Electrical & Electronics Background reading

Presented by

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Small biography

- Ferdinando Tonicello, Italian, resident in the Netherlands since 1997.
- Presently (since 2015) I am Electrical Lead Engineer in ESTEC.
- I am author/co-author of a number of papers and patents on power system, power supply and conditioning for space applications.
  In the course of my professional life, I have been preparing a number of tutorials on the electrical and electronic domain for space applications.
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Some initial info

- Interactive! Do not think it will be just “listening”
    - Q&A approach

“Q&A”

- Standards are only mentioned briefly at the end… the important message being what drives the standard requirements

- Attempt to let you naturally conceive electrical & electronic requirements for space applications

- You will forgive me if I will go quickly on some slides reporting technological details… in any case the presentation will be available to you for further consultation

Slide 176

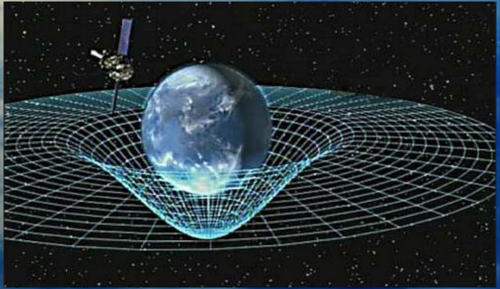
“Key” slide 97

Agenda

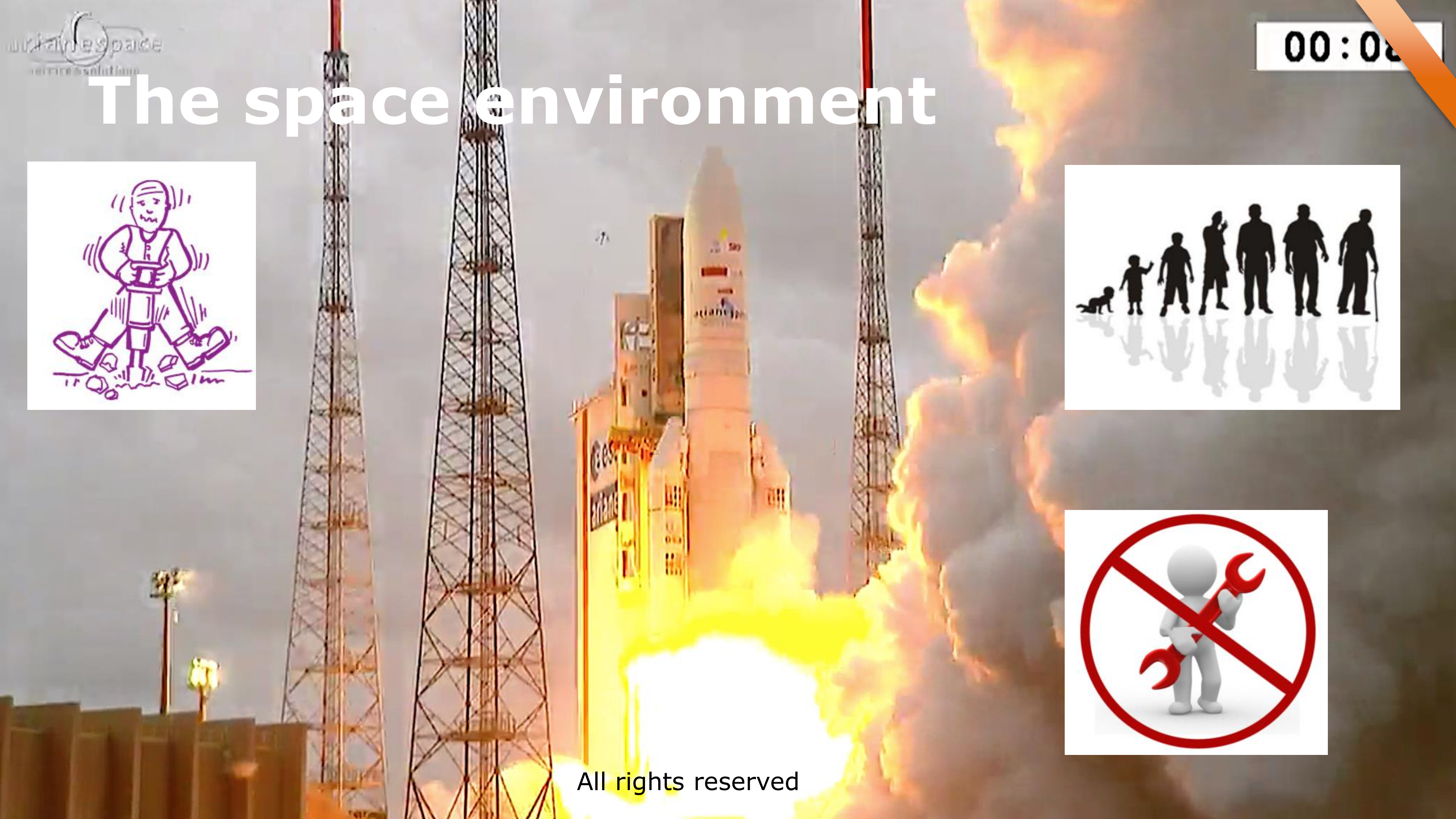
- The space environment – small introduction!
- Typical space missions
- Orbits
- Electrical/electronic for space
- Electrical/electronic architecture
- Power subsystem architecture
- Power generation
    - Some concepts on spacecraft charging
- Energy storage
- Power management and distribution
- Electro-Magnetic Compatibility issues
- Brief introduction to ECSS (E-20) standards

The space environment

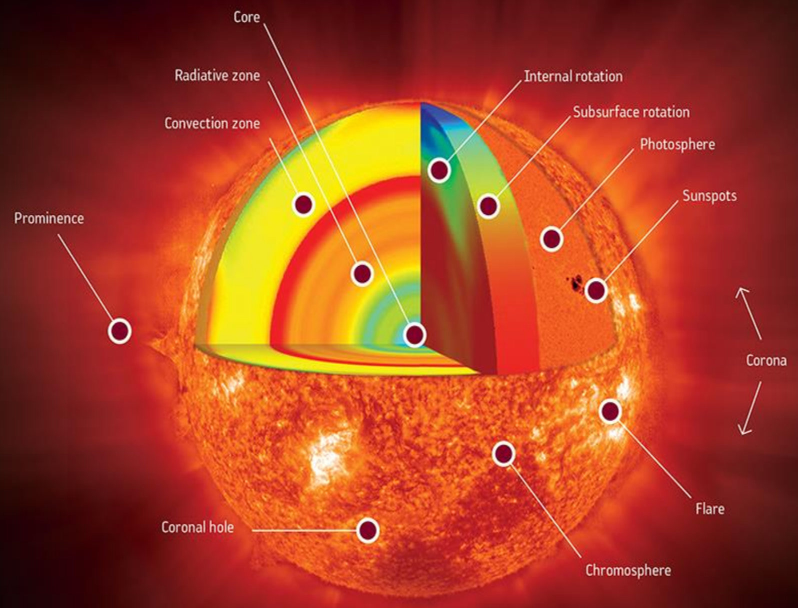
- What is special about space environment?
- How does it impact electrical and electronic functions?

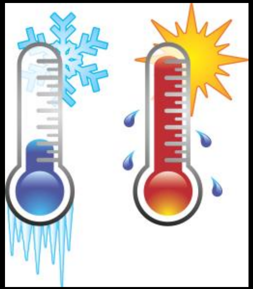


The space environment



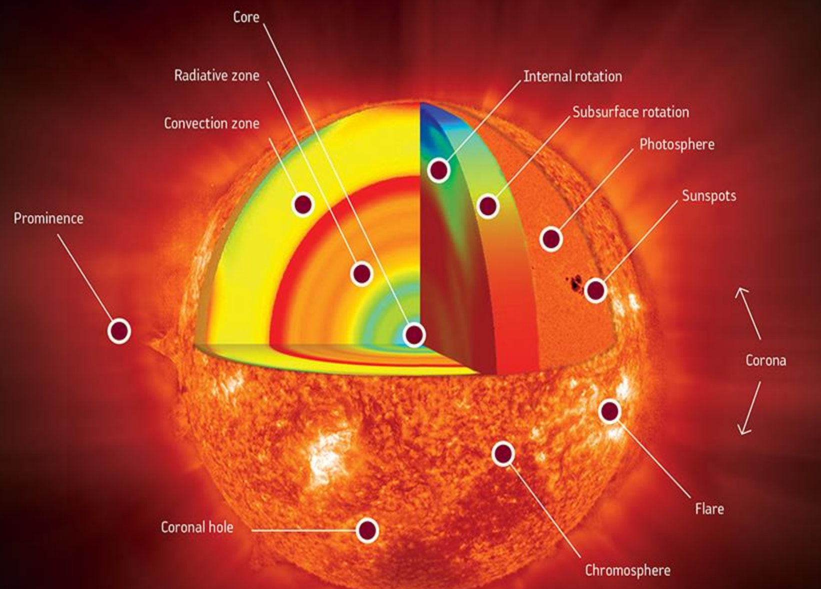
The space environment

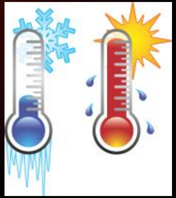




The space environment

- No “ambient” temperature
  - Temperature results from balance between incoming sun flux and radiation toward empty space (4°K)
  - Metallic piece exposed to sun in earth orbit may reach 200°C
  - Solar generator at end of GEO eclipse may typically be at -180°C
- Sun flux varies in inverse square of sun distance
- But… inside the spacecraft the temperatures are typically benign thanks to the thermal control!
  - Typ -20 to +60 °C or less





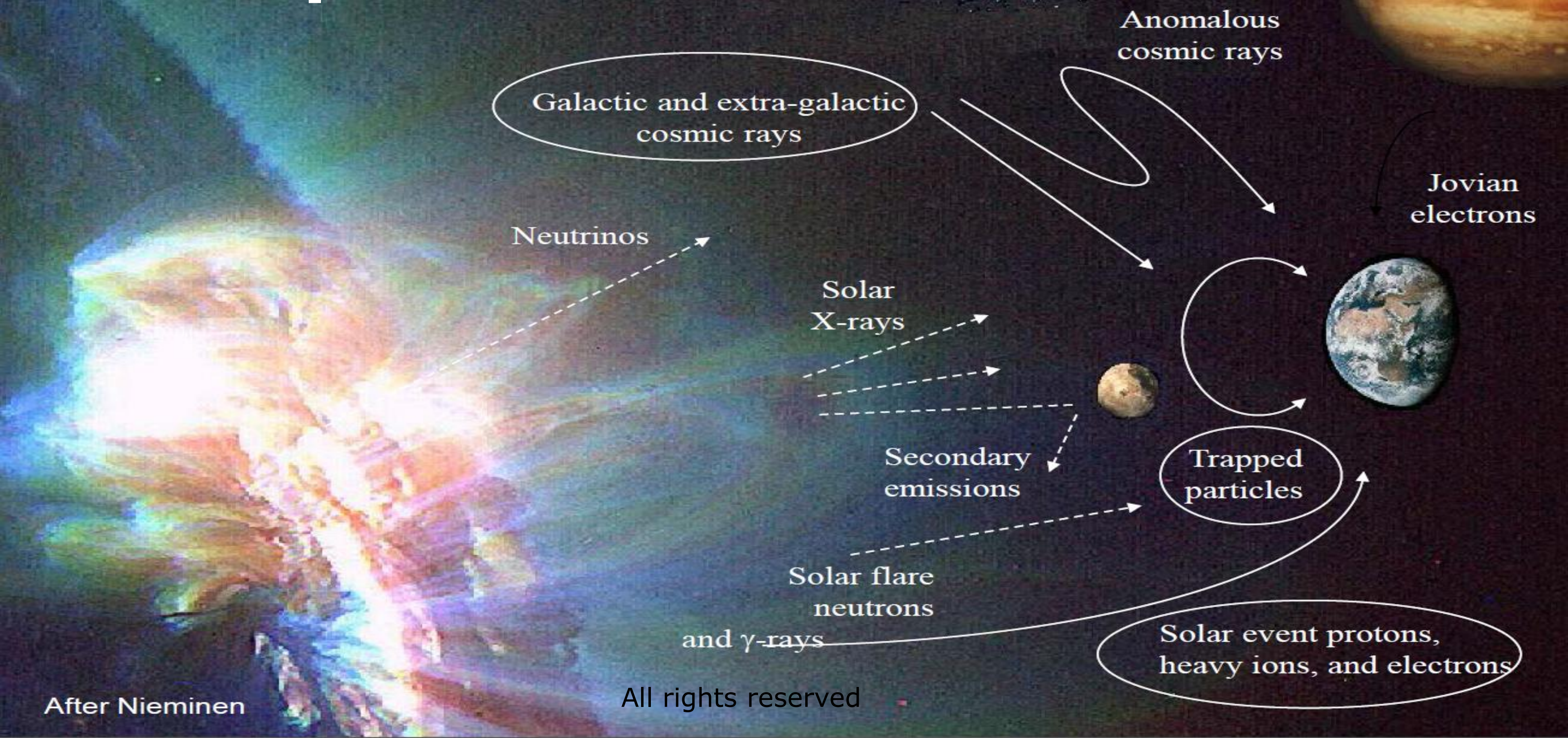
The space environment - vacuum

- The main consequences of vacuum conditions:

1. **Not** possible to dissipate heat by **convection**, hence exploit conduction (and irradiation) to keep temperatures under control

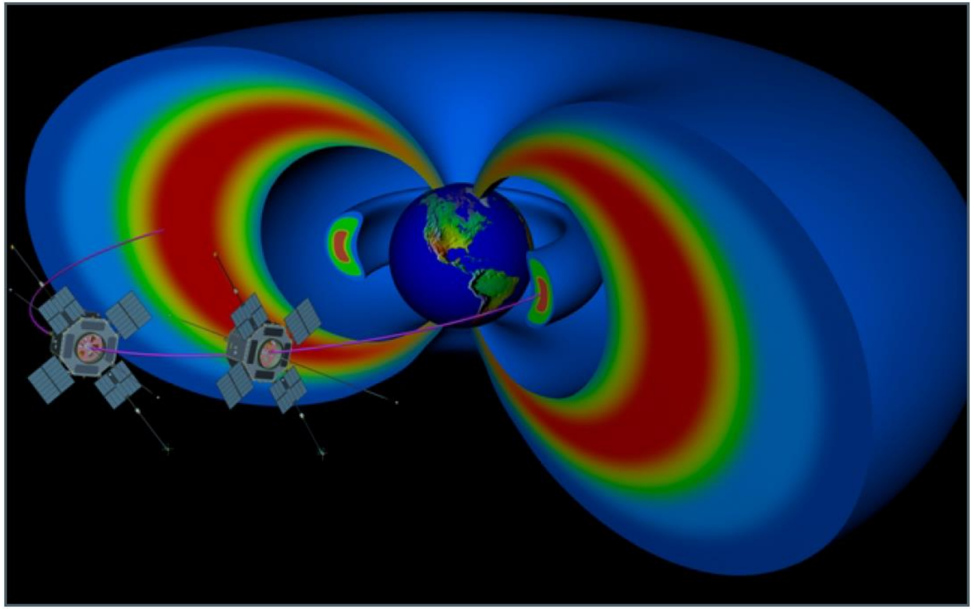
2. Need to take into account **out-gassing** properties of materials

The space environment - radiation



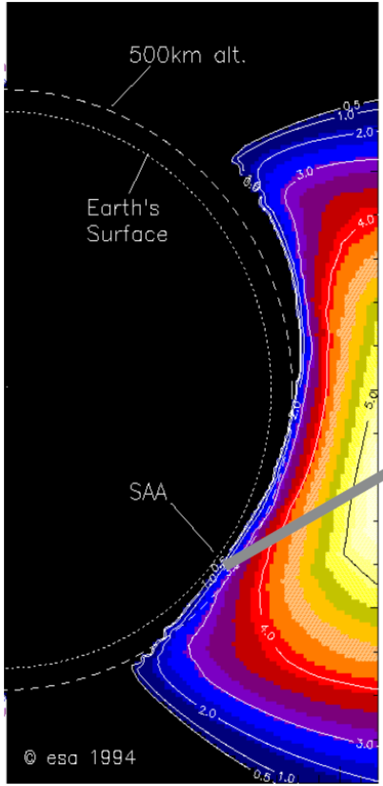
The trapped radiation belts

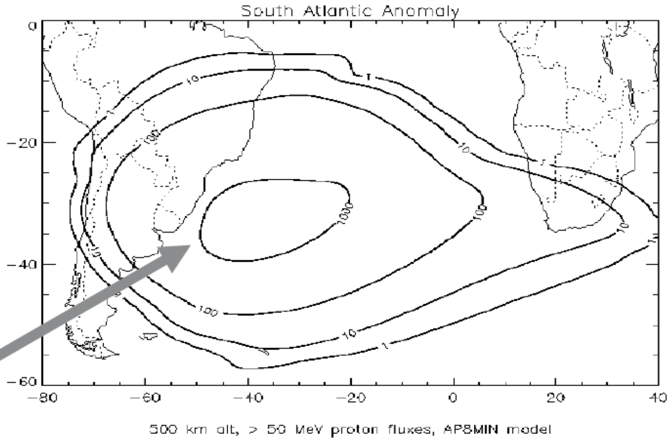
- Van Allen belts. Discovered during first space missions.
- Electrons and protons trapped in Earth Magnetic field (Lorentz force)
- Dynamic environment
- Inner belt is dominated by a population of energetic protons up to ~400 MeV energy range
- Inner edge is encountered as the South Atlantic Anomaly (SAA)
- Outer Belt is dominated by a population of energetic electrons up to 7 MeV

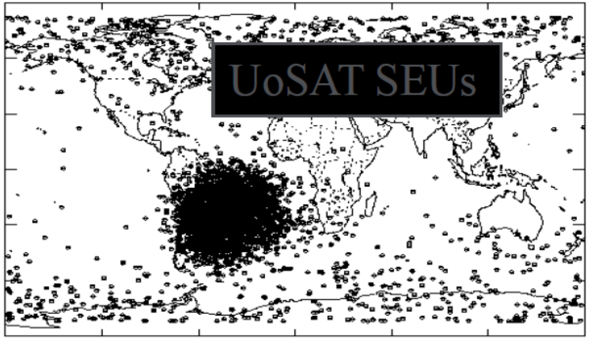


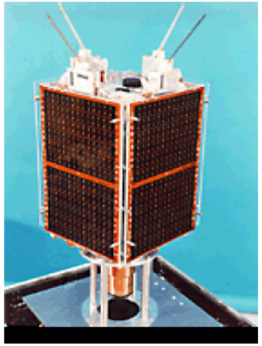
NASA, Radiation Belts Storm Probe mission

The South Atlantic Anomaly









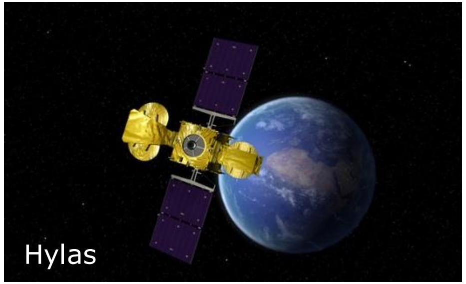
Interaction of Radiation Particles with Electronic Devices and Materials

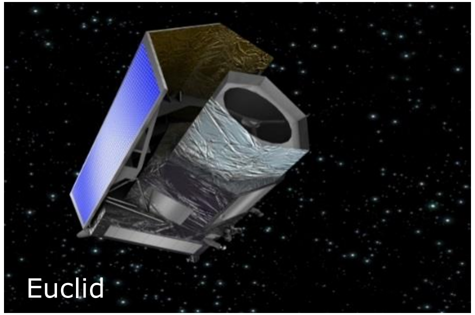
- The effects of radiation on electronic devices and materials depend on :
  - *Type of radiation (photon, electron, proton...)*
  - *Rate of interaction*
  - *Type of material (Silicon, GaAs..)*
  - *Component related (process, structure, etc.)*

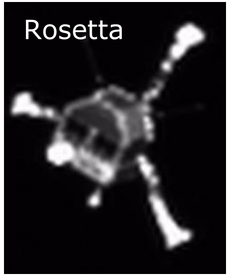
- **Consequences : Ionization (Total Ionizing Dose – TiD - and Single Event Effects - SEE -) and Displacement Damage**

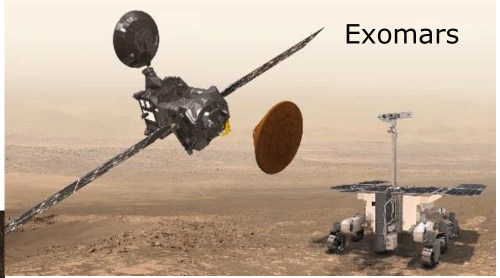
Space missions


Ulysses


Hylas


Euclid


Rosetta


Exomars


Galileo


BepiColombo


Cryosat

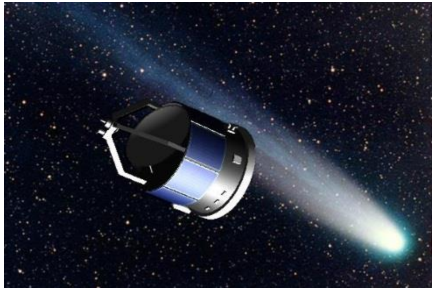

Huygens

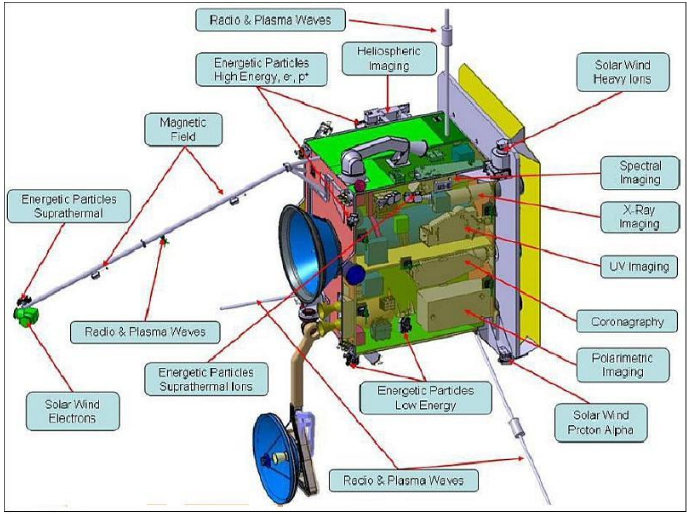
Mission classes

- **Scientific**
- **Earth observation**
- **Telecommunication**
- **Navigation**
- **Manned (not covered in detail)**

