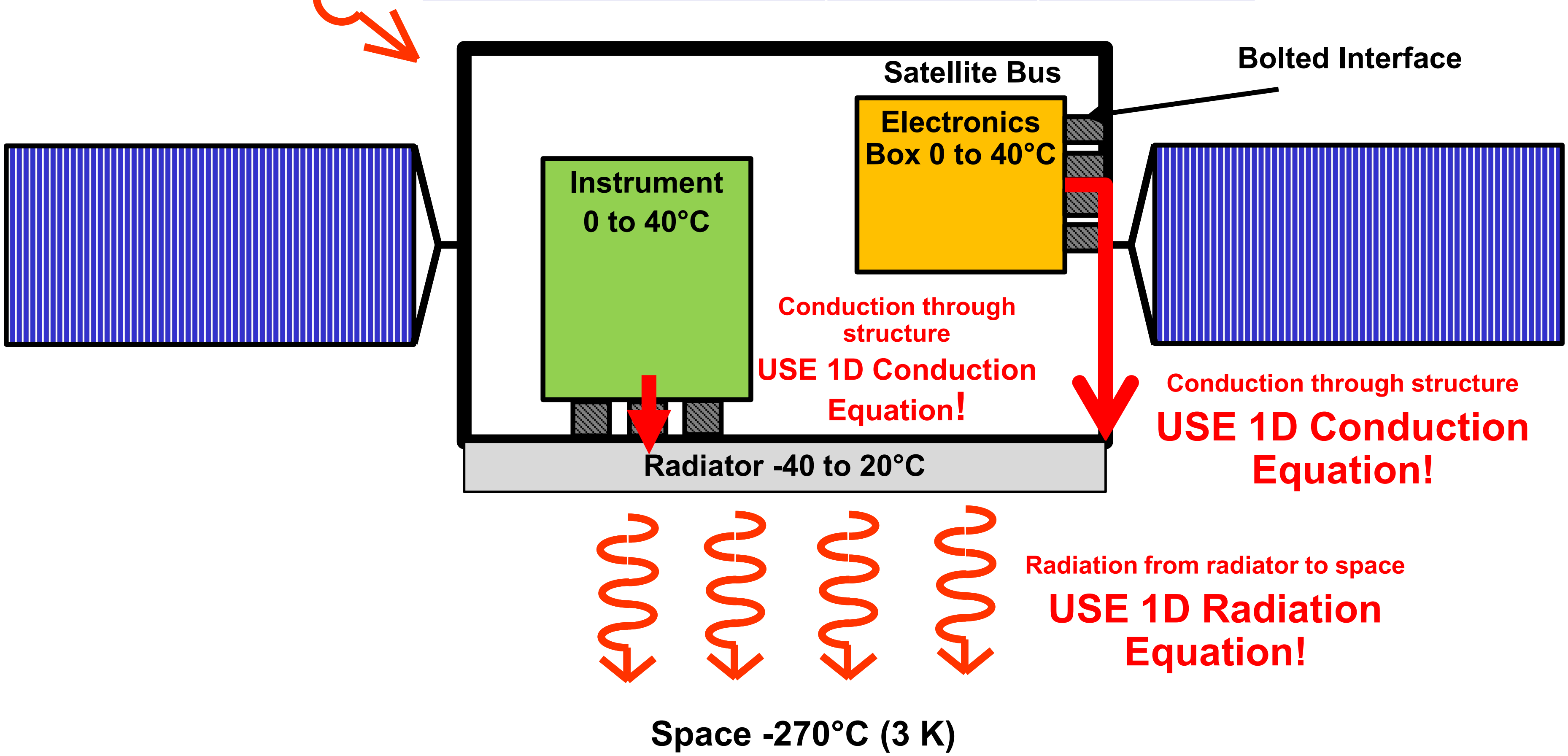
Other Flat Plate Examples are in the Backup Slides