Scientific

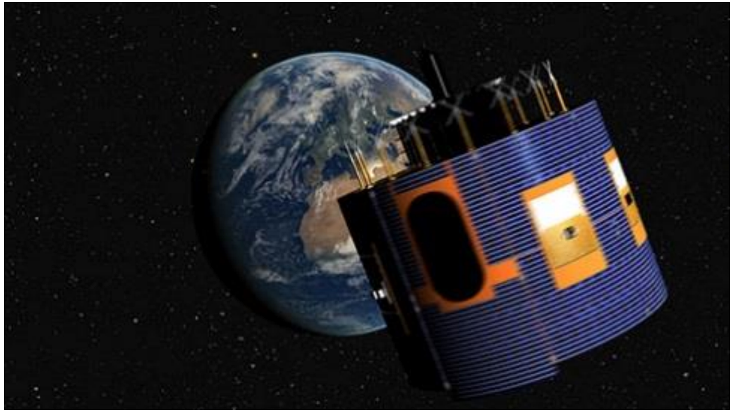
- Can be on many types or orbits or on ground (probes, rovers)
- Most of the time a "one shot" spacecraft
  - Cost and complexity prevent considering a rebuild
  - Mission might not be feasible outside of a given time window (e.g. Giotto to the Halley comet)
  - Robust and reliable solutions preferred !
- Usually aggregates payloads (experiments or instruments) from many laboratories or universities
  - Ancillary functions (especially power conversion) are usually not their core expertise
  - So the electrical designer shall make their life as easy as possible

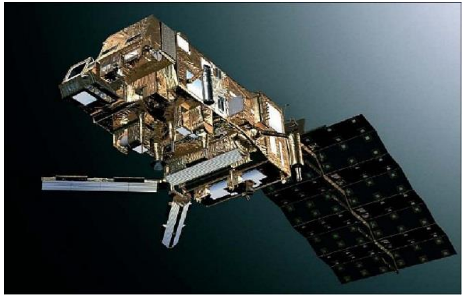




Earth Observation

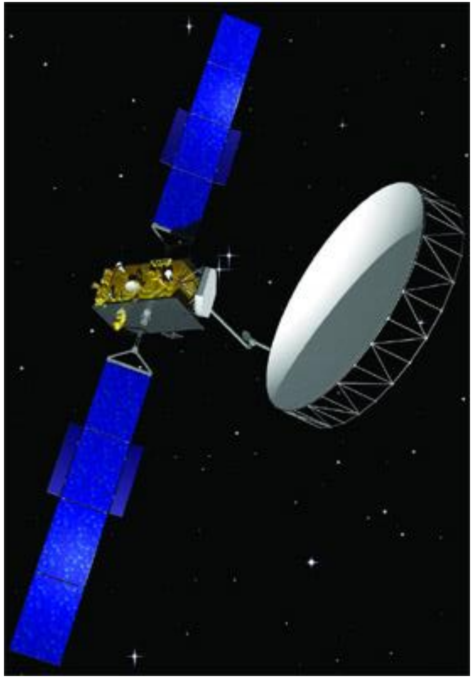
- Most of the time in Low Earth Orbit (LEO), some in Geostationary Orbit (GEO), e.g. the Meteosat series
- Sun Synchronous Orbits often preferred
  - Usually means one eclipse per orbit
- May carry a single or many instruments
  - Single: Synthetic aperture radar (SAR)
  - Multiple: e.g. METOP





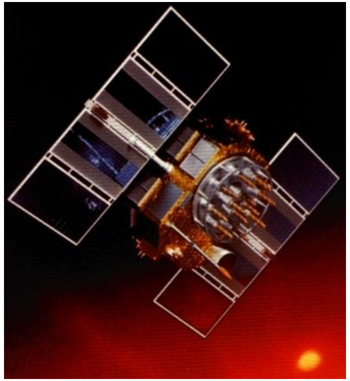
Telecommunication

- Most of them are in (GEO), some constellations (Iridium, O3b, Globalstar) are in “high” LEO (not addressed here)
    - GEO satellites are mostly high power (10kW or more)
- Dictates high voltage power bus (100V is typical)
- Means also large thermal dissipation



Navigation

- Navigation constellations are all in Medium Earth Orbit (MEO) to ensure multiple satellite visibility from ground with a reasonable quantity of them
  - The clocks need very high thermal stability
  - The attitude is not constrained around the Nadir
    - Optimum solar array pointing achievable (and desirable)



Orbits

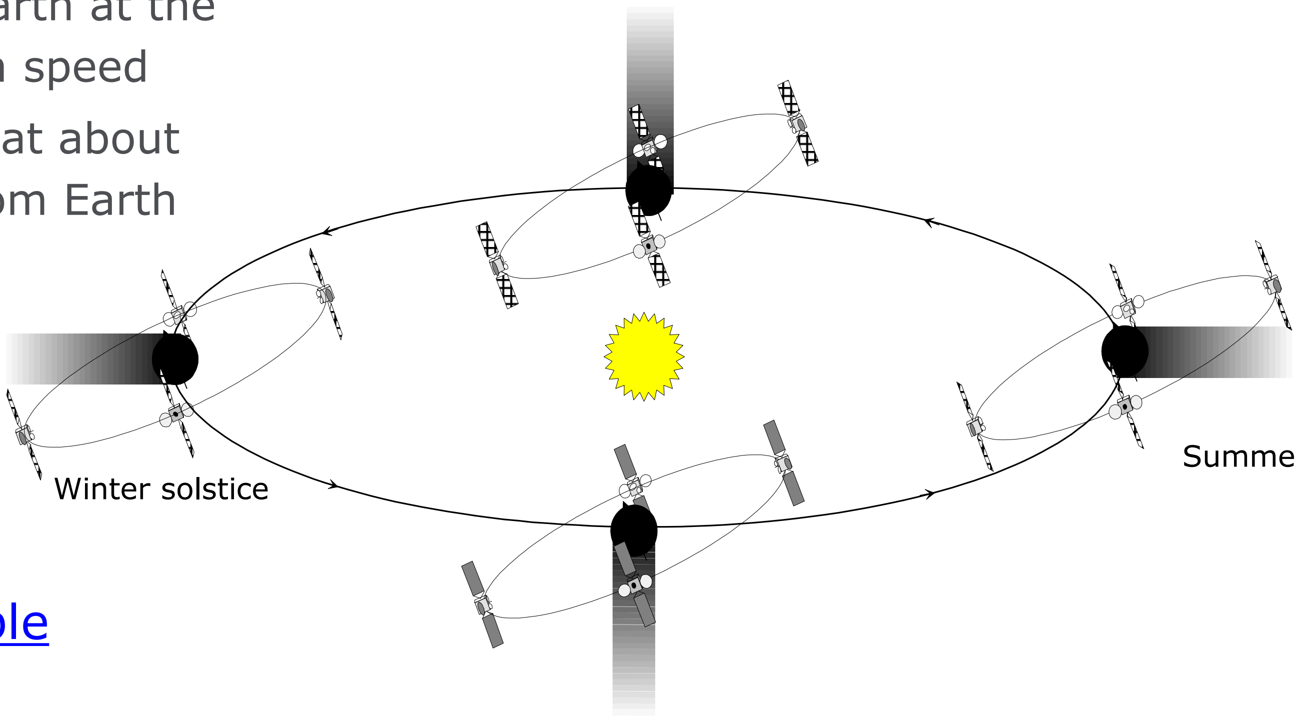
- Geostationary
- Low Earth Orbit
- Sun Synchronous Orbit (SSO)
- Sun Orbits
- Sun/Earth Systems Lagrange Points
- Around a planet…

Or

- On a planet…

Geostationary Orbit (GEO)

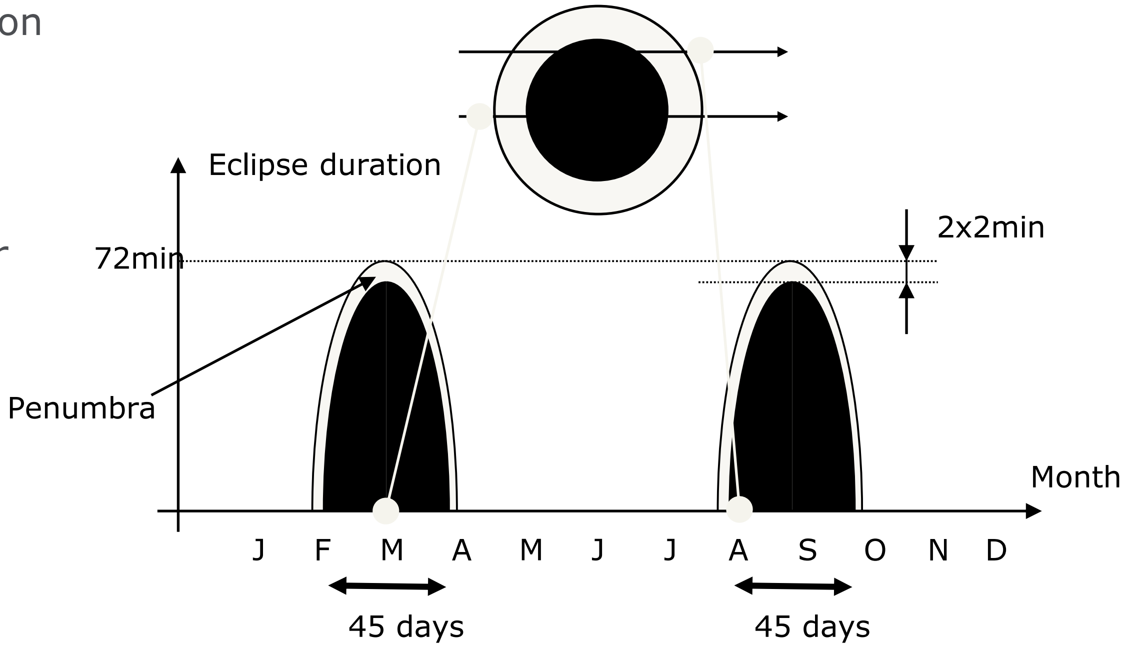
- The satellite rotates around the Earth at the same rotation speed
- Spacecraft is at about 36.000Km from Earth



[GEO example]

Geostationary Orbit (GEO)

- Eclipse regime is benign
  - 90 eclipses per year
  - 1350 on a 15 years mission
- Maximum sun incidence at solstices
  - Occurs outside of eclipse season, when no need for battery recharge
  - More or less balanced budget around the year



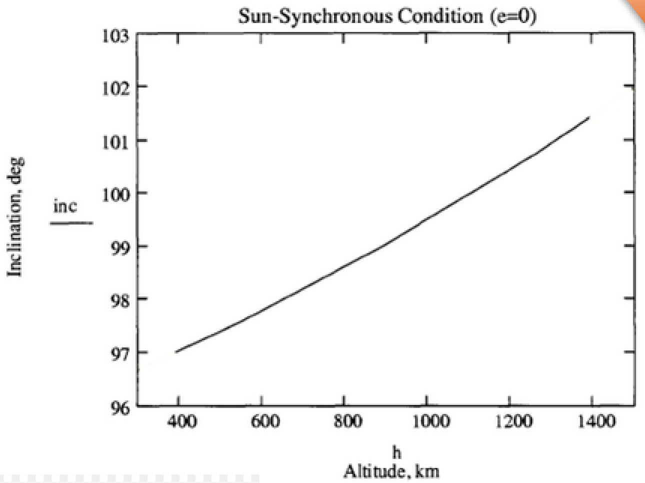
Low Earth Orbit (LEO)

- Close to the Earth!
  - Altitude from 160 to 2000 Km
  - Corresponding orbital period from 88min to 127min
- Eclipses repeat at high rate!
  - Typ 1 orbit every 1.5 hours, approx. 1 hour sun, 0.5 hours eclipse
  - 5840 orbits per year!
  - 58400 orbits per 10 years mission!

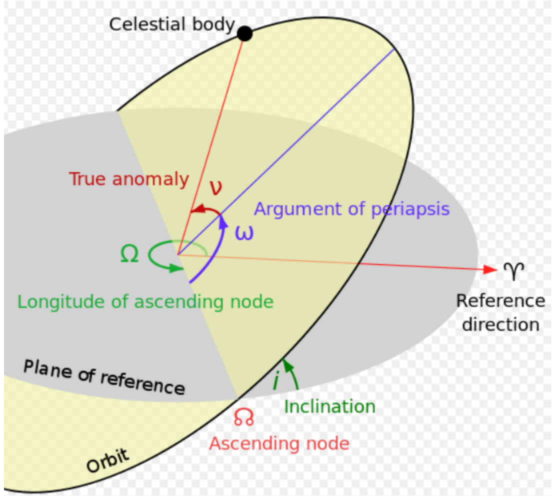
LEO vs GEO

Sun synchronous orbit

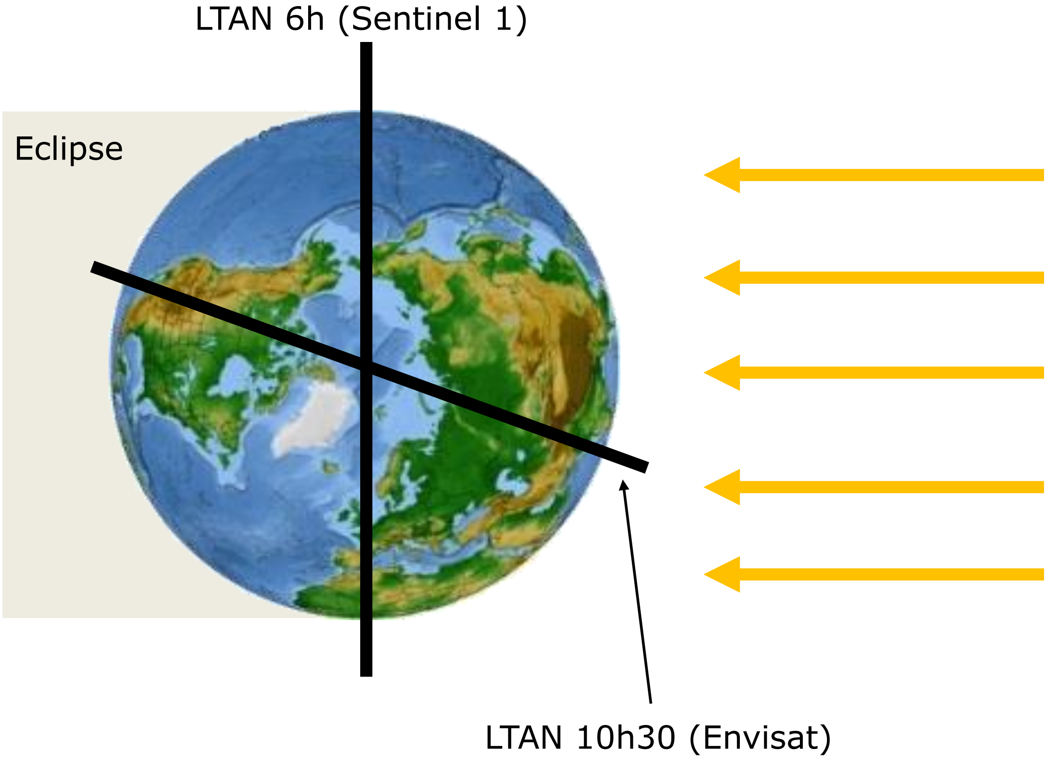
- Particular case of Low Earth Orbit
- Satellite altitude and inclination are selected to guarantee that the satellite passes over any given point of the planet's surface at the same local solar time.



- Eclipse regime depends on Local Time of Ascending Node (LTAN)



Sun synchronous orbit (SSO)



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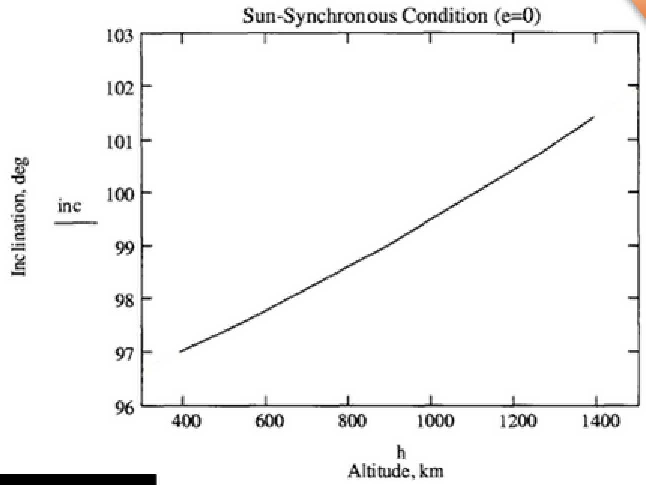
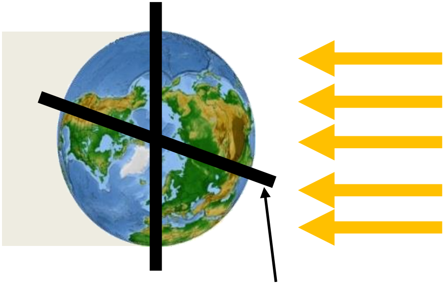
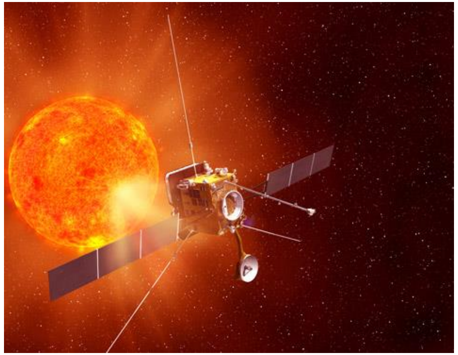


- Eclipse regime depends on Local Time of Ascending Node (LTAN)
  - Moderate on dusk/dawn orbits (6/18h LTAN)
  - Severe on e.g. imaging missions
- Solar array sun angle varies little over the year



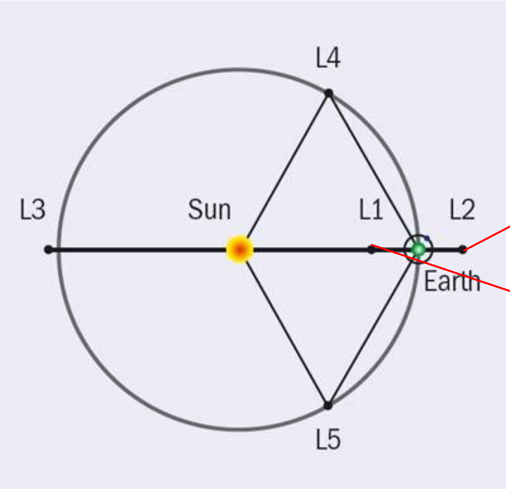
Sun orbits

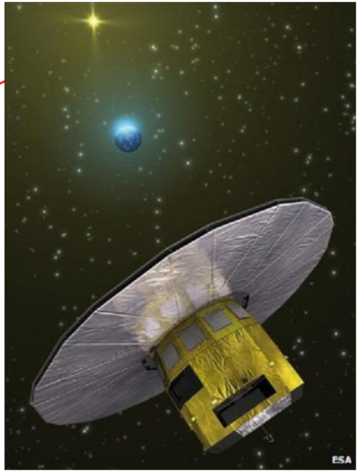
- Very large variations of the Sun distance
  - Large ranges of illumination and temperature
  - Resulting into large changes in solar array performance, when it can work… otherwise resorting to Radioisotope Thermal Generators (RTGs)
- Almost no eclipses
  - Batteries sized on occasional events (LEO phases – LEOP -, Orbit Correction Maneuvers, Planetary flyby, contingency)



Sun/Earth system Lagrange points

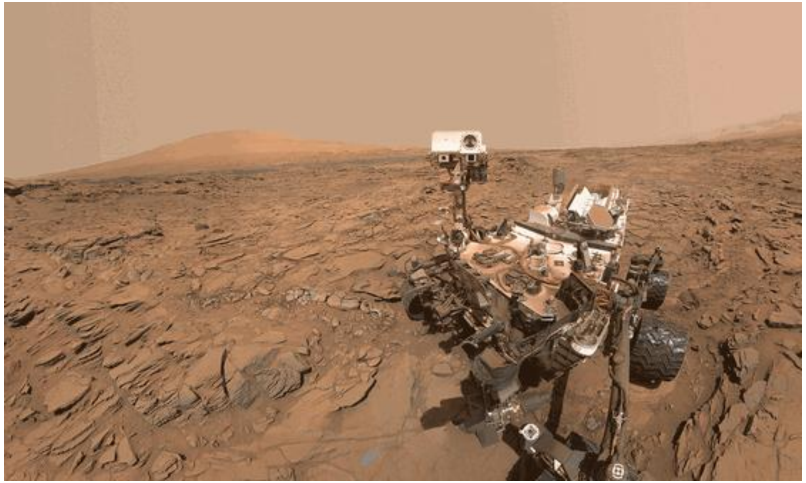
- Equilibrium points due to the combined effects of the Sun and the Earth
- “Halo” orbits
- Stable solar array operation, negligible number of eclipses





On a planet…

- Variable Sun illumination
  - Over one day or over the orbit
  - May depend on atmospheric transmission (e.g.Mars), location
- Possibly very long nights
  - Mars: very similar to Earth
  - Moon: 14 Earth days



Electrical/electronic for space

- But before attacking this topic, let us remember…

$!

Electrical/electronic for space

- **Cost**, **lifetime** and **lack of servicing** immediately point at

  - Control of failures and failures propagation: **failure tolerance**

  - **Careful** and **conservative** design
    - Staying below max capability of all technologies involved!
    - Use only proven technologies which achieved the required minimum readiness level
    - Relying on extensive analyses and test verification
    - Adopting an incremental design and development philosophy

Failure tolerance

- Once launched, (usually) no repair is possible
- The rule:

  "No single component failure shall result in a significant loss of spacecraft operation."

  has a very important consequence for the redundancy, reliability and performance aspects of the electrical design.

  - For example, a controlled reduction of power capability might be allowed after a Single Point Failure or a full Single Point Failure Free (SPFF) approach can be required.
    - for manned missions this requirement is enlarged such that double failures shall be tolerated without impacting the safety of astronauts.

Failure tolerance

- Modular concepts are introduced (hot or cold redundancy schemes)
  - Separation of critical sub-circuits (both mechanically and electrically)
  - Redundant connectors on critical lines

- Protections or specific features are incorporated in each module to avoid failure propagation due to short circuits, over current, over voltage conditions.

Specific Protection!

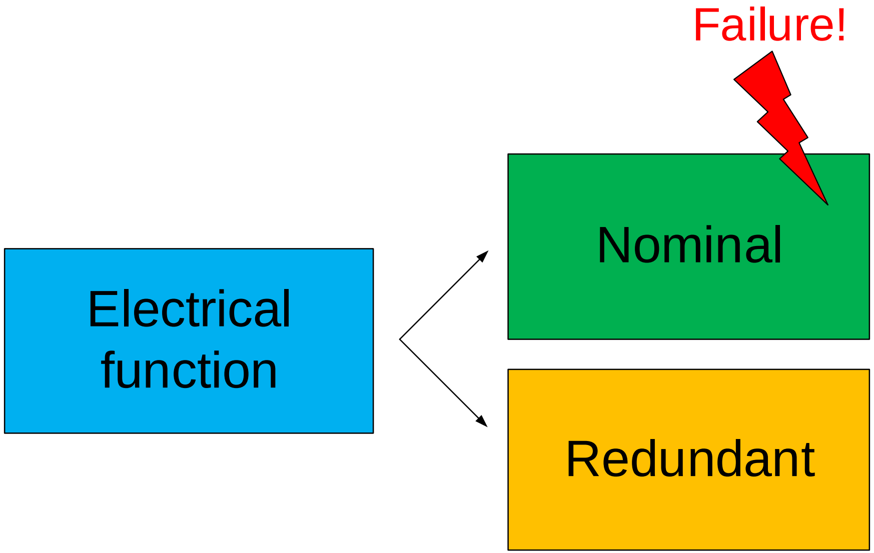
- Autonomy
  - Time-critical failures shall be resolved by on-board intelligence
  - In particular The power system shall not be controlled by an “intelligence” whose circuits are supplied by the power system itself

Redundancy

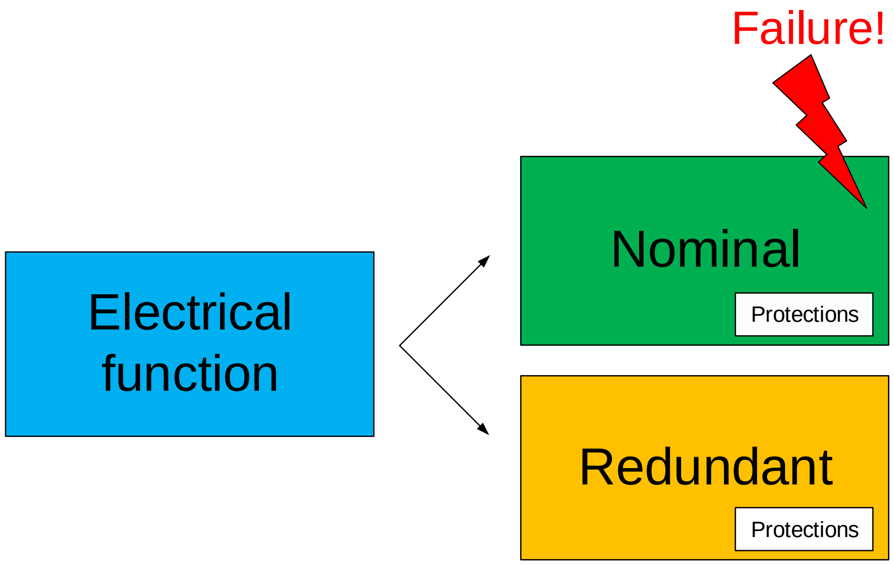
- **Cold** redundancy: the redundant part is inactive and it is activated only when the nominal side fails
- **Hot** redundancy: all redundant part are active at the same time, in case of failure either the failed part is disconnected or it becomes not operative

Specific redundancy!

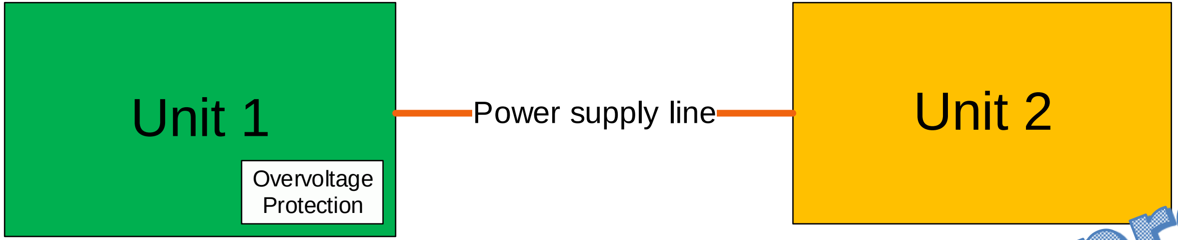
Redundancy



Redundancy & protections!

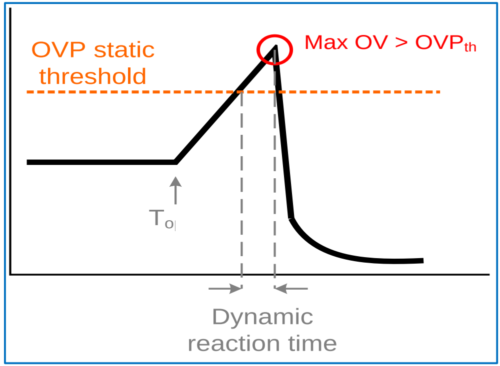


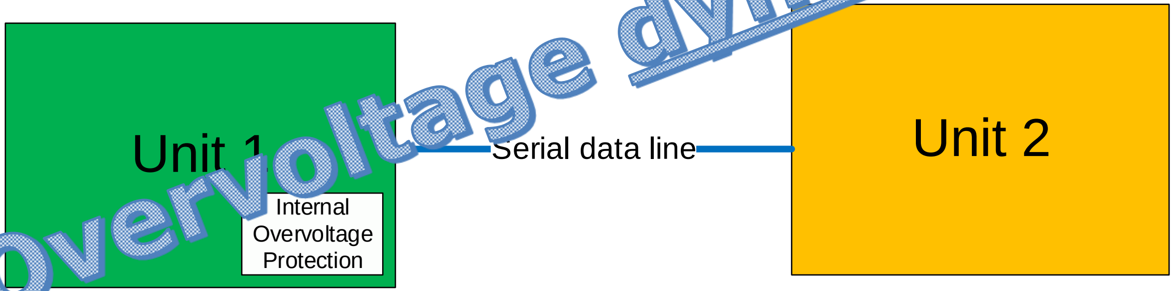
Protections: to avoid failure propagation!



Overvoltage protection in unit 1 avoids that in case of failures unit 2 sees at power supply line inputs voltage in excess of its rated capability.

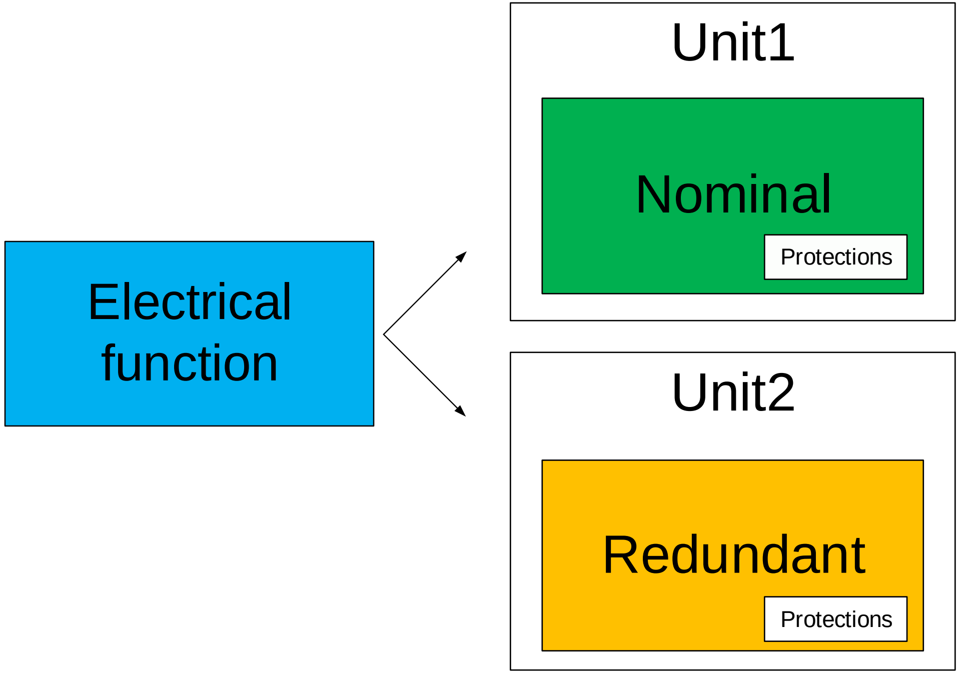
Overvoltage dynamic protection!



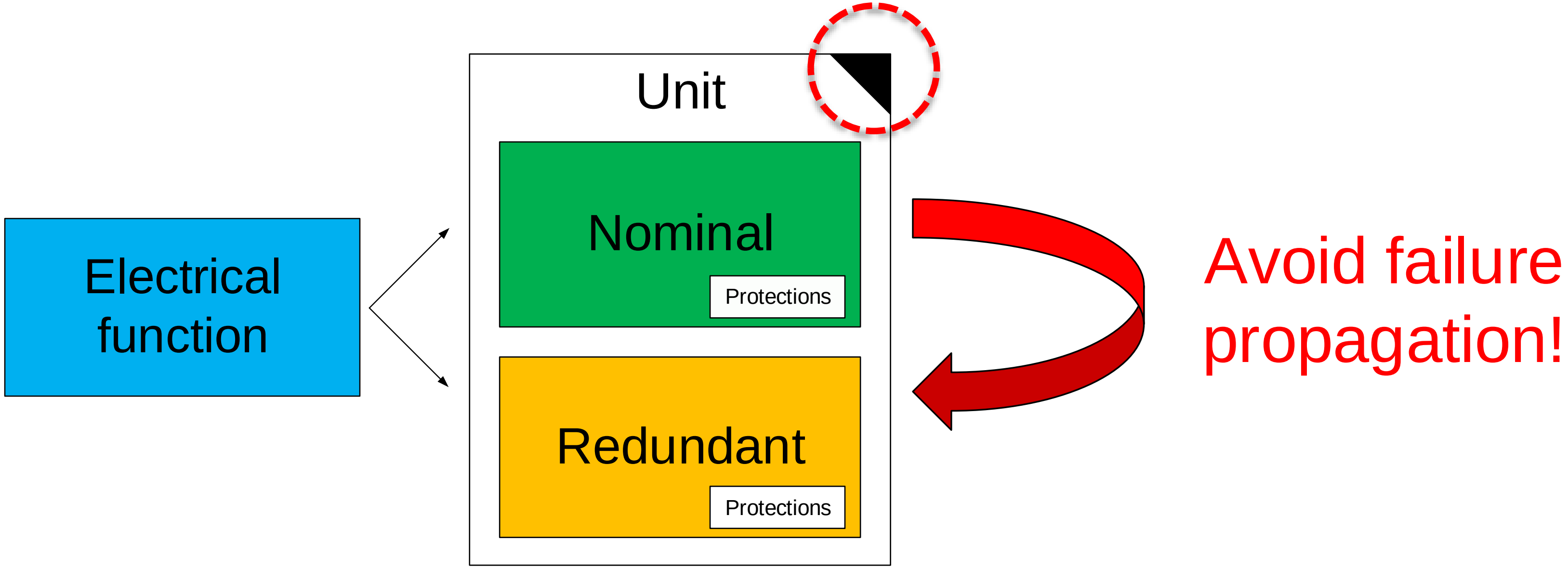


Internal overvoltage protection in unit 1 avoids that in case of failures unit 2 see <u>at serial data line</u> input voltage in excess of its rated capability

Redundancy in different units

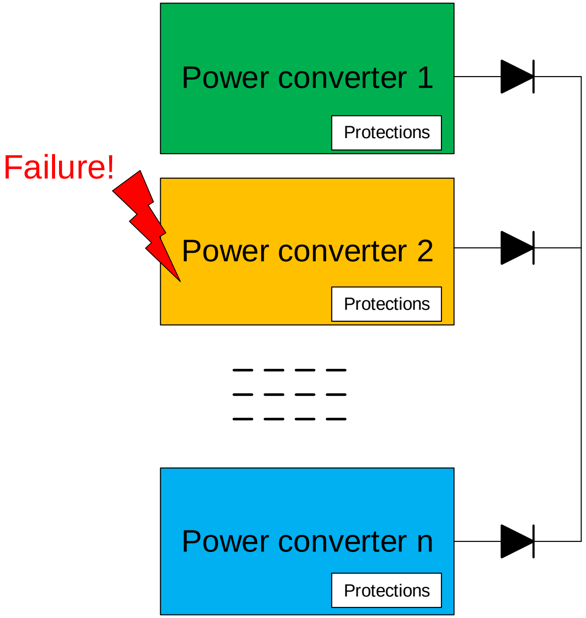


Redundancy in the same unit

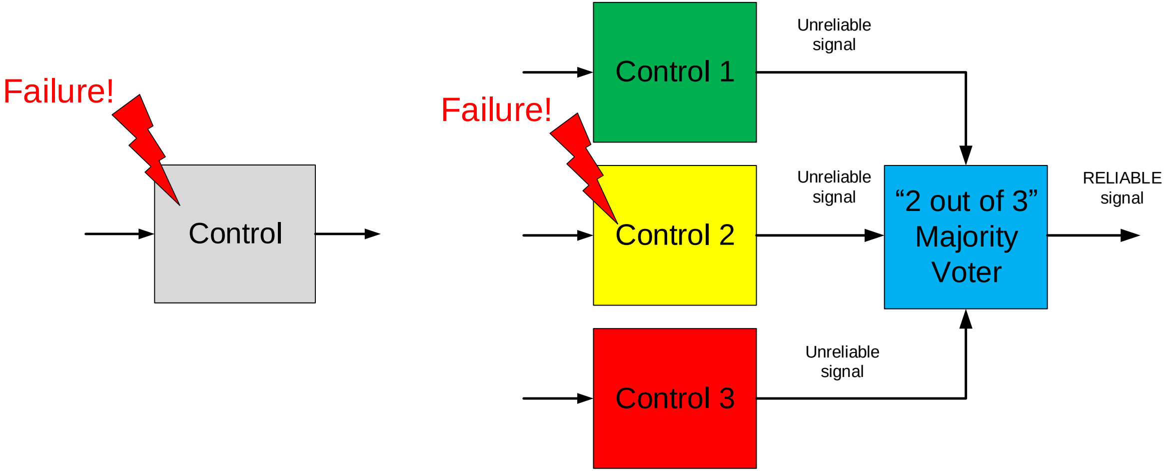


Hot Redundancy examples

Power example



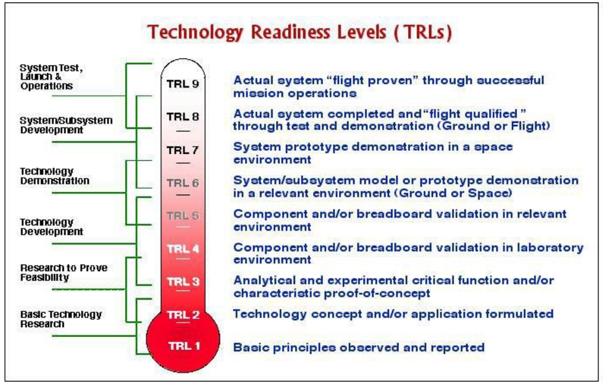
Signal example



- Analogue or digital signals

Electrical/electronic design: careful & conservative

- Staying below max capability of all technologies involved!
  - For EEE components, derating concept: absolute Maximum ratings * factor of utilization (< 1)
    - Voltage, Current, Power, Temperature
  - Derating rules: see ECSS-Q-ST-30-11A, Derating - EEE components
- Use only proven technologies which achieved the required minimum readiness level
  - For TRLs, *see ISO standard 16290 and ECSS-E-HB-11A,Technology Readiness Level (TRL) guidelines*



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carefu

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  - For EEE utilizati
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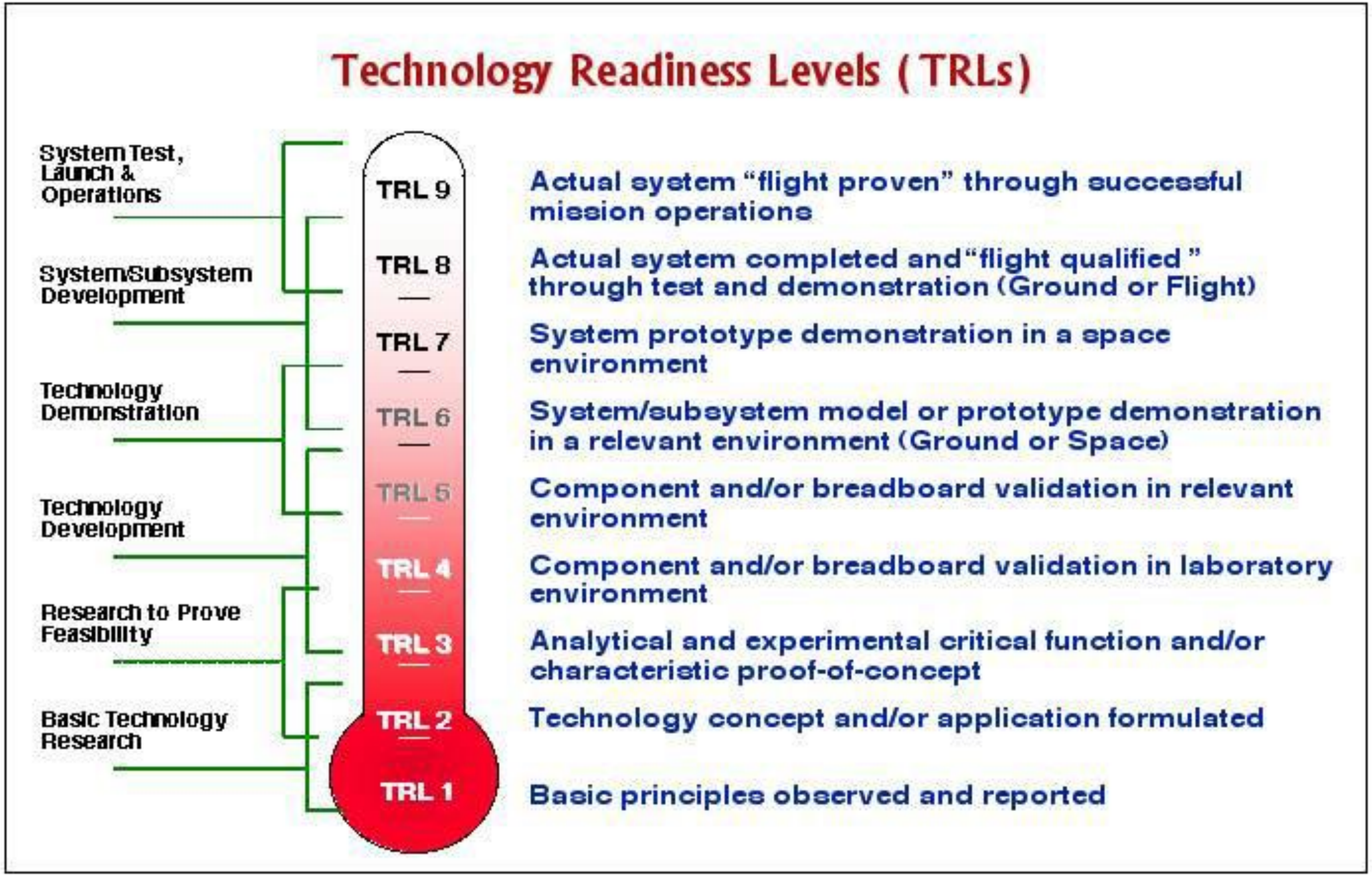


Again on Derating

In Space, parts are used far below their rating.