Energy Balance: Satellite



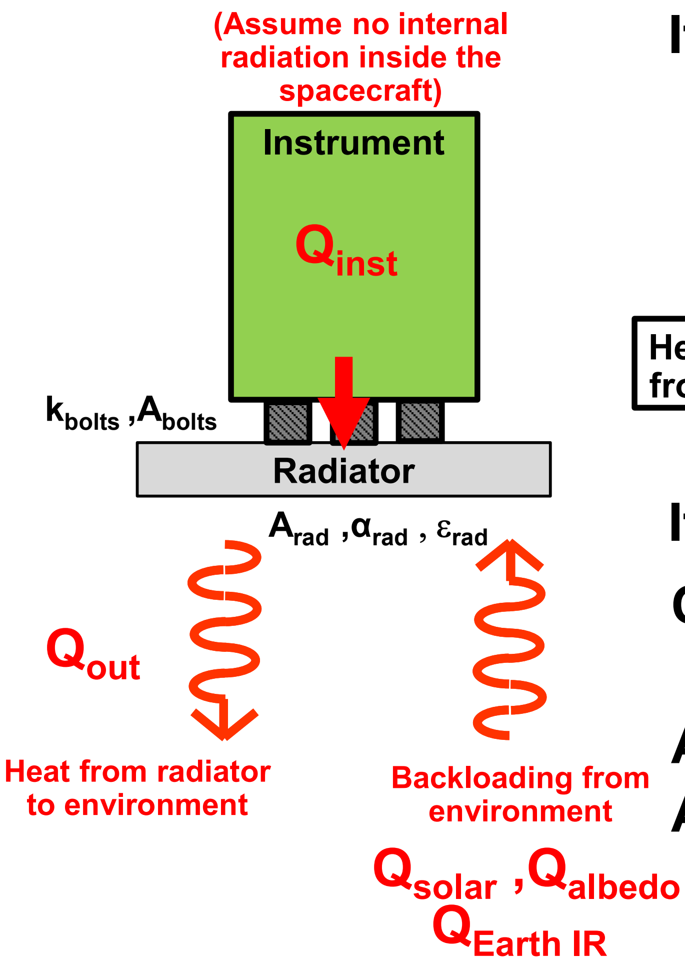
Worst case environmental loads:

	Hot Case	Cold Case
Solar Flux	1412 W/m^2	1322 W/m^2
Albedo	0.35	0.25
Earth IR	267 W/m^2	211 W/m^2
Component Powers	Max.	Min./ Off



Energy Balance: Satellite

- Let's use the example of the instrument dissipating to a radiator, then to the environment



If control volume was around instrument:

$$Q_{in} = Q_{out}$$

$$\underbrace{Q_{inst}}_{\text{Heat dissipation from instrument}} = \underbrace{(\#\ bolts)[\ k_{bolts} A_{bolts}\ (T_{inst} - T_{radiator})]}_{\text{This can be used to find the bolted interface area needed to conductively transfer instrument heat, or resultant instrument temperature based on radiator temperature}}$$