Typical requirements are to use semiconductors at:

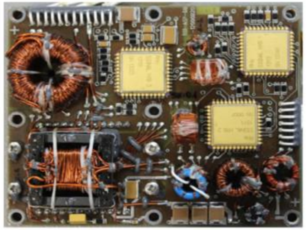
- < 80% of rated voltage
- < 75% of rated current
- < 60% of rated power
- junction temperatures < 110°C (rating is at 150°C / 175°C)

Derating brings to a necessary under- utilisation of components

- Low performance
- High volume/mass

6 W / cm^3

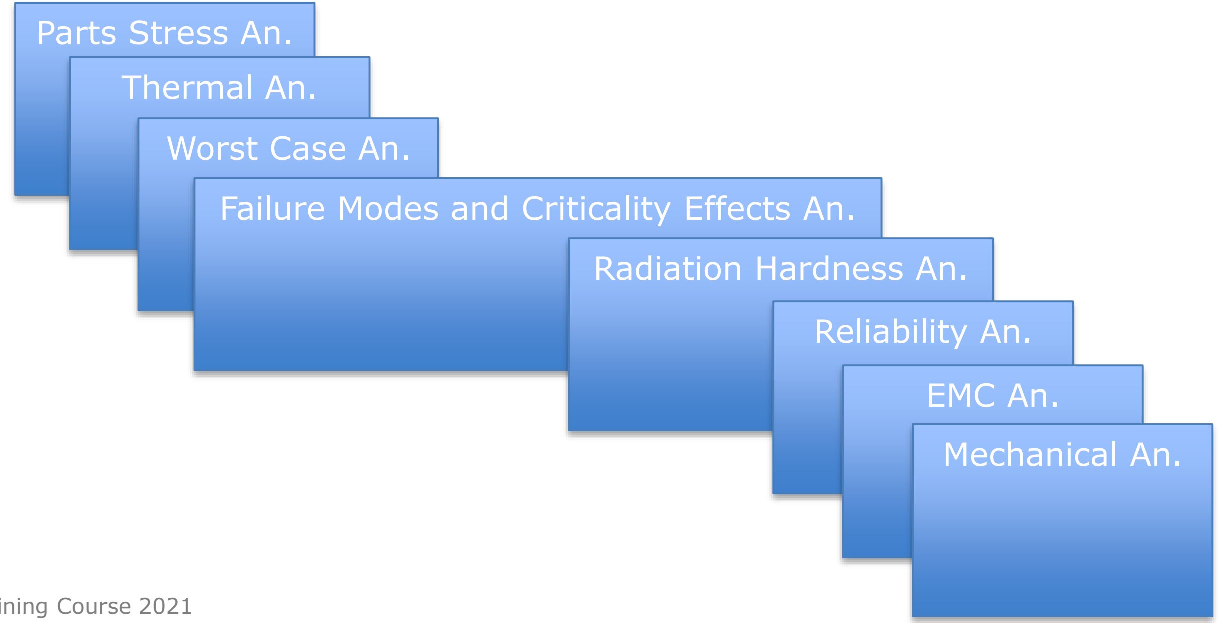
0.075 W / cm^3



95 x 79 x 23.5 mm

Electrical/electronic design: careful & conservative

- Analyses: PSA, THA*, WCA, FMECA*, RHA, RA*, EMCA*, MA*… normally done at equipment level (apart * that are also done at subsystem or at satellite level)



Electrical/electronic design: careful & conservative

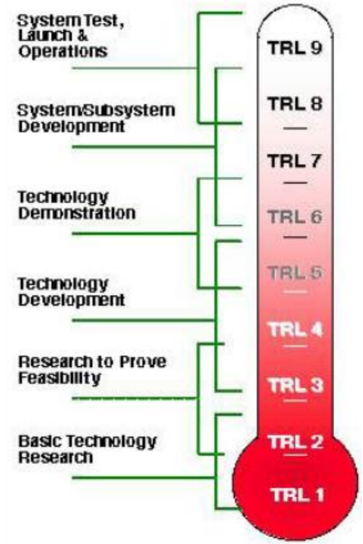
- Test verification at unit, subsystem, system (e.g. satellite) level

- Test verification
  - functional, performance
  - in ambient and in vacuum
  - Ambient temperature and at temperature extremes, according to defined cycles

  - Electro-Magnetic Compatibility (EMC) tests
  - Vibration, shock tests with functional, performance tests before and after, and also during vibration if necessary

Electrical/electronic design: careful & conservative

- Adopting an incremental design and development philosophy
  - The TRL of the sub-part shall be higher than the part the sub-part will belong to



Electrical component — High TRL

↓

Equipment

↓

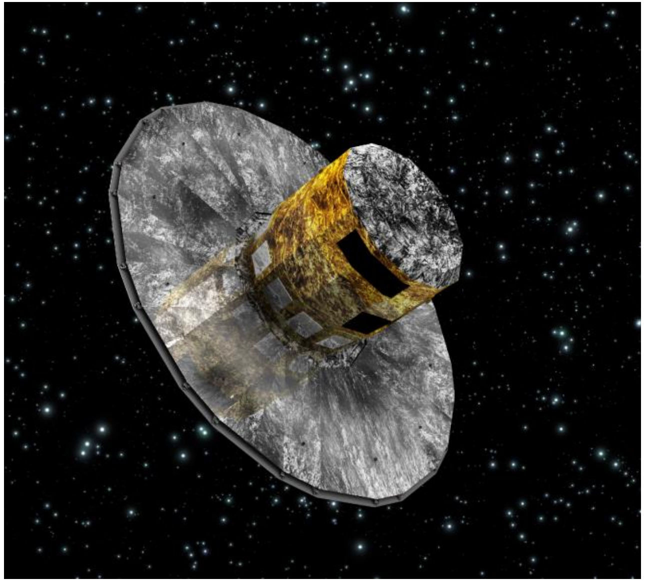
Subsystem — Low TRL

Summary so far...

	Space application	Terrestrial application
Radiation	Total Dose Irradiation, Displacement Damage and Single Event Effects are key drivers. Radiation hardened or at least radiation tolerant components shall be used.	Total dose irradiation / Displacement damage are negligible and Single Event Effects are rare due to the screen by the Earth atmosphere
Thermal dissipation	Conduction/radiation only. A component running at 40°C in a terrestrial environment might produce case temperatures exceeding 200°C in vacuum.	Convection / forced air is commonly used to keep electronic components under acceptable temperatures
Lifetime	Typically, very extended (up to 18 years operation in telecom satellites)	Typically limited (few years design goal)
Reliability/failure tolerance	Generally, no repair capability during lifetime Important de-rating needs (ECSS-Q-30-11C)	Generally repair is easily feasible: change parts, components
Main design drivers for power electronics	Design mainly driven by reliability and performances (efficiency, mass, volume)	Main driver are usually cost and compactness.

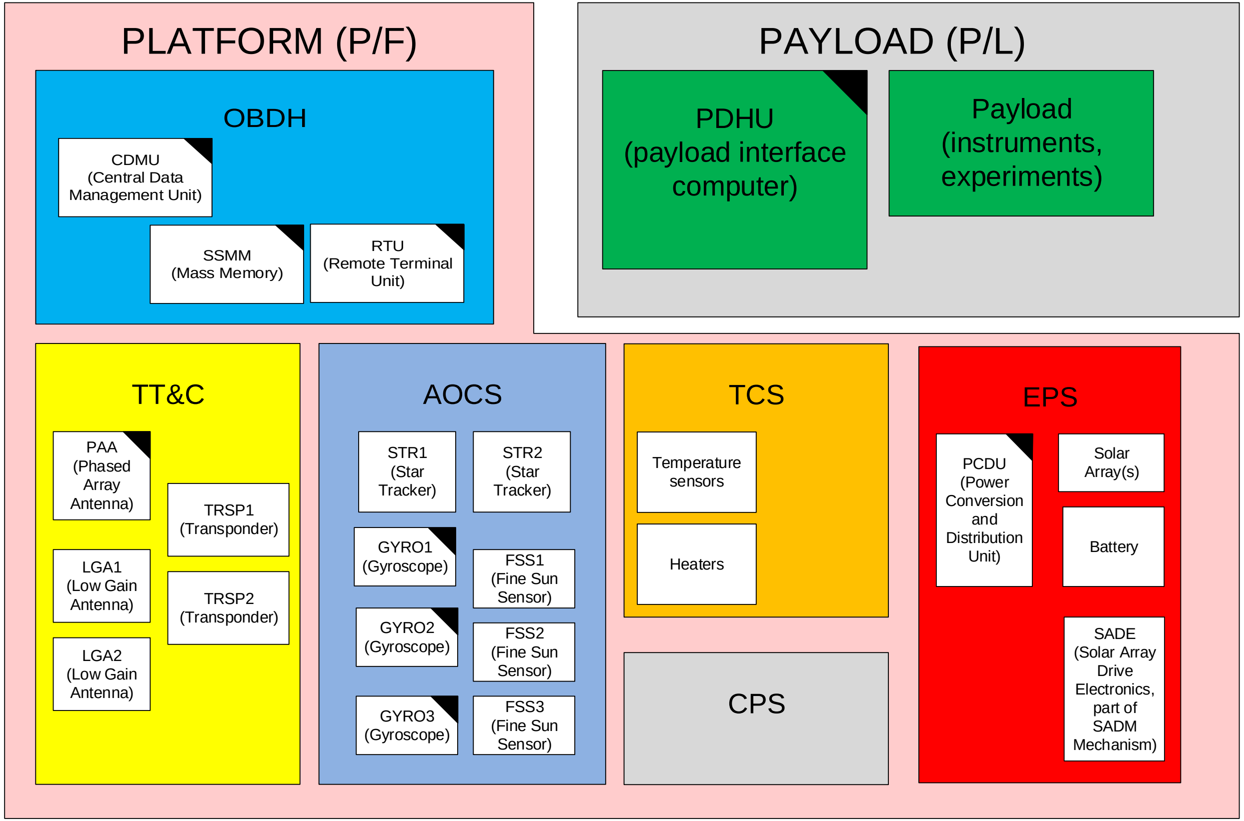
Electrical/electronic architecture

- We will take inspiration from one scientific satellite launched in 2013… **GAIA**



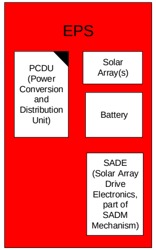
Gaia is a mission to chart a three-dimensional map of our Galaxy, the Milky Way, in the process revealing the composition, formation and evolution of the Galaxy.
It will provide unprecedented positional measurements for about one billion stars – about 1 per cent of the Galactic stellar population – in our Galaxy and Local Group, together with radial velocity measurements for the brightest 150 million objects.
Combined with astrophysical information for each star, provided by on-board multi-colour photometry, these data will have the precision necessary to quantify the early formation, and subsequent dynamical, chemical and star formation evolution of the Milky Way Galaxy.

Electrical/electronic architecture



Power systems architecture

Typical configuration (regulated bus - more extensive presentation on Power Management and Distribution section -):



Power Generation

- Provide the power required by the spacecraft from launch until the mission completion
- Usually a solar array for near sun satellite applications, sometimes a Radio-Isotope Thermoelectric generator (RTG) for spacecrafts that have to operate at great distances from the sun

Power Conditioning

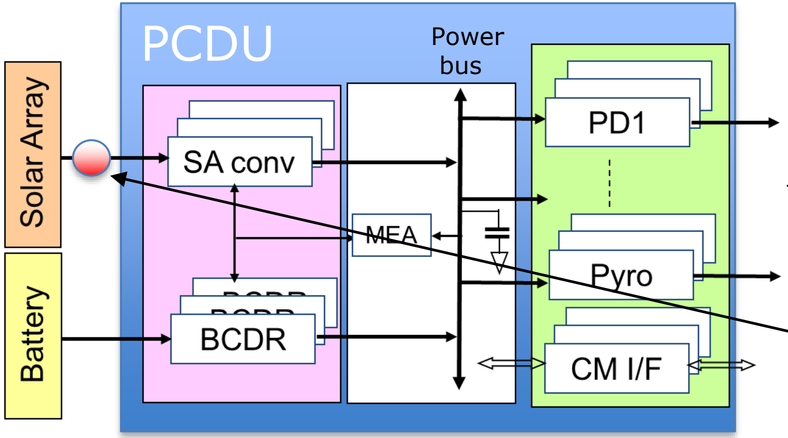
- Provide autonomous control of the power generation and energy storage
- Condition the power bus to a defined voltage range
- Provide command and telemetry capability for health check and control

Power Distribution

- Distribute the power to all the spacecraft loads
- Provide load switching capabilities
- Protect the power lines to avoid failure propagation between loads and EPS

Energy Storage

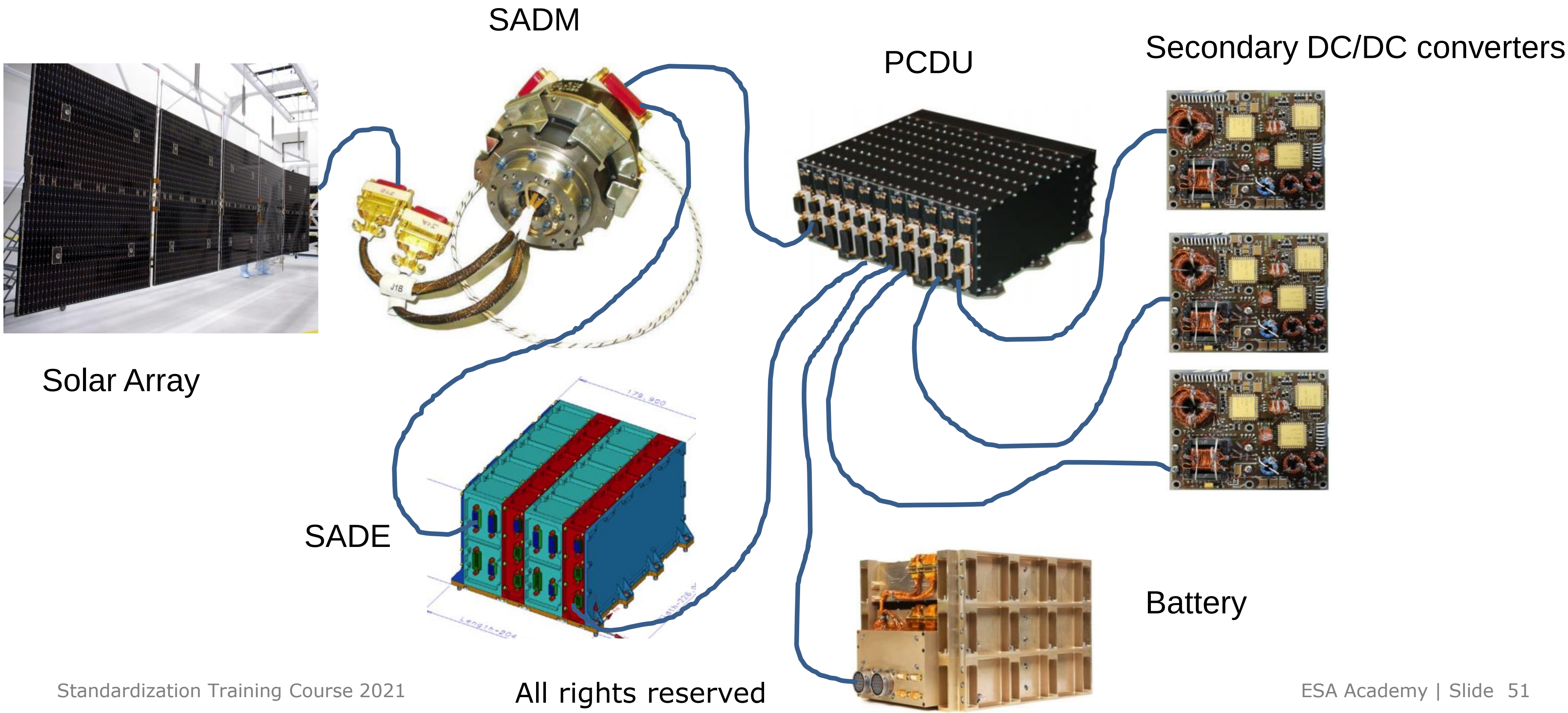
- Store energy during sunlight
- Ensure a continuous source of electrical power during eclipses
- Complement the power source as needed (e.g. peak power demands, manoeuvres)



SADE

(Solar Array Drive Electronics, part of SADM Mechanism)

Power systems architecture – a real view

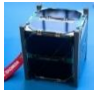
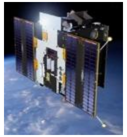


Power systems requirements

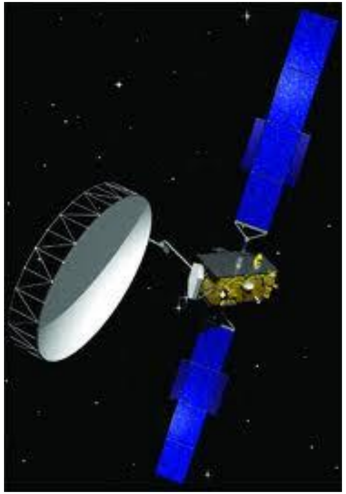
- Power and energy range
- Equipment Power (voltage) needs
- The main bus

Power and energy range

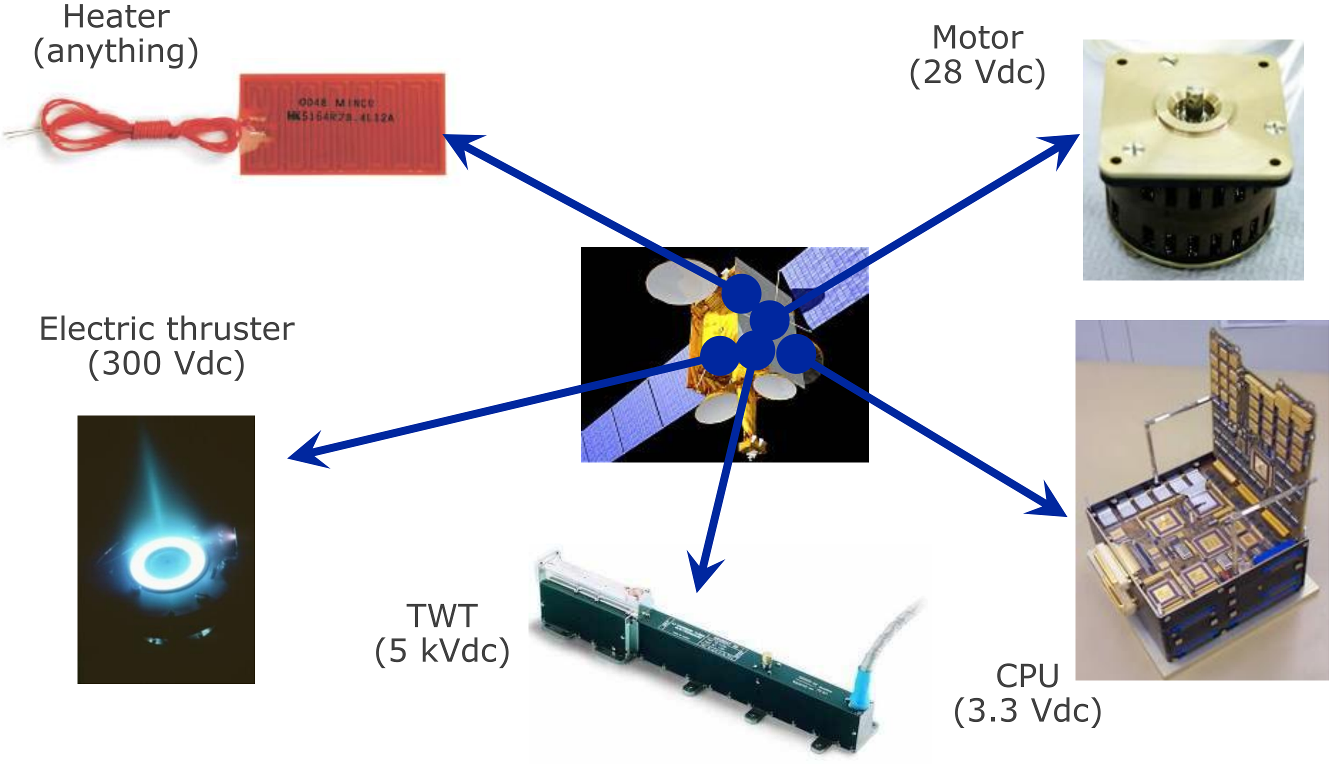
- Power: from watts to tens of kW


- Energy: from < 1kWh to tens of GWhs



Equipment power (voltage) needs



The main bus

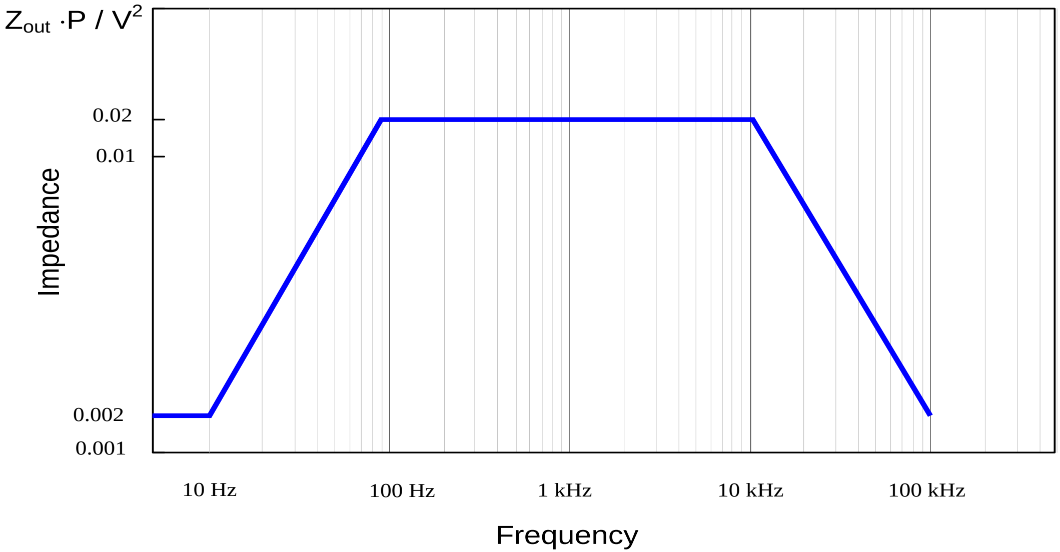
- The typical configuration of satellites over some hundred W's is to generate a reliable, single main bus line and distribute it to the various users on different protected lines
- It might be regulated (within ±1% or less) or unregulated (voltage range depending on battery characteristic)
- Typical regulated or unregulated voltages depend on power needs:
  - From about 500W to 1.5kW, 28V bus is used
  - Up to 8kW or so, 50V bus is used
  - For higher power, 100V bus is used.
- The rationale? => "Q&A" !!!

The main bus

- Typical regulated or unregulated voltages depend on power needs:
  - From about 500W to 1.5kW, 28V bus is used
  - Up to 8kW or so, 50V bus is used
  - For higher power, 100V bus is used.
- The rationale?
  - It depends on bus impedance considerations: it is difficult to design output impedance below 10 milliohm without an unwanted effect of the intrinsic connections and components resistance.
  - For the design of a bus with 10 milliohm output impedance such that a 50 % load modulation induces a (required) 1 % voltage change maximum:
    $$0,5\ P/U \times 0,01 < 0,01U \text{ which means } P < U^2/0,5$$
  - Thus for U = 28 V, P < 1,57 kW; U = 50 V, P < 5 kW; U = 100 V, P < 20 kW.
  - In practice, at 50 V for example, higher power has been used on telecom spacecraft buses, because the 1 % voltage change referred to a lower load change of 20 % to 30 % instead of 50 %. See ECSS-E-ST-20C req. 5.7.2g.

Main bus impedance

- The fully regulated bus impedance shall be below the following mask at the point of regulation



Power System Transients & Stability

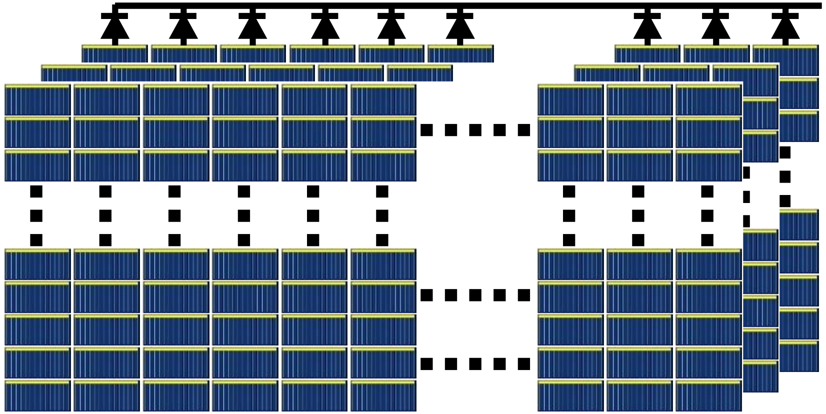
Very demanding!!!