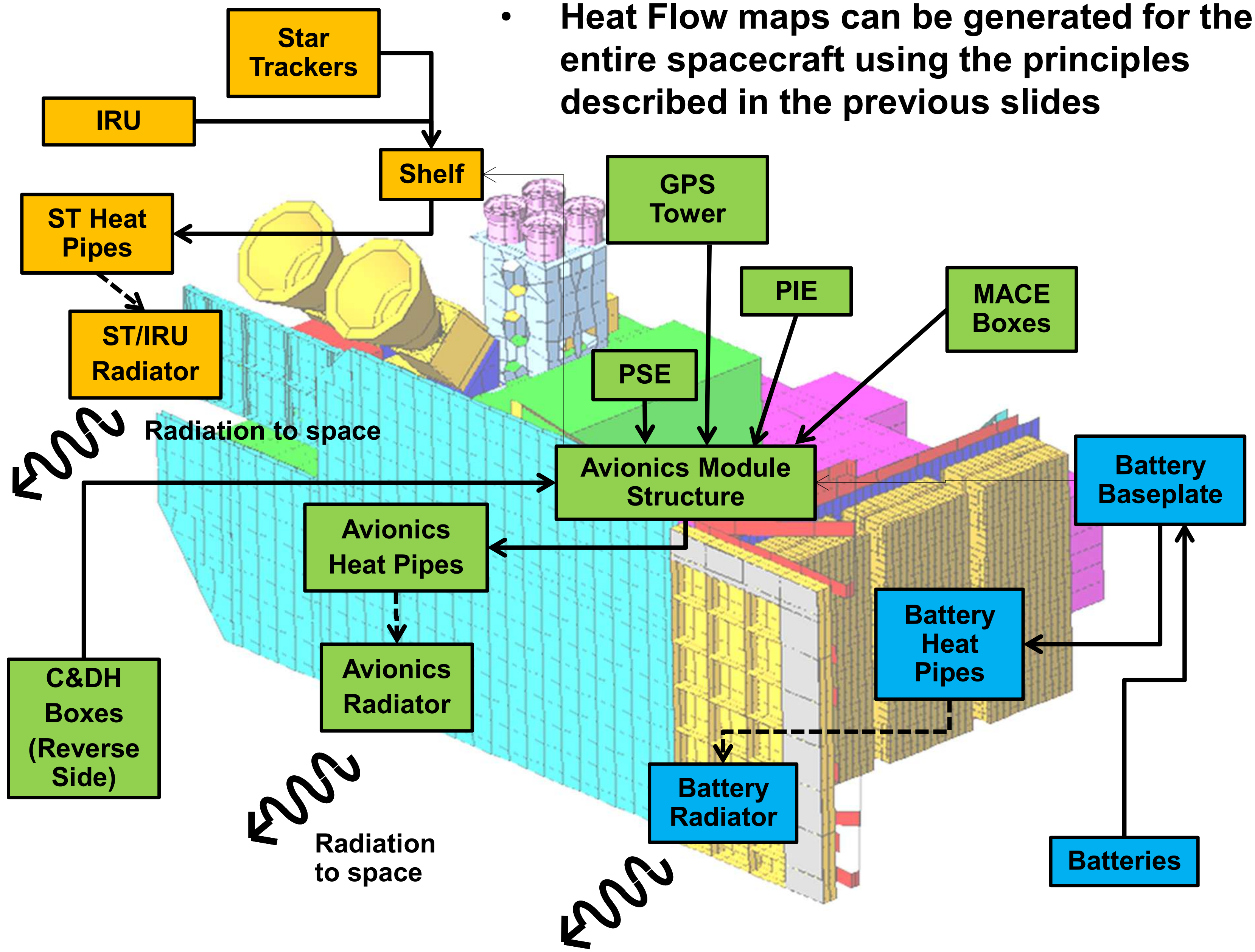
If control volume was around radiator:

$$Q_{in} = Q_{out}$$

$$A_{P,\ Solar}\ Q_{solar}\ \alpha_{rad} + (Albedo)\ A_{P,\ Albedo}\ Q_{solar}\ \alpha_{rad} + A_{P,\ Earth\ IR}\ Q_{Earth\ IR}\ \varepsilon_{rad} + Q_{inst} = \mathcal{F}\ \sigma\ A_{rad}\ \varepsilon_{rad}\ (T_{rad}^4 - T_{space}^4)$$

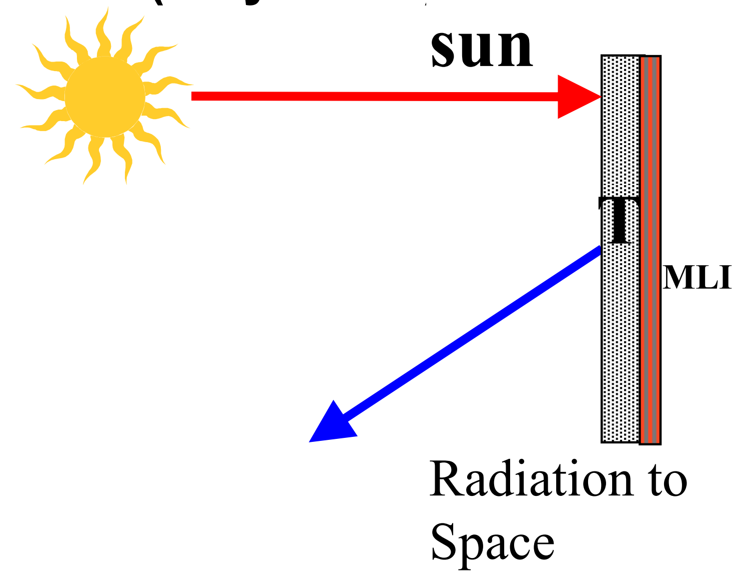
Heat Flows

- Heat Flow maps can be generated for the entire spacecraft using the principles described in the previous slides



Thermal Example Problems: Insulated Plate

- **Solar vector normal, insulated back-side, black (very black)**
- $\alpha = \varepsilon = 1.0$; A= 1.0



$$T^4 = (A_P * S * \alpha) / (A * \sigma * \varepsilon)$$

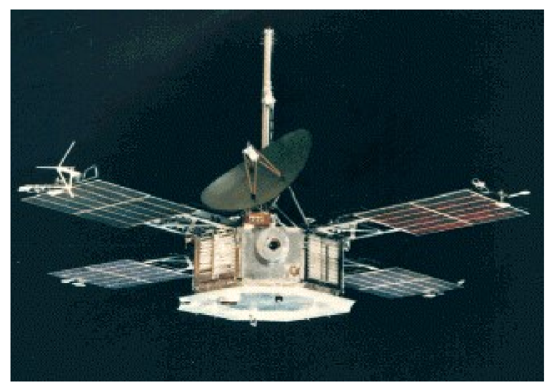
$$A_P = A = 1.0\ m^2$$

$$S = 1367\ W/m^2$$

$$\sigma = 5.67 \times 10^{-8}\ W/m^2\ K^4$$

$$T = 121\ °C$$

Related Fact



Many NASA Satellites, especially earlier satellites (Mariner 5 pictured), have a hexagonal or octagonal bus. Radiators can take advantage of these angles, and it simplifies analysis of radiative couplings to the environment

Thermal Example Problems: Uninsulated Plate

- **Solar vector normal, black (very black)**
- $\alpha = \varepsilon = 1.0$; **A= 1.0**