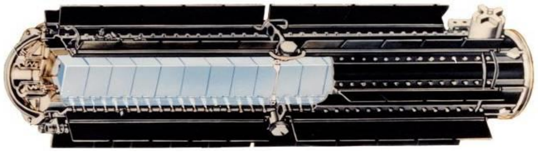
- Dynamic Performance (Main bus level)
  - 0.5% static regulation
  - 1% overshoot for a 50% load step
  - 5% for any source and load transient

- Stability
  - 50^0 phase margin/ 6dB gain margin for primary power system (main bus regulation) and for secondary power system (all other regulators/converters in the satellite)
  - for primary power system, the previous stability requirement shall be met after any after any single failure

Power generation

Solar Array (SA)





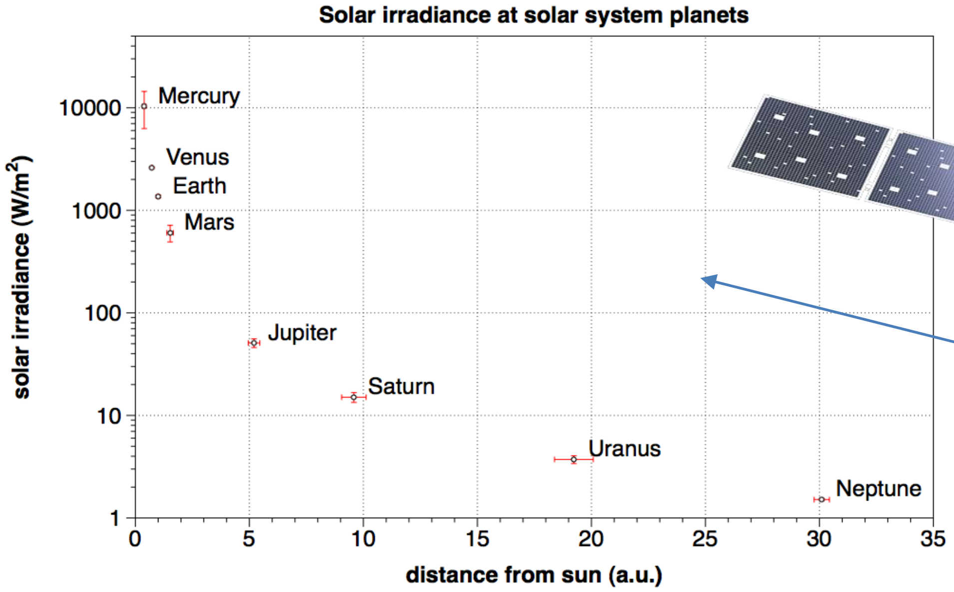
Radioisotope Thermoelectric Generator (RTG)

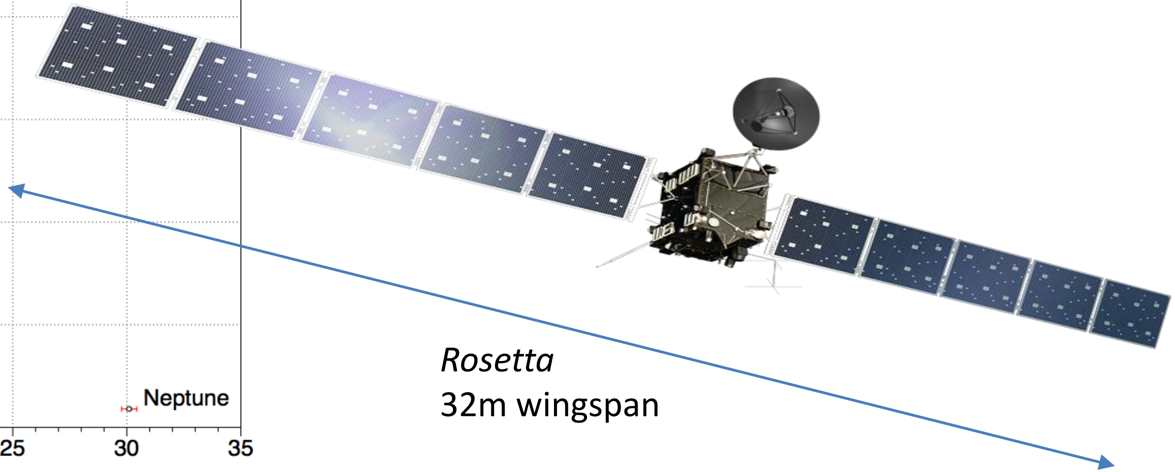
Power generation

Solar or nuclear power? Where is the boundary?

=> Outer solar system missions, due to the $1/r^2$ weakening of the solar flux.

- (ESA's *Rosetta* spacecraft required 64 m^2 of solar array to just *survive* at ~5 a.u.)





Rosetta
32m wingspan

Power generation – Solar arrays

Rosetta in orbit around comet Churyumov–Gerasimenko

What are solar cells ?

- A solar cell …

- Absorbs light (‘photons’ …. as many photons as possible)
- Uses the energy of each photon to excite electrons
- Harvests the energy of the electrons as an electric current

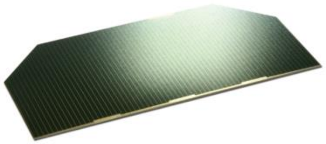


Image courtesy of Azur Space

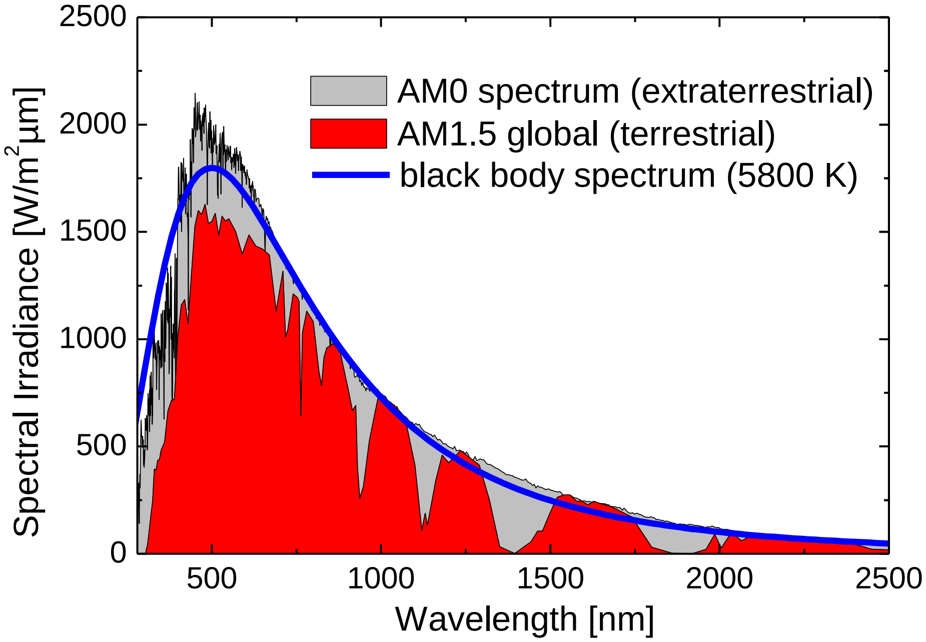
The solar spectrum

Total power from the sun in earth orbit :

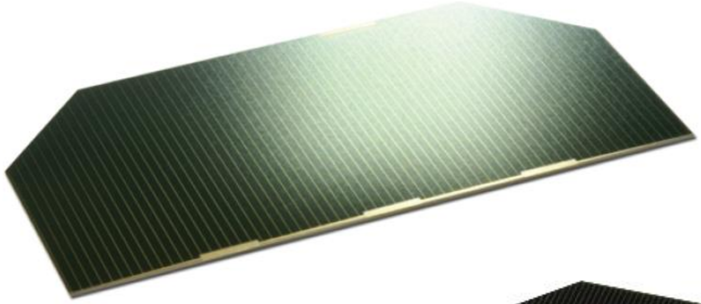
⇒ AM: air mass

⇒ solar constant

1367 W/m²



Some approximate numbers

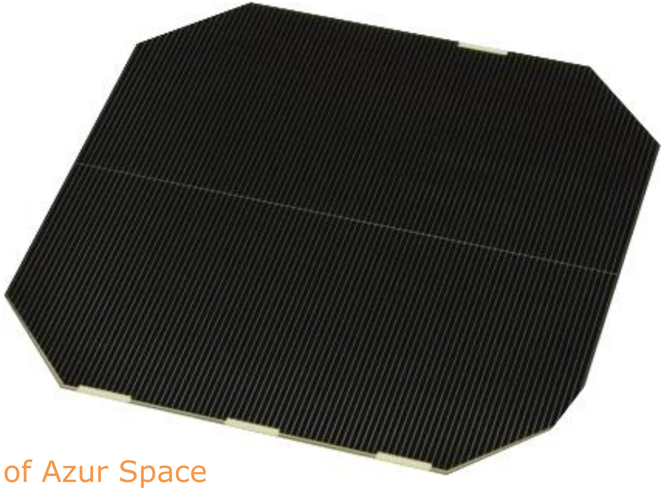


„Rough order of magnitude“ numbers :

30 $cm^2 \approx$ 1 W at 50C in space around the Earth

=> 70W/kg, 300W/m2 at array level at launch

Current array designs roughly cover the range up to 20kW



Images courtesy of Azur Space

Electrical characteristic of a solar cell

Convention:
I-V curves are mirrored at the x-axis

Power curve:

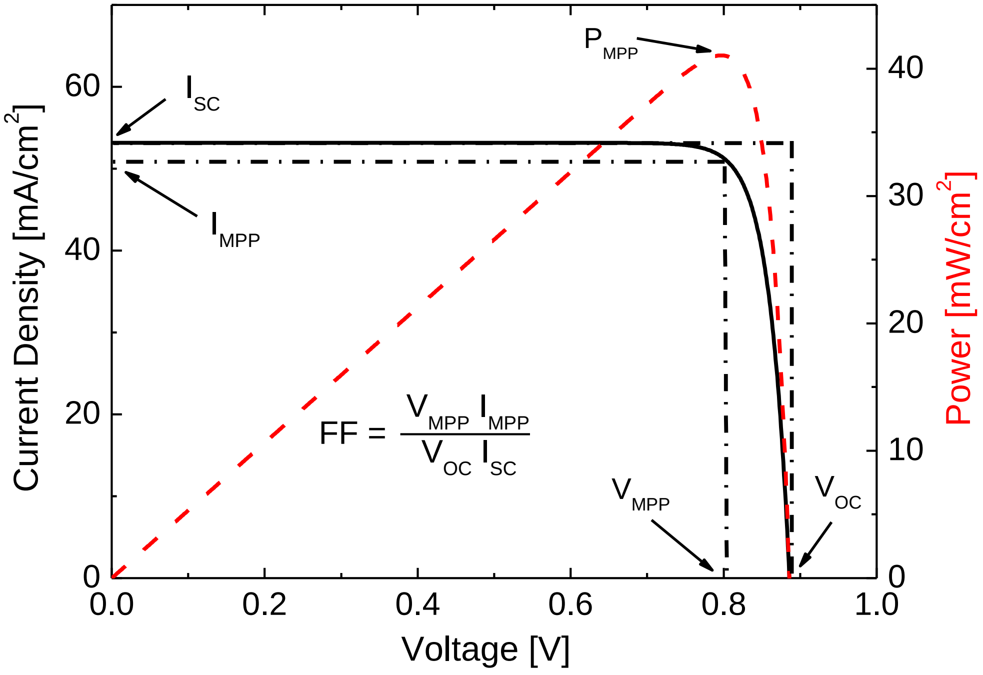
$$P = I \cdot V$$

I_{SC}: short circuit current

V_{OC}: open circuit voltage

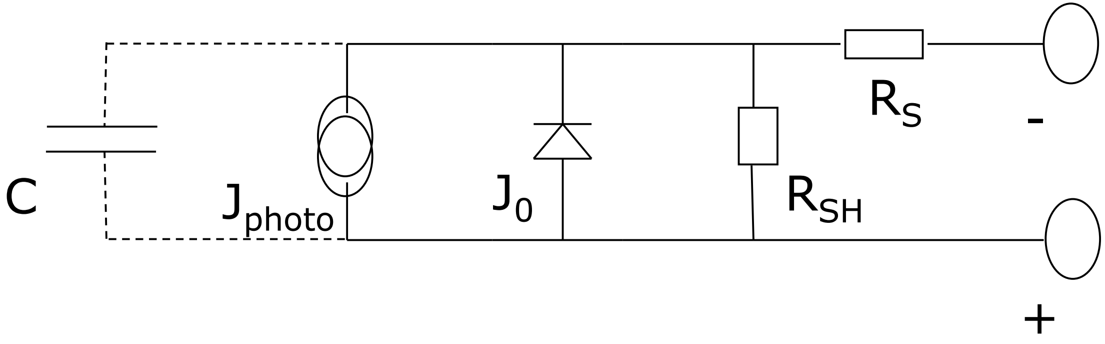
P_{MPP}: maximum power point

FF: Fill factor



Electrical schematic of a solar cell

NB! <u>Single</u> junction cell



$$J(V) = J_0\left[\exp\left(\frac{qV}{k_B T}\right) - 1\right] - J_{photo}$$

Achievable efficiencies – single junction

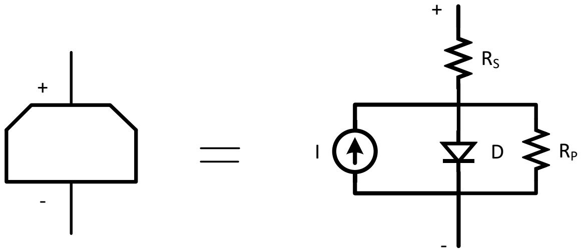
Theoretical efficiencies :

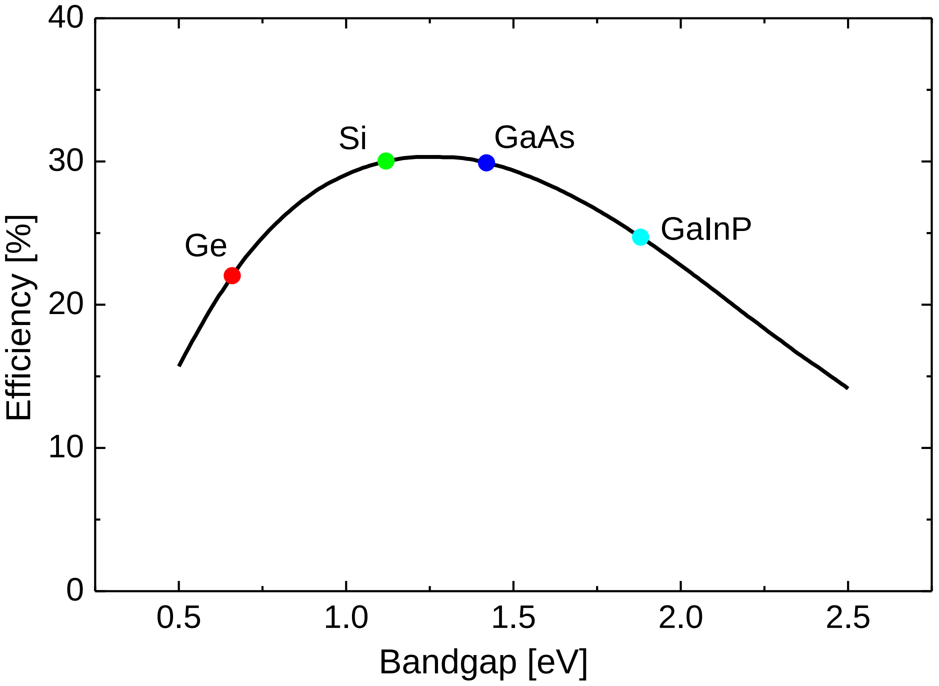
- Ge: 22 %
- Si: 30 %
- GaAs: 29.9 %
- GaInP: 24.7 %

Practical Efficiencies :

- Si: 21.1 % *
- GaAs: 22.1 % *

Equivalent electrical diagram of a single junction cell



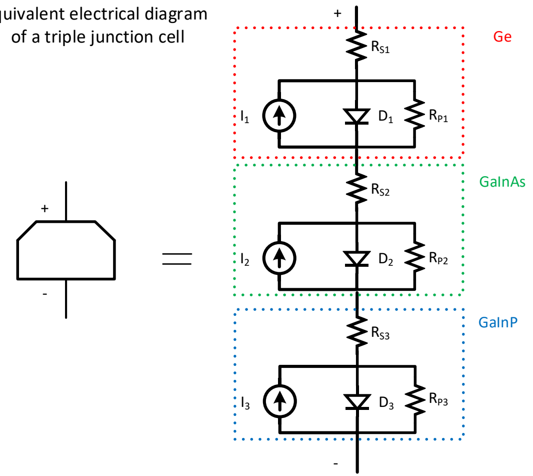


* S. Bailey, „Space and Terrestrial Photovoltaics: Synergy and Diversity", Progress in Photovoltaics, 2002, pp. 2329-2333

‘multi-junction’ solar cells

- It’s possible to achieve higher efficiencies by combining several solar cells ‘one on top of the other’
- Individual junctions have to be connected by a tunnel diode or other external connection

Equivalent electrical diagram of a triple junction cell



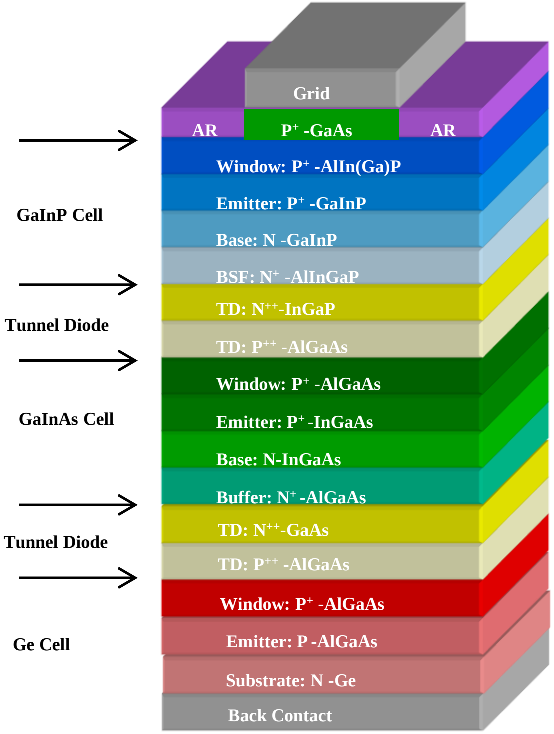
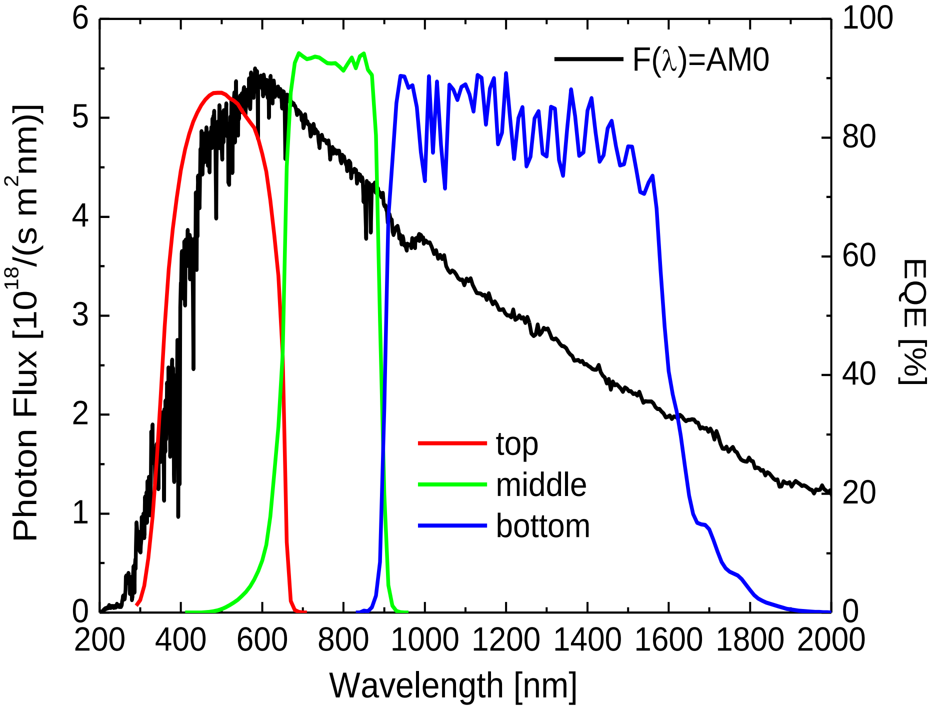
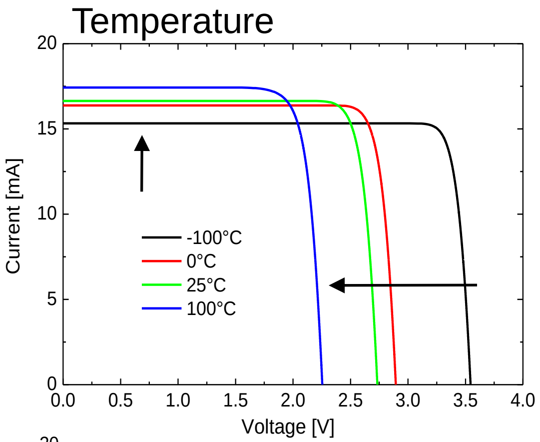


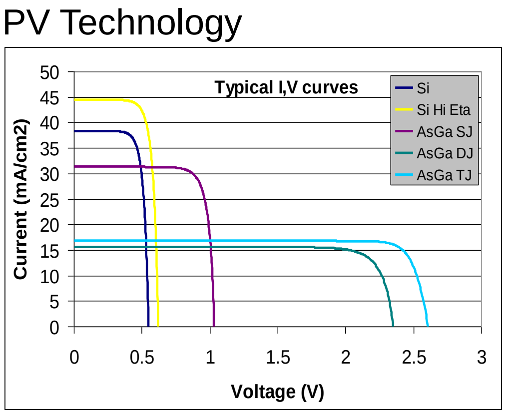
Image courtesy of Cesi

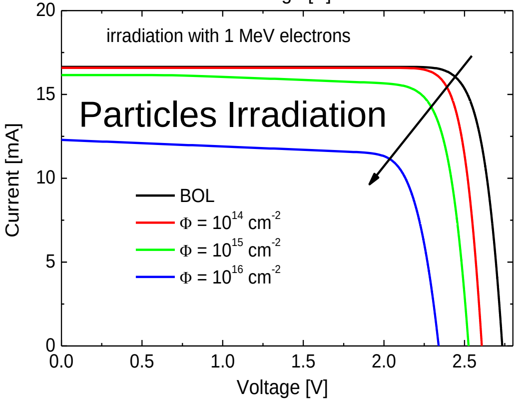
Current matching

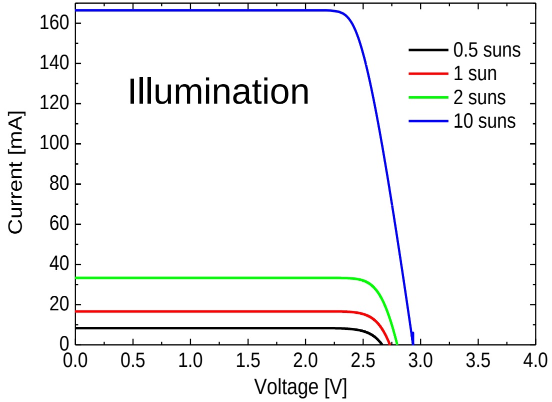


Solar cells & environment

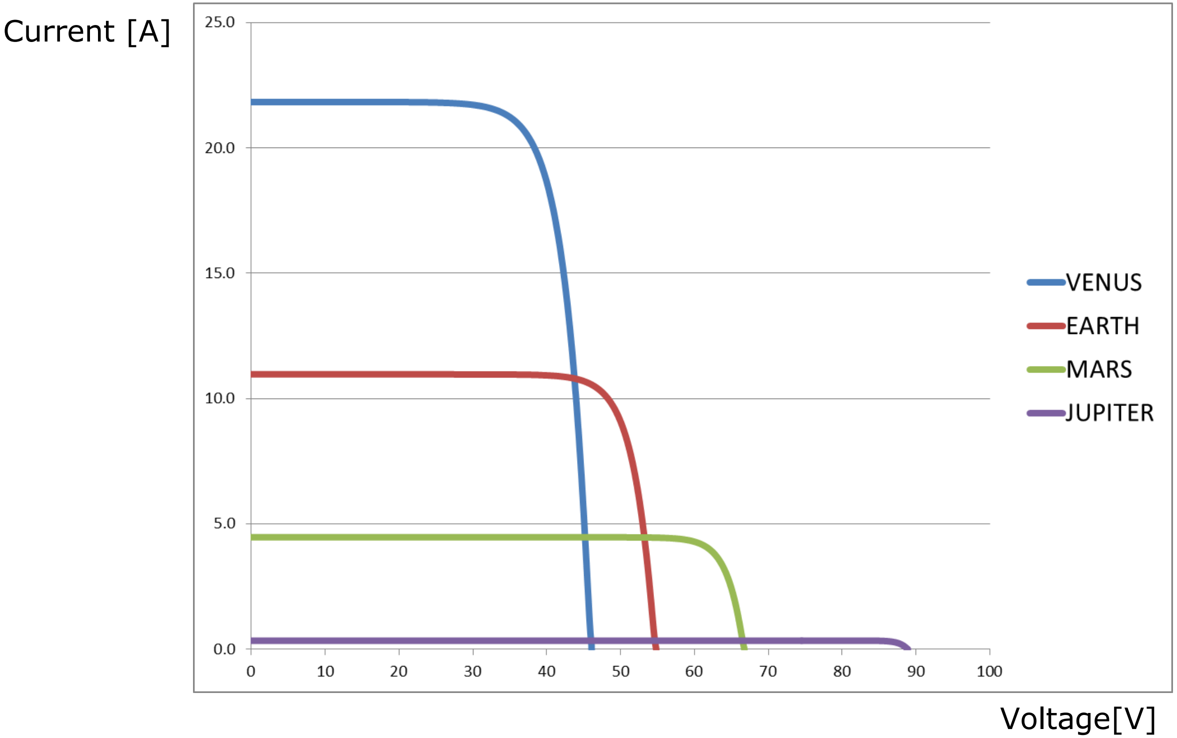








Same solar array at different sun distances



State-of-the-art 3J cell

Material: GaInP/GaInAs/Ge
2 cells on 4" wafer
Dimensions: 4 x 8 cm^2 with cropped corners (30.18 cm2)

Thickness: 150 ± 20 μm

Area: 30.18 cm^2

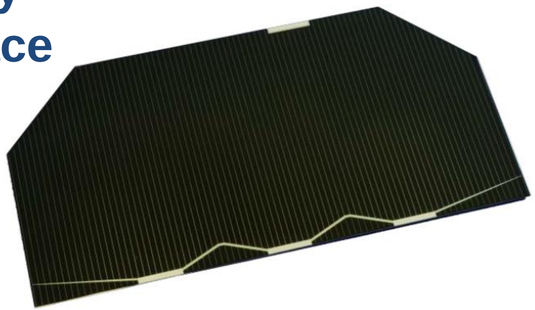
av. weight: $\leq$ 86 mg/cm^2

av. efficiency (BOL): 29.5 %

av. efficiency (EOL at 1E15e/cm2-1MeV eq.): 26.5 %

$\Rightarrow$high remaining factor of 0.89 (after 15 years in GEO)

3G30% by Azur Space

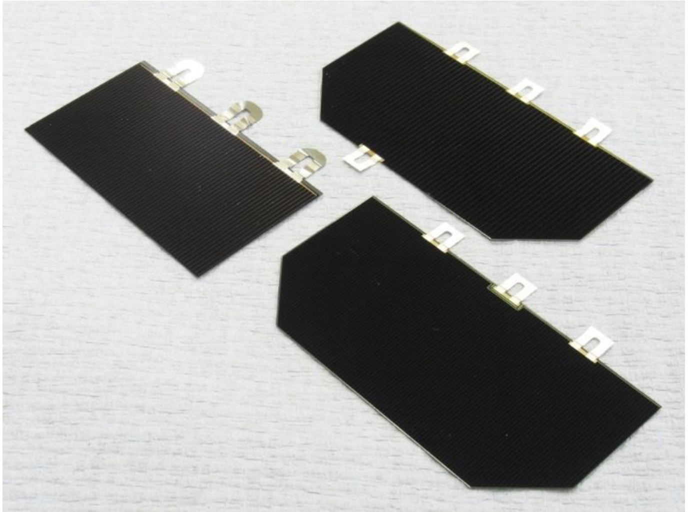


29.5 % · 1367 W/m2 · 30.18 cm2 ≈ **1.2 W**

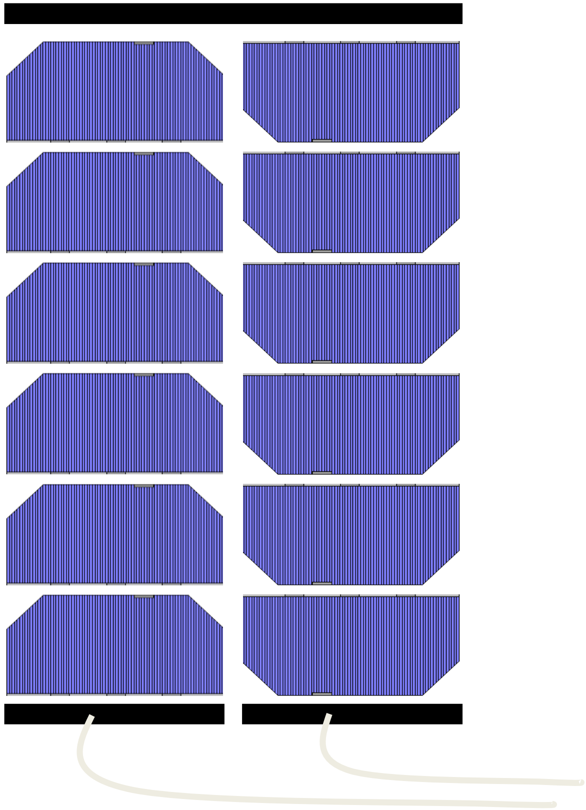
From wafers to solar cell assemblies

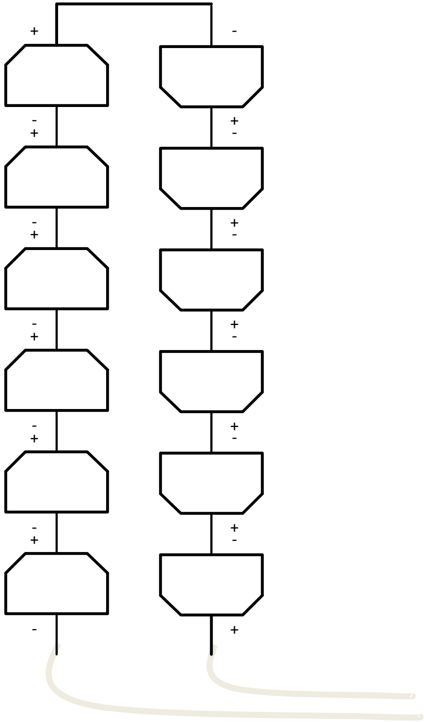
Cells are processed with a grid contact to allow electrical connection

Connection tabs and coverglass are applied to make a solar cell assembly

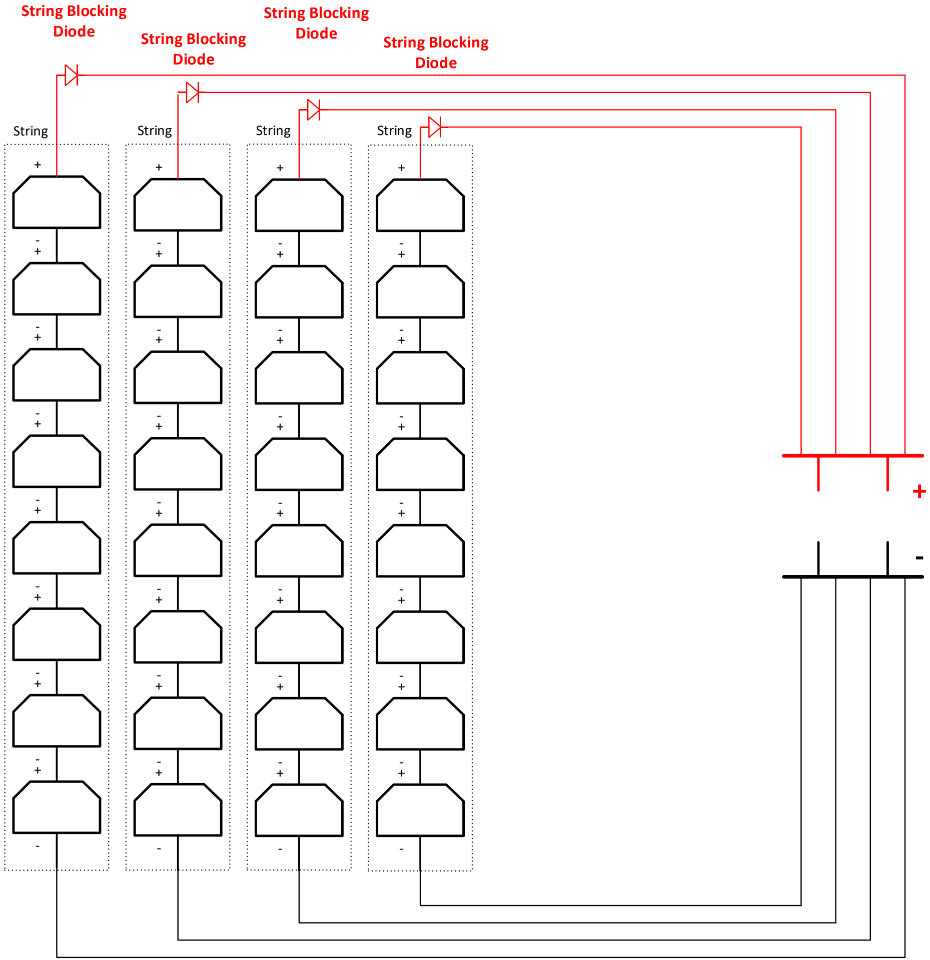


Cells are connected in series to make strings





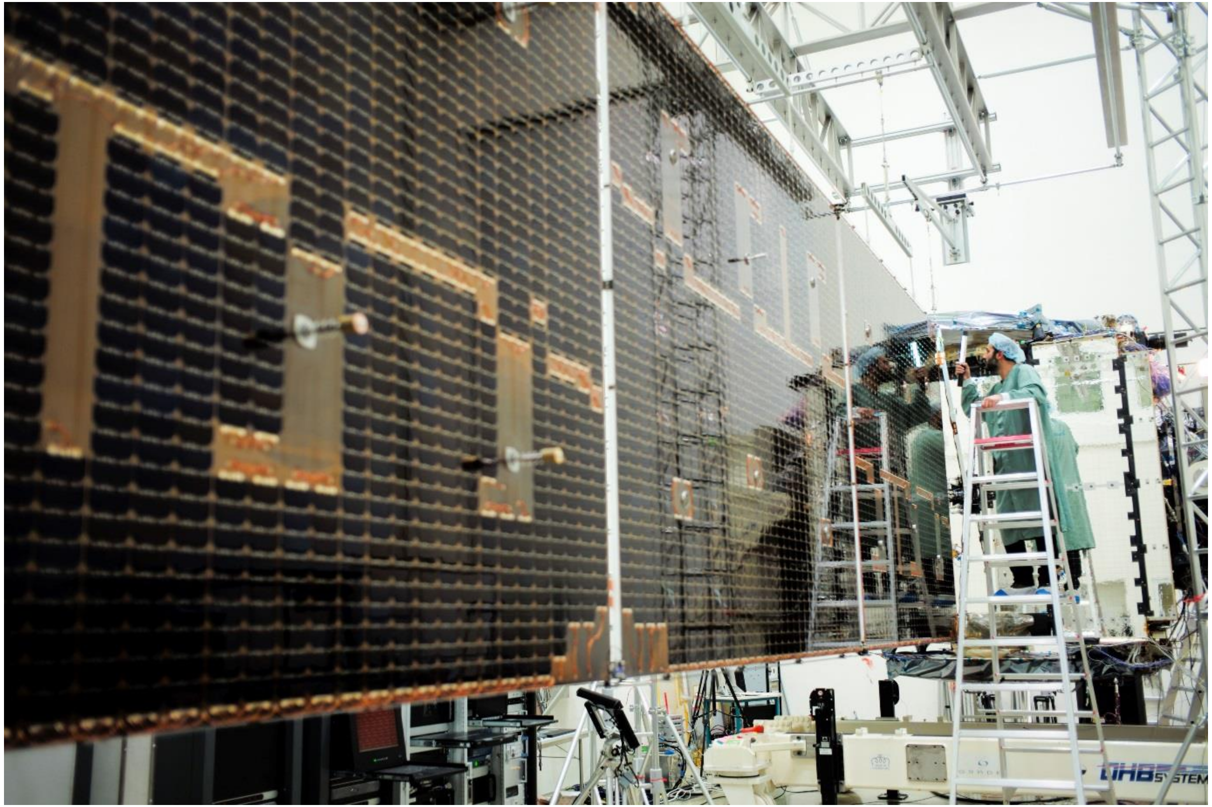
Strings are connected in parallel to make sections



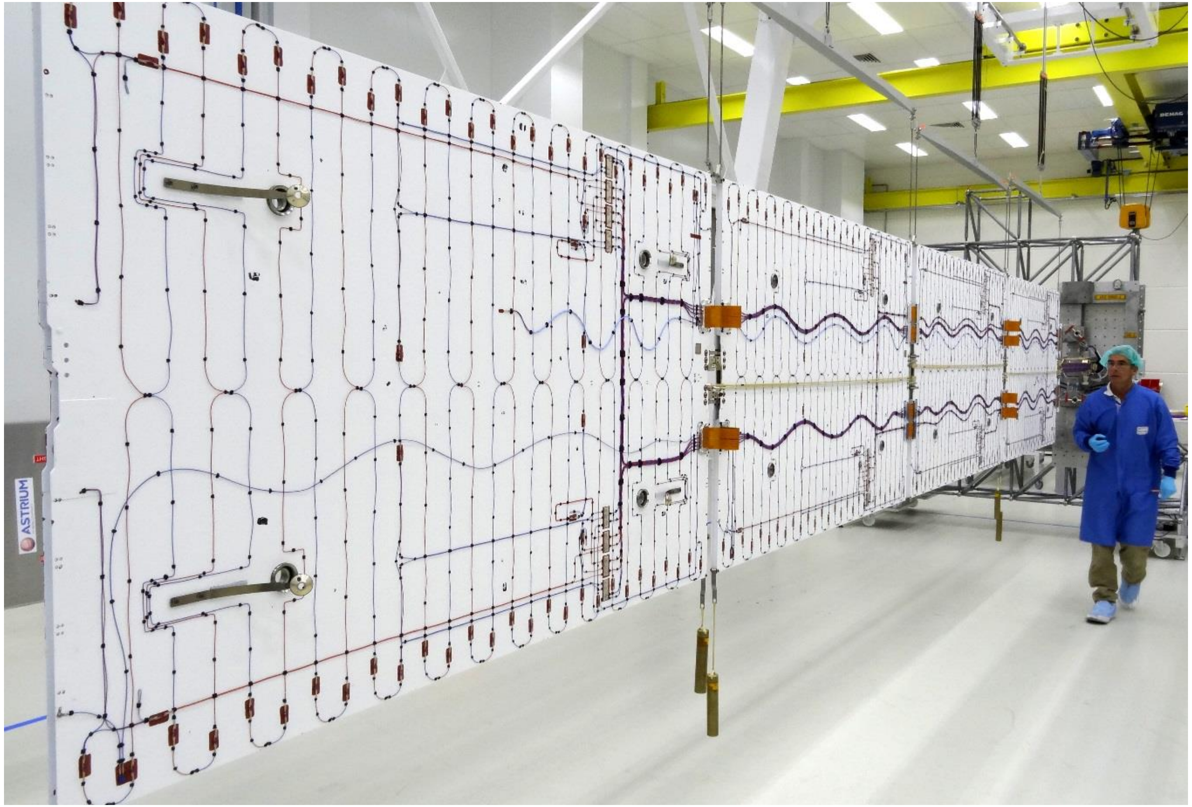
Specific redundancy!

Additional strings are added for redundancy to cover for all possible failures / loss factors (inherent failures, micrometeoroids, ESD effects, etc)

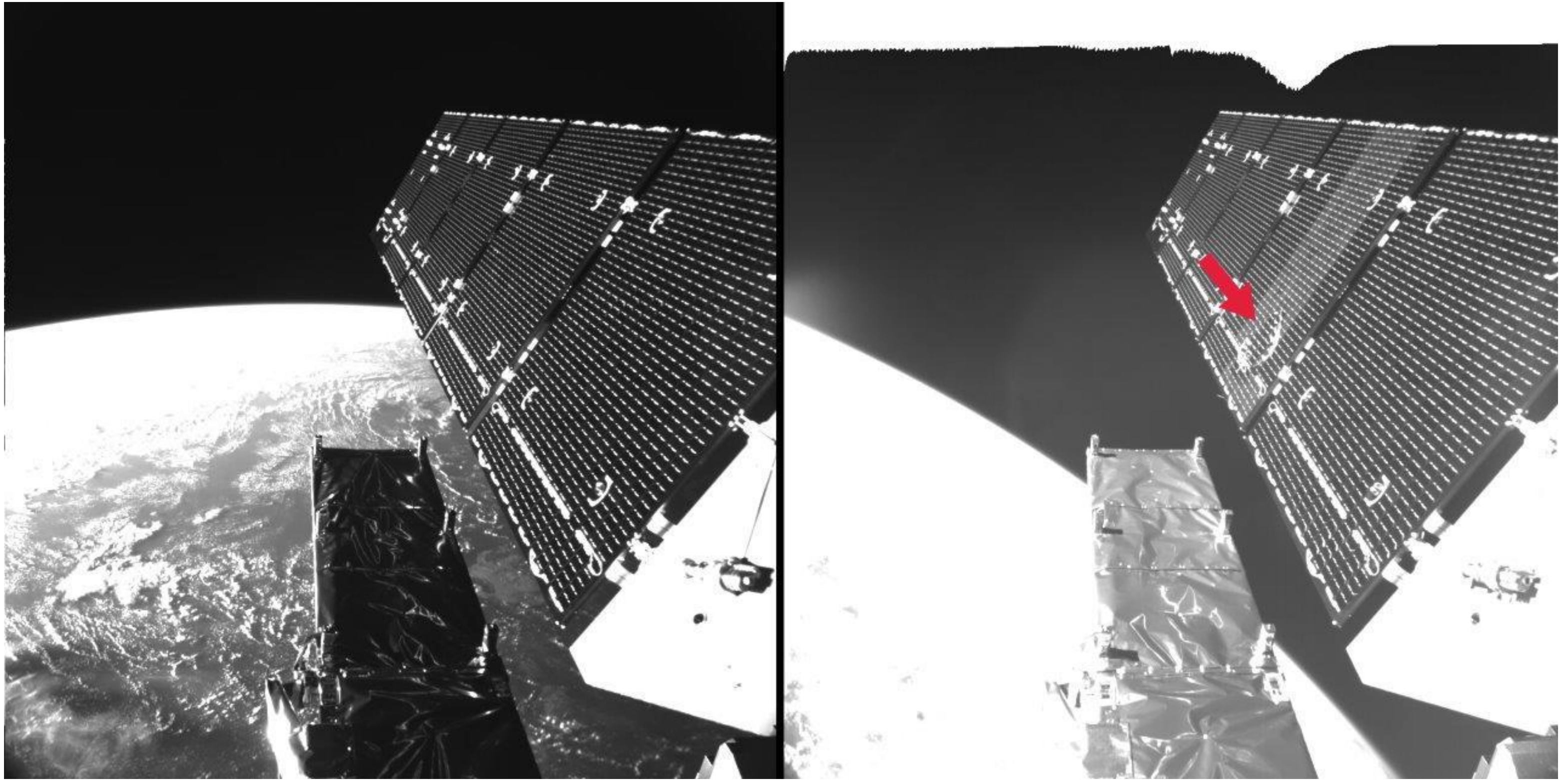
Small GEO solar array front



Rear side of ATV solar array



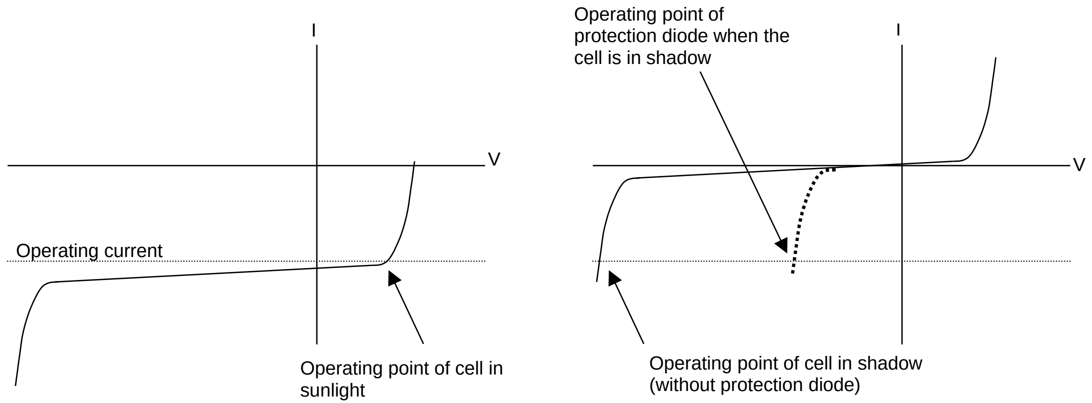
Sentinel 1A solar array, micrometeorite impact



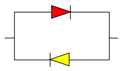
Reverse bias protection

Specific Protection!