$$T^4 = (A_P * S * \alpha) / (A * \sigma * \varepsilon)$$

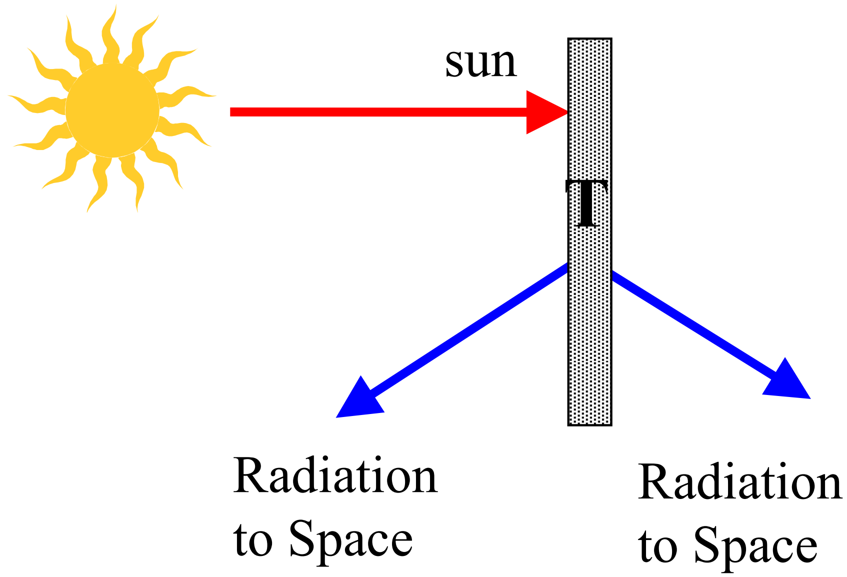
$A_P = 1.0\ m^2$

$A = 2*A_P\ m^2$ **(both sides)**

$S = 1367\ W/m^2$

$\sigma = 5.67 \times 10^{-8}\ W/m^2\ K^4$

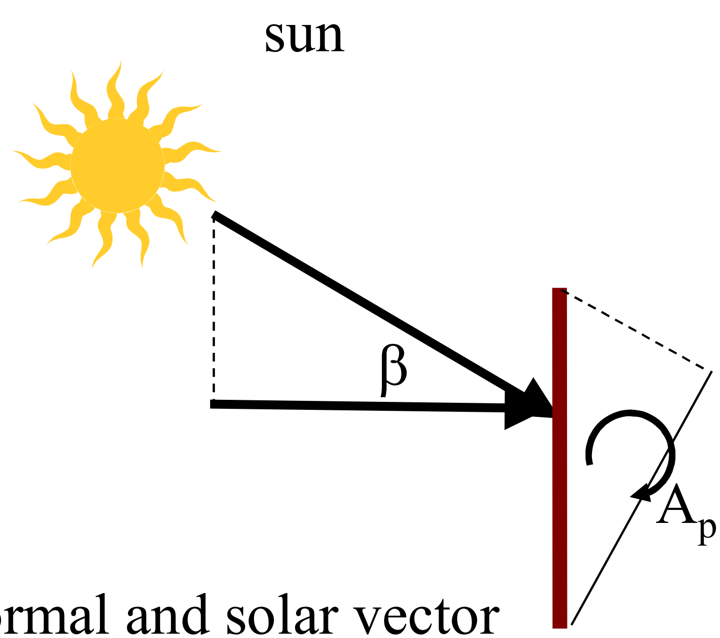
$T = 58\ °C$



This is very similar to a solar array, where the back (non-illuminated) side is used to provide additional cooling and increase reliability. Note that for solar arrays, you would need to subtract off the power generated and transferred to the S/C in the energy balance equation.

Thermal Example Problems: Sun on Spinning Surface

$$A_P = A \cos \beta$$



Where:

A_P = Projected Area

A = Surface Area

β = Angle between surface normal and solar vector

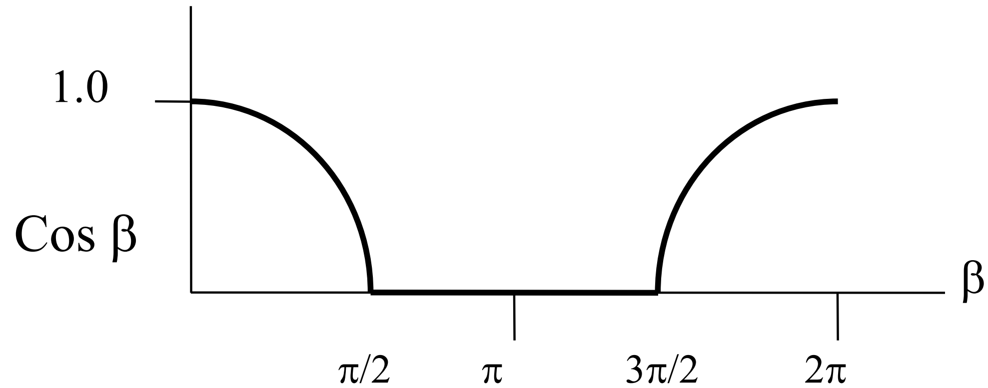
Thermal Example Problems: Spinning Flat Plate (Black; Insulated on back)

- **Solar vector normal, black**
- $\alpha = \varepsilon = 1.0$; **A= 1.0**

$$T^4 = (A_P * S * \alpha) / (A * \sigma * \varepsilon)$$

Spin average $A_P = A/\pi = 1/\pi\ m^2$
$A = 1\ m^2$ (one side)
$S = 1367\ W/m^2$
$\sigma = 5.67 \times 10^{-8}\ W/m^2\ K^4$

T = 23 °C



$$A_{P,\,avg} = \frac{1}{2\pi}\int_0^{2\pi} A\cos d\beta = \frac{1}{2\pi} * [\int_0^{\pi/2} A\cos d\beta + \int_{\pi/2}^{3\pi/2} A0\,d\beta + \int_{3\pi/2}^{2\pi} A\cos d\beta]$$

$$A_{P,\,avg} = \frac{A}{2\pi} * [\sin\beta|_0^{\pi/2} + \sin\beta|_{3\pi/2}^{2\pi}]$$

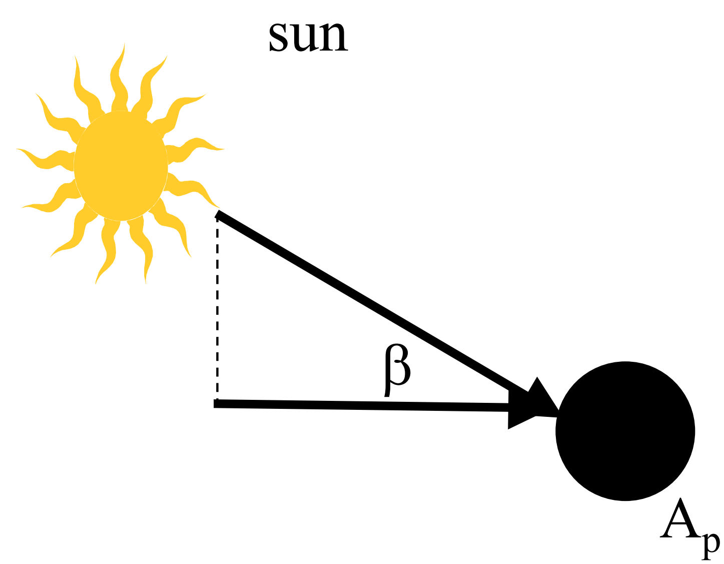
$$A_{P,\,avg} = \frac{A}{2\pi} * [(1-0)+(0)+(0-(-1))]$$

$$A_{P,\,avg} = \frac{A}{2\pi} * [2] = \frac{A}{\pi}$$

Question: What would be the average temperature of a spinning *uninsulated* plate?

Thermal Example Problems: Property Uncertainty

- **Black Sphere in the Sun**
  - $(\alpha / \varepsilon)_{NOM}$ = 0.96/0.87 = 1.103
  - $(\alpha / \varepsilon)_{MIN}$ = 0.94/0.89 = 1.056
  - $(\alpha / \varepsilon)_{MAX}$ = 0.98/0.85 = 1.153



$T^4 = (A_P * S * \alpha) / (A * \sigma * \varepsilon)$

$A = 4\pi r^2$ m^2 (sphere)

$A_P = \pi r^2$ m^2 (circle)

$A_P / A = 1/4$

$S = 1367$ W/m^2

$\sigma = 5.67 \times 10^{-8}$ $W/m^2\ °K^4$

MAX: T = 15 °C

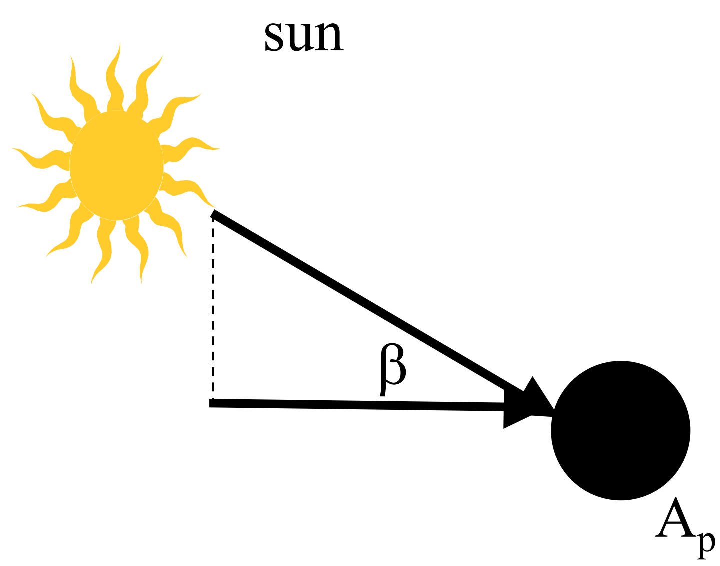
NOM: T = 13 °C

MIN: T = 9 °C

T = 13°C +2°/- 4°C

Thermal Example Problems: Seasonal Variation

- **Black Sphere in the Sun**
  - $(\alpha / \varepsilon)_{NOM}$ = 0.96/0.87 = 1.103
  - $(\alpha / \varepsilon)_{MIN}$ = 0.94/0.89 = 1.056
  - $(\alpha / \varepsilon)_{MAX}$ = 0.98/0.85 = 1.153