- Reverse bias of the cell can occur for a variety of reasons including shadow or operation of a string close to 0V by the power system



Solar cell

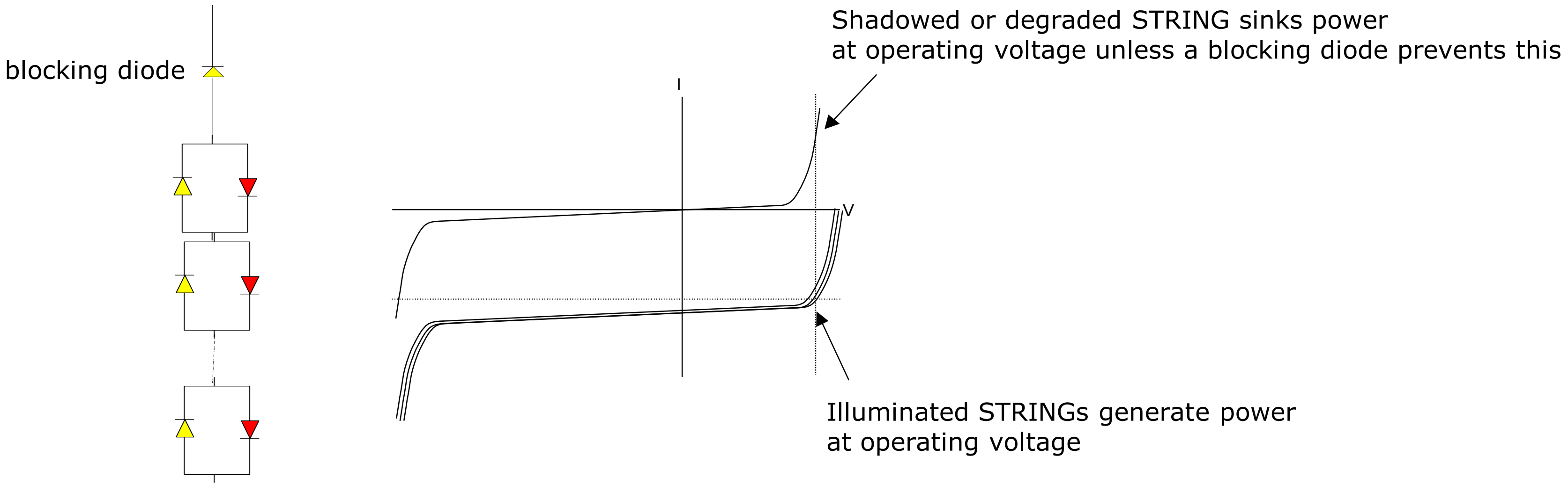


diode

Blocking diodes

Specific Protection!

- Blocking diodes are typically added to each string to ensure that no string can act as a ‘sink’ for the power generated by other strings

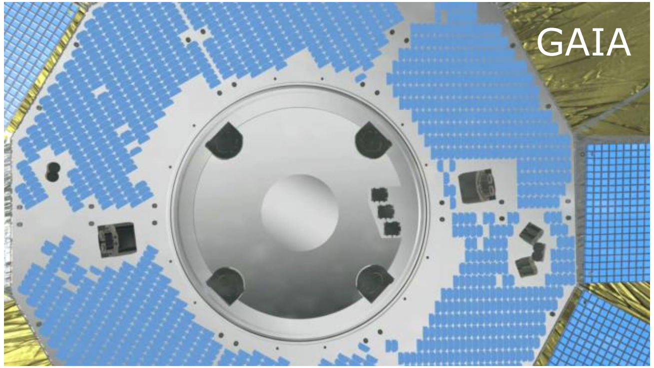


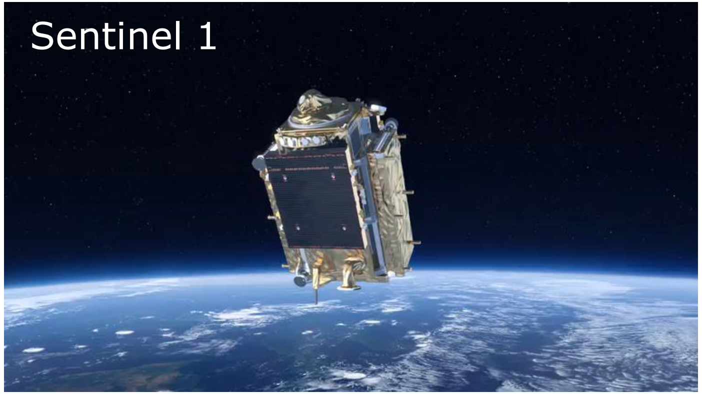
Solar Array, conceptual design

- The design of the solar array is a complex trade-off with many inputs, at the level of the satellite system

- Fixed, body mounted arrays are used for some low power missions in earth orbit
- Deployable arrays with fixed orientation may be an option for favourable orbital configurations
- Deployable arrays which can be oriented (usually rotated) are used otherwise

Examples of solar array configurations





Solar array temperature ranges

- Deployed solar array radiates on both sides:

$$\frac{\alpha_c - \eta}{\varepsilon_r + \varepsilon_f} q_S = \sigma \bar{T}^4$$

- Body mounted solar array radiates from front side only

$$\frac{\alpha_c - \eta}{\varepsilon_f} q_S = \sigma \bar{T}^4$$

- Earth orbit simple calculation (η = 20%):

no albedo

Deployed	
Operational	47 °C
Non operational	68 °C

Stowed / body mounted	
Operational	108 °C
Non operational	133 °C

Eclipse exit (approx.) :
-90°C LEO orbit
-170°C GEO orbit

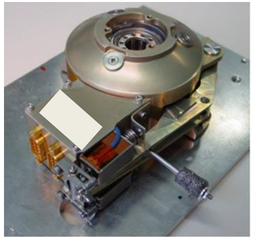
α absorption
ε_f emission from front
ε_r emission from rear
η conversion efficiency
q solar flux
σ Stephan constant
T temperature

Detailed requirements

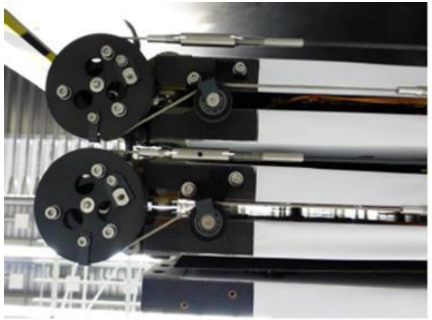
- Power output under different mission conditions
- Electrical interface requirements
    - ranges of current, voltage
- Mechanical requirements
    - Mass, volume
    - Center of gravity
    - Natural frequency when deployed
    - Tolerance of the launch environment (vibration, acoustic noise, shock)
- Thermal requirements
- Reliability
- Cost

Apart from the solar cells …

- Electrical
  - ‘Bleed resistors’ to limit damage in case of insulation loss
  - Thermal sensors
  - Electrical harness / wiring / connectors



- Mechanical
  - Hinges to allow folding / deployment
  - Hold down / release mechanisms
  - Springs and / or motors to provide energy for deployment
  - Inserts, Brackets and clamps

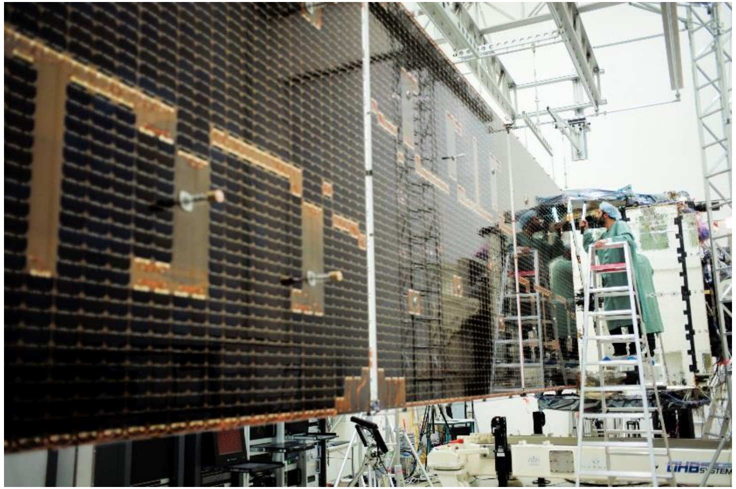


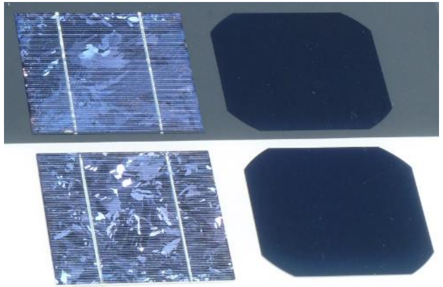
Images courtesy of Airbus

Space vs terrestrial solar cells

Space solar cells	vs	Terrestrial solar cells
Sun spectrum is unfiltered (AM0)		Sun spectrum is filtered by the earth's atmosphere
Electron and proton radiation cause degradation (also micrometeorites, ozone etc.)		No particle radiation
UV is significant		UV is attenuated by atmosphere and optics
Only alternative is nuclear power		Many alternatives, both renewable and non-renewable
Mass is important since launch costs are high		Mass is not an issue
High efficiency cells are most cost effective at system level		Many material technologies are potentially cost effective but >90% are silicon

Space vs terrestrial solar cells





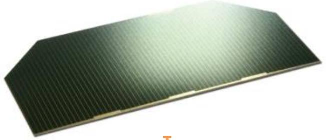
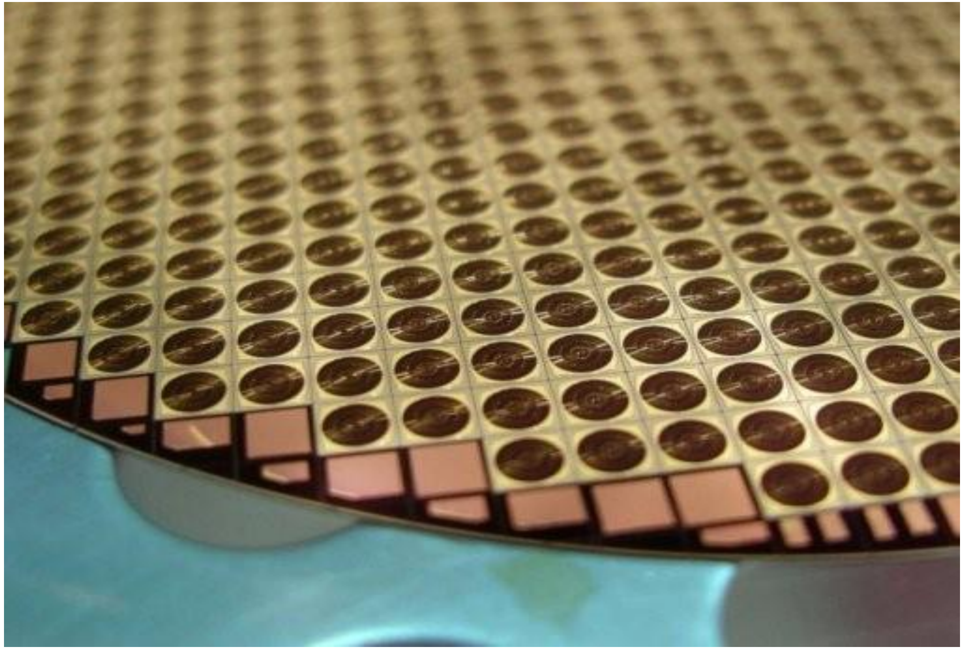
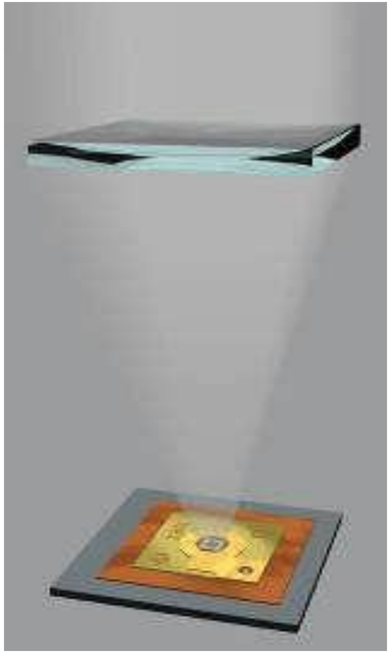


Image courtesy of Azur Space

Concentrator solar cells

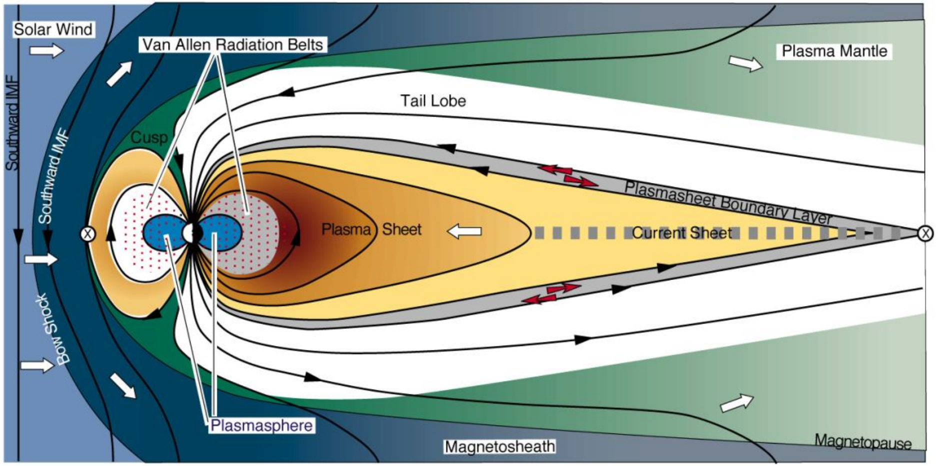




- Image courtesy of Azur Space

Spacecraft Charging

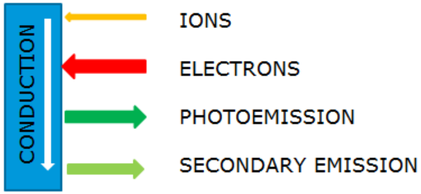
- Space is not empty.
  - It is filled with 'plasma' (electrons and ions) of low density but high temperature (~1x10^8K or 10keV in GEO)
  - Even higher energy particles (~MeV) are in the radiation belts
  - The environment varies with location and 'space weather'



- These charged particles stick to the spacecraft, charging it electrically.
- Sparks (electrostatic discharges) from one charged surface to another interfere with and damage electrical circuits and components.
- Many spacecraft failures and anomalies have resulted.

Spacecraft Charging

- Both the environment and materials play a role in spacecraft charging
- The following space environment features have a crucial influence on spacecraft charging:
  - Plasma – ions and electrons, 0eV to ~50keV
  - Energetic particle radiation, ~100keV to ~5MeV
  - Sunlight – ejects negative charge by photoemission
- Materials differ in terms of conductivity and yield of secondary emission and photo-emission.



Surface potential arises from the total current from electrons, ions, photoemission, secondary emission, conducted current and more…

Important: Different surfaces reach different potentials.

Spacecraft Charging, Effects

- Most critical engineering concerns
  - Surface charging due to charge accumulation on **external** surfaces. High levels of differential potential may lead to Electrostatic Discharge (ESD).
  - **Internal** charging due to more energetic penetrating electrons. ESDs may be generated within the spacecraft Faraday cage and in close proximity to vulnerable components.
  - ESDs on the **solar array** can cause secondary arcing

- Additional concerns include
  - Current leakage and power loss effects on solar arrays
  - Environment modification. Can be a critical problem for scientific plasma measurements .
  - Electric propulsion interactions with the environment
  - Electrostatic tethers current collection and voltage generation

Spacecraft Charging, Control

Most important:

1. Keep dielectric materials at not more than -1kV w.r.t. conductors
2. Keep dielectric materials at not more than +100V w.r.t. conductors
3. Maintain internal electric field not more than 10MV/m
4. An ESD on the solar array shall not lead to a sustained arc

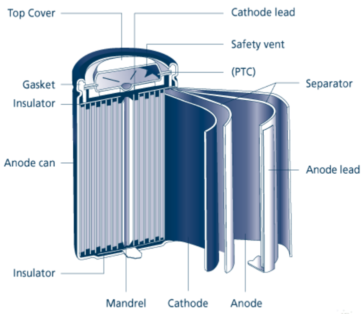
Additionally, take care of

- Grounding of conductors
- Selection of leaky insulators
- Selection of high photo/secondary emission yield materials
- Testing of solar arrays

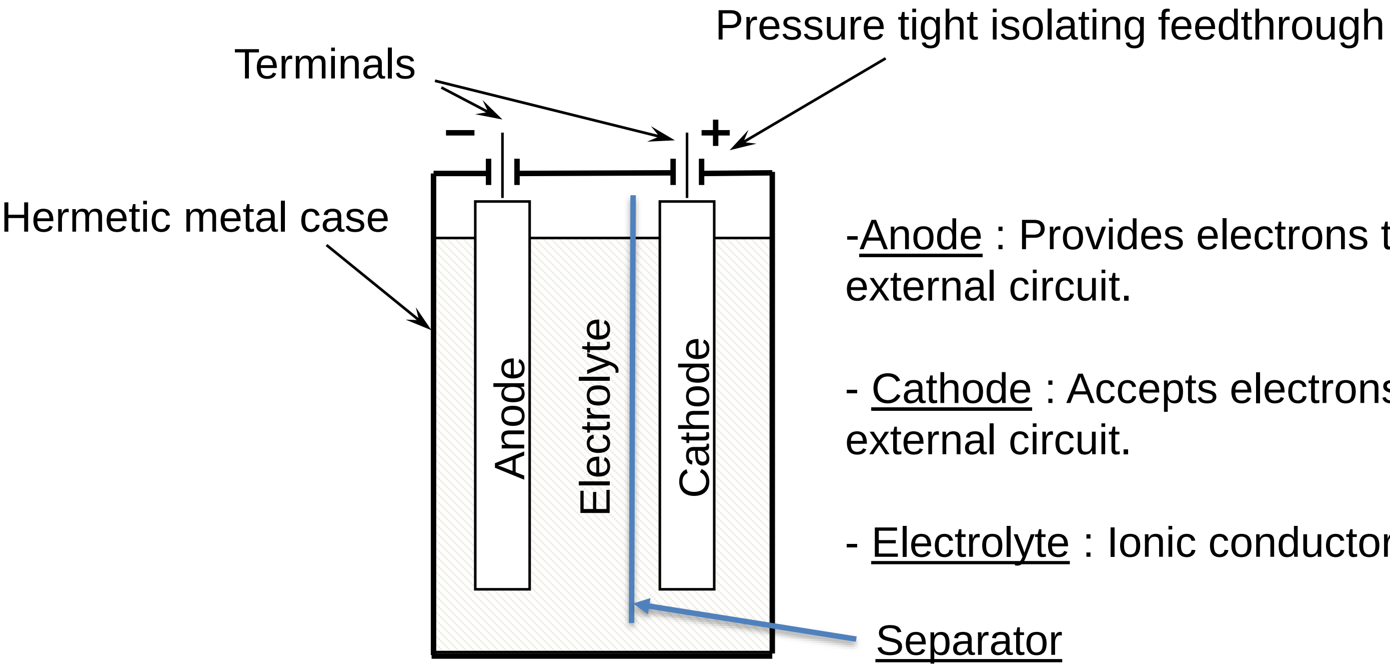
Spacecraft Charging, dependence on orbit

- Different requirements apply in different orbits because of their different environments:
  - Low Earth Orbit (LEO): consider
    - Surface charging for scientific s/c only
    - Solar array ESD effects
    - High-voltage interactions
  - Polar Earth Orbit: consider
    - Surface charging with some relaxation
    - Solar array ESD effects
    - High-voltage interactions
  - Medium Earth Orbit (MEO) or GEO: consider
    - Surface charging
    - Solar array ESD effects
    - Internal parts and materials

Energy storage



Battery cell



-<u>Anode</u> : Provides electrons to the external circuit.

- <u>Cathode</u> : Accepts electrons from the external circuit.