$T^4 = (A_P * S * \alpha) / (A * \sigma * \varepsilon)$

$A = 4\pi r^2$ m² (sphere)

$A_P = \pi r^2$ m² (circle)

$A_P / A = 1/4$

- $S_{MAX} = 1422$ W/m²
- $S_{NOM} = 1367$ W/m²
- $S_{MIN} = 1312$ W/m²

$\sigma = 5.67 \times 10^{-8}$ W/m² K⁴

MAX: T = 19 °C
NOM: T = 13 °C
MIN: T = 7 °C

T = 13°C +/- ~6°C

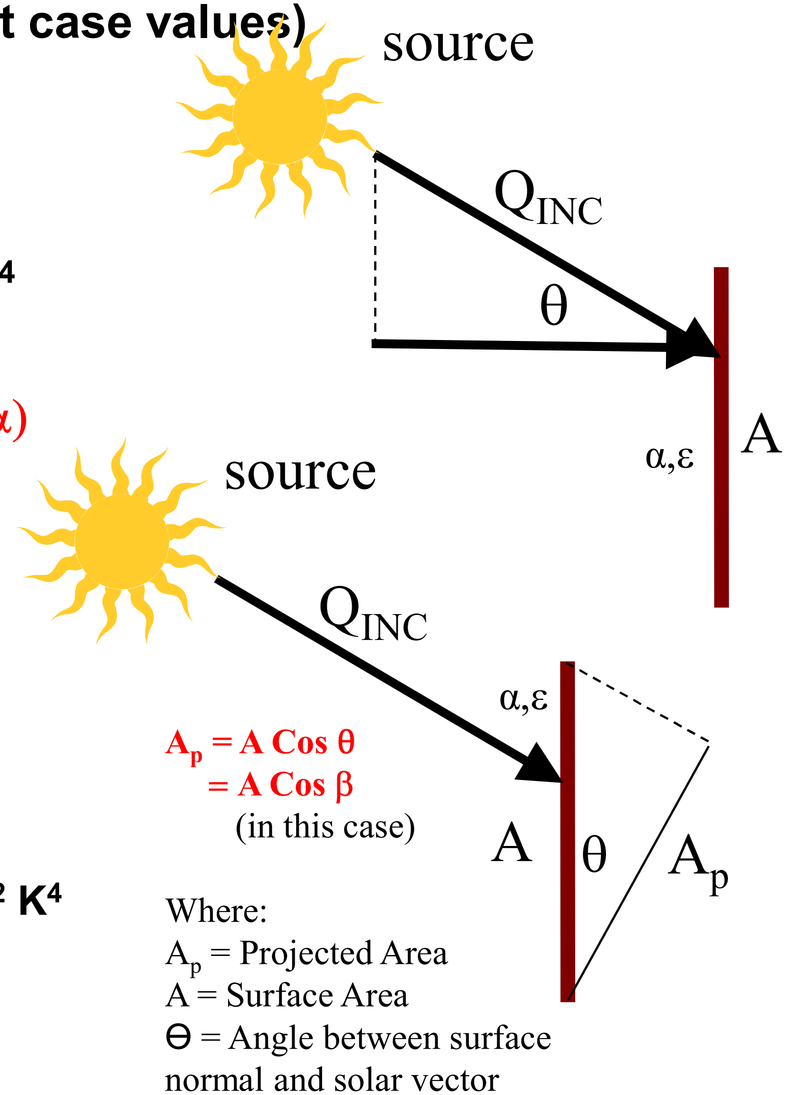
Thermal Example Problems: Radiator Sizing, additional information

- $Q_{in} = Q_{out}$ (Size radiator in hot case: use only hot case values)
- $Q_{in} = A_P{*}S{*}\alpha + Q_{albedo} + Q_{Earth\ IR} + Q_{internal}$
  $Q_{out} = \sigma * A * \varepsilon * T^4$
- So:

  $A \cos\theta\, S\, \alpha + Q_{albedo} + Q_{Earth\ IR} + Q_{internal} = \sigma A \varepsilon T^4$

  $Q_{albedo} + Q_{Earth\ IR} + Q_{internal} = A\,(\sigma \varepsilon T^4 - \cos\theta\, S\, \alpha)$

  $A = (Q_{albedo} + Q_{Earth\ IR} + Q_{internal}) / (\sigma \varepsilon T^4 - \cos\theta\, S\, \alpha)$



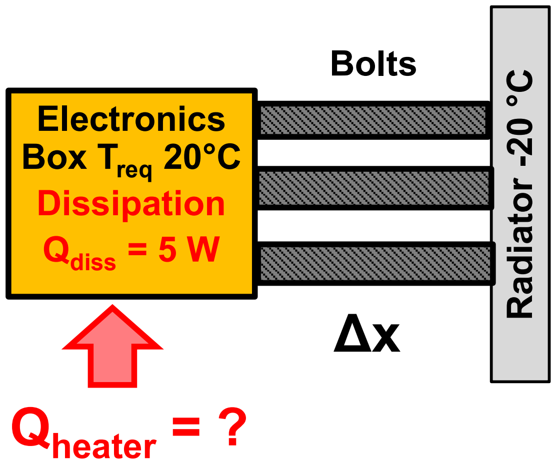
- Where:

	SI
Q - Heat Flow	Watts
T - Absolute Temperature	K
A - Surface Area	m^2
A_P - Projected Area	m^2
S - Solar Constant	1367 W/m^2
σ - Stefan-Boltzmann Constant	5.67×10^{-8} $W/m^2\,K^4$
α - Solar Absorptance	
ε – IR Emittance	

From the calculations, you will see that it is most desirable to place the radiator on the surface furthest away from any environmental loading, and make the radiator coating as low of an (α / ε) ratio as possible

Thermal Example Problems: Heater Sizing, Conduction-driven example

- What is the heater power needed here? (only size heaters in cold case)



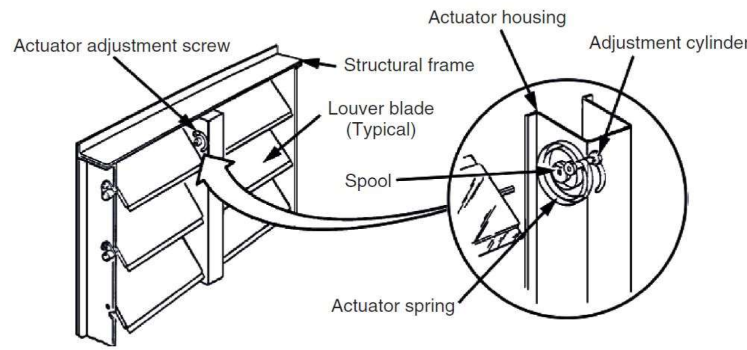
$Q_{in} = Q_{out}$

$Q_{diss} + Q_{heater} = (\#\ bolts)*[k_{bolts}\ A_{bolts} / \Delta x_{bolts}]*(T_{required,\ box} - T_{radiator})$

$Q_{heater} = (\#\ bolts)*[k_{bolts}\ A_{bolts} / \Delta x_{bolts}]*(T_{required,\ box} - T_{radiator}) - Q_{diss}$

Active Example: Louvers

- **Typically a bimetallic spring-actuated device placed over a radiator which modulates heat transfer from spacecraft to environment**
  - When closed, prevents view from radiator to space
  - When open, allows radiative coupling to environment
  - Louvers can reject ~6x more heat in open configuration vs. closed configuration
  - Louver actuators driven by radiator temperature
  - Louvers generally can hold components to a 10 K stability range: can reduce or negate requirement for heater power
- **Louvers have decades of flight heritage**
  - Extremely successful and reliable thermal component
  - There has never been a known complete louver failure in flight



“Venetian-blind” type louver

“Pinwheel” type louver