- <u>Electrolyte</u> : Ionic conductor

Two classes:

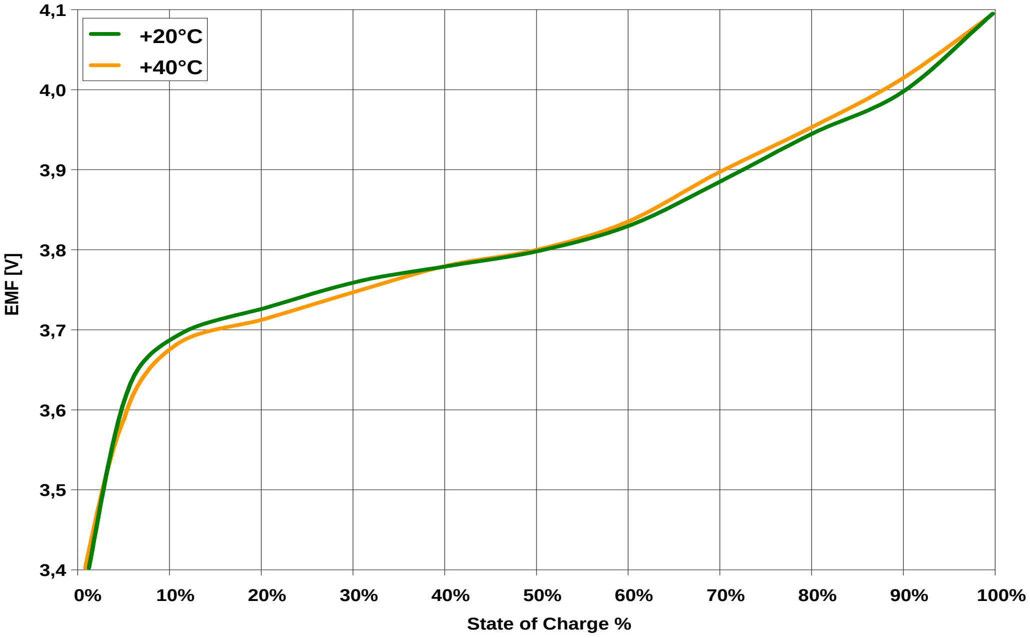
- Primary (non-rechargeable)
- Secondary (rechargeable)

Battery basics

- A **battery** is an assembly of electrically interconnected cells in series and / or parallel
- The **gravimetric energy density, "Specific Energy"** of a battery is the electrical energy per unit mass [Wh/kg] available when fully discharged from a fully charged state at a given rate and temperature
- The **specific volumetric energy density, "Specific Volume"** is the energy per unit volume [Wh/l] available when fully discharged from a fully charged state at a given rate and temperature
- The **Wh efficiency** – more commonly called roundtrip efficiency is the ratio of energy provided between full discharge and following full charge at a given rate and temperature
- The **DoD (Depth of Discharge)** is the amount of energy taken out of the battery per cycle. Looking in the opposite direction, the **SoC (State of Charge**) is 100 - DoD.
- The **capacity** [Ah] is the measure of Ah released from the battery at discharge and consumed at charge. The **nameplate capacity C** is the Ah released at a full discharge under standard conditions (discharge rate and temperature)

Batteries performance

The performance of the battery depends on the temperature and SoC



The voltage depends on the SoC

The temperature range is limited.

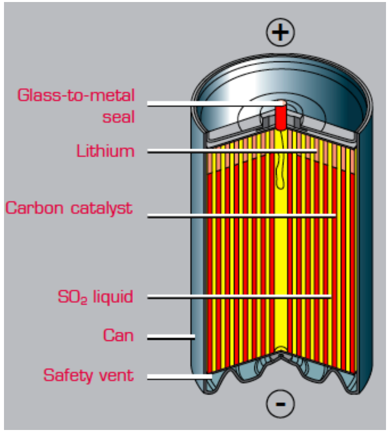
The internal resistance increases at low temperature

Battery space requirements

- High specific energy (and volumetric density)
  - Primary cells: > 500 Wh/Kg (Li-$SOCl_2$, Thionyl chloride)
  - Secondary cells: up to 180 Wh/Kg for Li-Ion if long life (shelf + cycle) life is desired
- Long lifetime
  - Shelf life – with no operation
  - Cycle life – with cycles applied
- For some applications, high power capability
  - Combination with supercapacitors
- Safety/reliability
  - No break-up/explosion possible
  - Safety mechanisms
- Predictable behavior
  - "small" spread in characteristics between cells that make up a battery

Primary Cells

- In space, primary cells are based on Li anodes
  - Much higher energy densities > 500Wh/kg can be achieved compared with alkaline cells.
- Flat discharge characteristic
- Low self discharge
- Very long shelf life
- Very long operating life (15 to 20 years for Li-SOCl2)
- Wide operating temperature range



SAFT Li-SO2 cell design

The Li – chemistries for space include:

- $Li\text{-}SO_2$ Sulphur Dioxide liquid cathode
- $Li\text{-}SOCl_2$ Thionyl Chloride liquid cathode
- $LiSO_2CL_2$ Sulphuryl Chloride liquid cathode
- $Li\text{-}MnO_2$ Manganese Dioxide solid cathode
- $Li\text{-}CF_x$ Carbon monofluoride solid cathode

Primary Cells – Lithium based

Chemistry	max rate	remarks	Temp. range	Wh/kg
Li-SO_2 , Sulphur dioxide	C 2C	Power High Power	-60 +70	>250
Li-$SOCl_2$, Thionyl chloride	C/10 C/2	Energy Power	-60 +85	500-700
Li-SO_2Cl_2 , Sulphuryl chloride	C/10		-80 +70	300-500
Li-MnO_2 Manganese dioxide	0.8C 1C	Power High Power	-40 +70	>280
Li-CF_x Carbon monofluoride	C/10		-40 +50	360-600

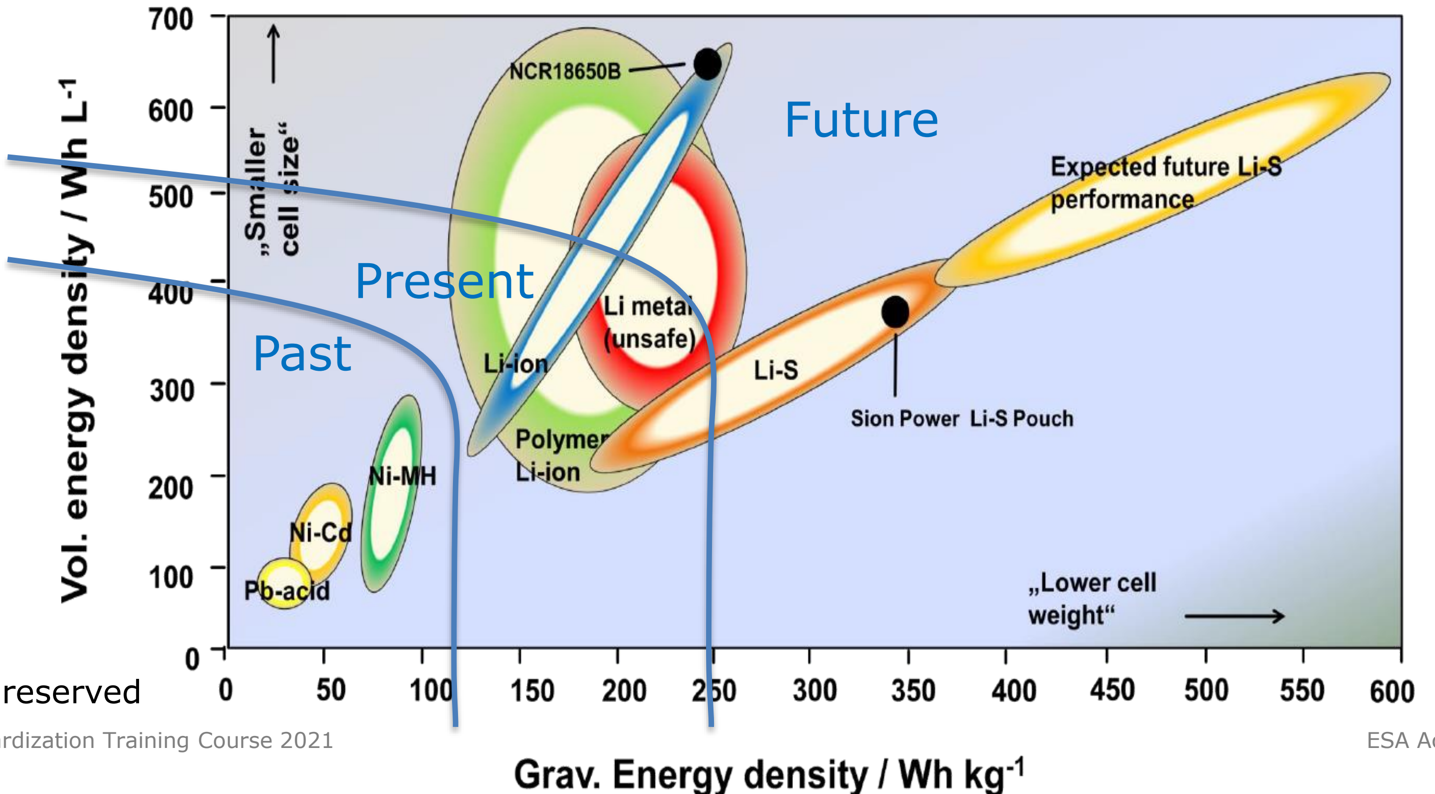
Some selected Li-primaries

- The highest specific energies can only be achieved at low rate
- **Most of the Lithium primaries need de-passivation after storage**

Primary Cells - Mission Design and Constraints

- Compared with rechargeable (secondary) cells, primary cells do have significantly higher shelf life up to 10 years, which makes them excellent candidates for long distance missions to planets like Saturn and Jupiter.
- However, the actual need of electrical energy is limited to a relatively short time span – usually hours to several days.
- The battery sizing and choice of chemistry therefore has to consider :
  - Required energy [Wh]
  - Required peak power [W], considering the rate capability at desired temperature
  - Capacity loss during cruise
  - Temperature requirements

Secondary Cells – Overview of space technology



Li – Ion cells ABSL

Based on fully commercial cells (COTS)

- 18650HCM, “low” energy density but high cycle life - space qualified
- 18650NL, high energy density at beginning of life - under ABSL internal qualification
- 18650HR, high power but limited energy density and cycle life - space qualified

At least for strings < 10 cells no cell balancing needed even for long missions

Cell arrangement before gluing to top plate.

Note kapton sheet between cell blocks. ECSS requires double isolation!

Li – Ion cells SAFT

Not in scale !

SAFT large cells (NCA - graphite):

VES180 S, VES140 SA, VES100 SA
(Number refers to nominal Wh)

Small VES16 4.5 Ah NCA graphite space qualified

MPS 176065 5.8 Ah $LiCoO_2$ - graphite commercial cell, qualified for space

Li – Ion Cell charge and discharge management, safety

- An increase in specific energy usually reduces the cell safety.
    - The energy chemically available in a typical fully-charged lithium ion cell is enough to heat it adiabatically to over 1000 ° C, causing a thermal runaway when mistreated.
- In lithium-ion cells there is no inherent overcharge protection reaction.
    - Overcharge, even at very low rates will result eventually in lithium metal plating on the negative and the positive becoming unstable and producing very reactive oxygen
- Even in vacuum conditions, this reactive combination can result in a serious fire which cannot easily be extinguished since both oxidant and reductant are within the sealed cell
- Additionally, most of the Li-Ion cells have a copper anode dissolving when overdischarged, the copper ions reduce again on the anode to copper metal forming dendrites which internally shorten the cells, no more electrochemical reaction happens.
    - Note: **discharge < 3V / cell** and charge > 4.1V or 4.2V/ cell is not recommended when cells are assembled to strings for flight batteries

Cell safety, protection against overcharge

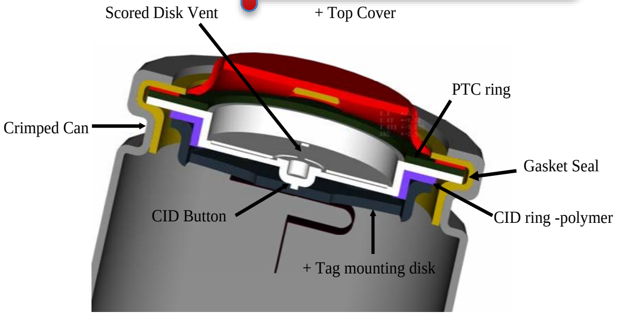
All commercial lithium ion cells have a device incorporated to open-circuit the cell in case of severe overcharge.

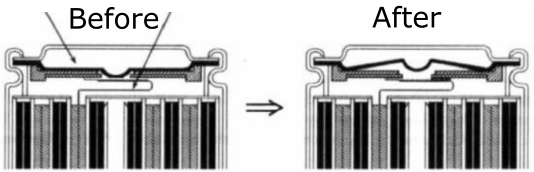
This usually consists of a pressure sensitive diaphragm which breaks the connection to one of the cell terminals. In addition additives in the electrolyte (eg. Lithium carbonate) are usually present in order to generate gas at lower states of overcharge to promote triggering of the device.

Overcharge protection devices may reduce the reliability and are sometimes omitted on space cell designs where necessary protection can be implemented outside the cells <see later>

To prevent the cell exploding, **all** cells are designed to leak before burst, usually by means of a pressure vent which opens well below the burst pressure of the cell case.

Specific Protection!





ABSL18650HC safety features

Cell safety, protection against external short circuits

Specific Protection!

To protect against **external** short-circuit most commercial cells have a **Polyswitch©** current-limiting device in series with one of the terminals.

This is a polymer-based device containing carbon particles that normally has a high conductivity.

In case of overcurrent the resulting heating of the Polyswitch causes the polymer to expand, the particles are separated, and the resistance increases, limiting the current to around 1C.

For space applications this device is of **marginal benefit** since overcurrent should be prevented by battery management electronics design, and the main negative effect of the device is to **increase cell resistance** throughout the mission, not just in emergencies.

Fortunately the device is **very reliable** and **extremely radiation tolerant** –

the polymer matrix is already made by radiation cross-linking.

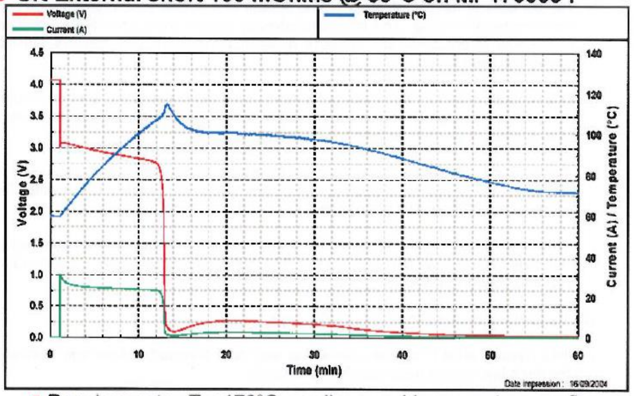
Cell safety, protection against over-temperature

Specific Protection!

Another standard safety feature is use of a “shut-down” separator.

This separator has 3 layers, the central one made of fibres of a polymer which melt and fuse together at a temperature below the critical one (80 vs 120) deg. C. Once fused, the reduced separator ionic conductivity limits the discharge current.

- UN External short 100 mOhms @ 55°C on MP176065 :



- Requirements : T < 170°C, no disassembly, no rupture, no fire.
- Results : T< 120°C, no disassembly, no rupture, no fire.

- These safety devices cannot protect against shorts due to foreign conducting particles which can penetrate the separator during life.
    - These are the cause of the well-published laptop fires.
    - It is a major concern for electric vehicles and manned space applications, it may exclude the use of highest specific energy chemistries
    - Preventing the presence of foreign conducting particles within the cell can is a key driver in the cell manufacturing process

Space Battery Layout, design drivers

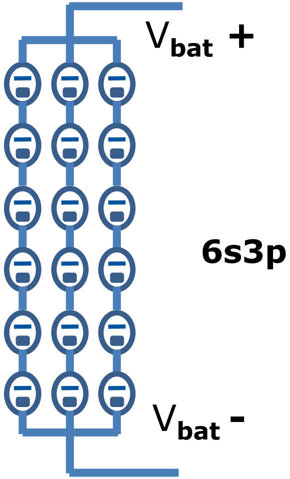
Specific redundancy!

- The number of cells in series gives the required battery voltage, the number of strings in parallel adds to the battery capacity. Extra strings needed for redundancy.
- Need of cell balancing, if required, adds to complexity, cost, weight and reliability issues.

Small cells ("s-p" configuration):

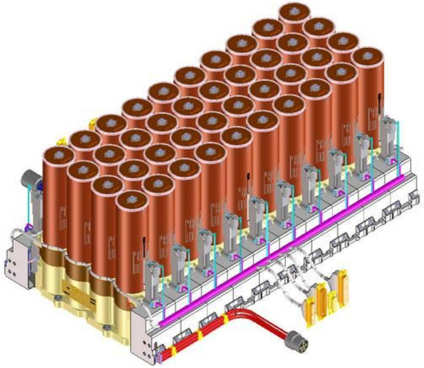
- Serial strings in parallel w/o interconnects
- Battery management on battery level
- Protections on cell level

PROBA1 6S6P 18650HC ABSL battery module

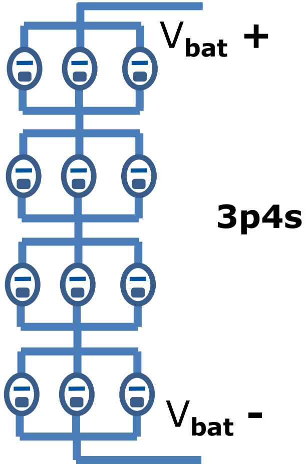


Large cells ("p-s" configuration):

- Cells in parallel interconnected
- Battery management by each parallel group
- By-pass protections for each parallel group



2x 2P10S VES180 SA SAFT battery module



Space Battery design drivers and sizing

Strict guidelines are given in the **ECSS-E-ST-20(C)**

- Need to consider all spacecraft modes and take into account battery and Solar Array ageing for modes which can occur at end of mission, this means that test data or accurate modelling results need to be available
- The sizing case is often one of the following:
  - Launch and early operational phase (LEOP) before the solar arrays are (fully) deployed and the attitude is stabilised
  - Full operational capability at end of mission
  - Safe mode requiring survival with no or insufficient SA power for specified time
- Battery sizing has to take into account margins
- Autonomous operations require safe under voltage protections to be able to recover from safe mode shutdown of the satellite, see next page.
  => a flat discharge curve is not so welcome than for many other operational aspects

Space Battery design drivers and sizing

- For GEO missions the sizing case is usually normal operation in the middle of the eclipse season (72 min discharge) at end of mission
  - At launch and safe mode the RF payload which is the main consumer is switched off and there is a large margin in battery energy
- For LEO, battery sizing can be quite difficult, depending on the power system topology, payload profile and orbital dynamics.
  - A simulation based on a full model of the electrical power system, including the battery, is generally necessary
- In case a MPPT (Maximum Power Point tracker) is used, to take full profit of the excess power delivered by the cold solar array at eclipse exit, the battery must be able to accept repeatately the corresponding surge current.
- Planetary missions often are driven by the cruising time to get there and the temperature the batteries are exposed to.
  - Often heaters are required to avoid battery oversizing.

Battery Electronic Management, fundamentals

- Use and lifetime limitations
  - Low number of charge/discharge cycles =>Higher DoD allowed => GEO !
  - High number of charge/discharge cycles =>Lower DoD allowed => LEO !

- Charge Management
  - For Li-Ion cells:
    - Constant current (C/3 – C/20) up to a defined, settable voltage limit
    - Then taper charge at the defined, settable voltage limit
    - Taper charge means regulating the voltage across the battery after this voltage limitation is reached.

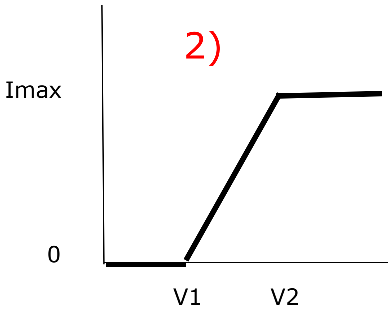
- Li-Ion Safe ranges to be guaranteed (even after a failure in the electronics)!
  - No Over-charge (typ 1-2% over the nominal end of charge voltage of 4.1V - 4.2V/cell)
  - No Over-discharge (typ 1-2% under the nominal end of discharge voltage of about 3V/cell)

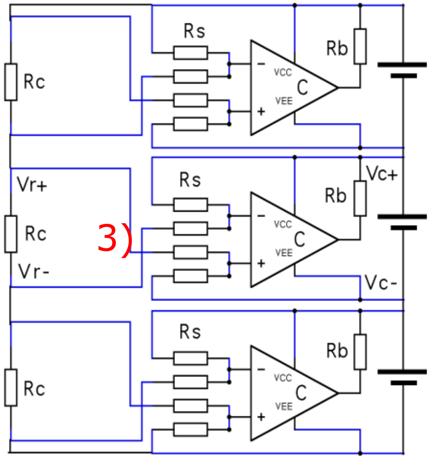
Electrical configuration – balancing issues

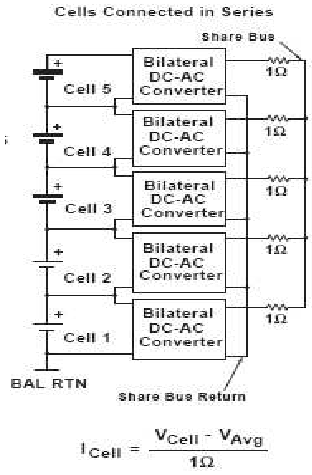
- Mainly applicable for large cells. Since Li-Ion cells have no overcharge capability and many Li-Ion cell technologies have a spread in self-discharge rates, a string of cells will get out of state of charge (SoC) balance after some time.
- The effective battery capacity will become limited on charge by the first cell to reach the end of charge voltage and on discharge by the first (often different) cell to reach the end of discharge voltage. Even worse, normally there is no single cell voltage monitoring on S/C.
- Consequently some means has to be implemented to compensate for this effect
- For that a cell (block) voltage monitoring becomes mandatory

Electrical configuration – some balancing methods

1. Fixed resistor is switched across highest voltage cell strings during charge
2. Individual cell string taper charge starts to discharge the cell when the cell(block) reaches V1 and limits to I(max) when V2 is reached
3. Cell strings with higher than average cell voltage are discharged with a current proportional to deviation from average, ESA patent – not flown
4. Aeroflex BEU: Each cell string is associated with a DC-AC converter, power is transferred from cells with higher than average voltage to those with lower V.





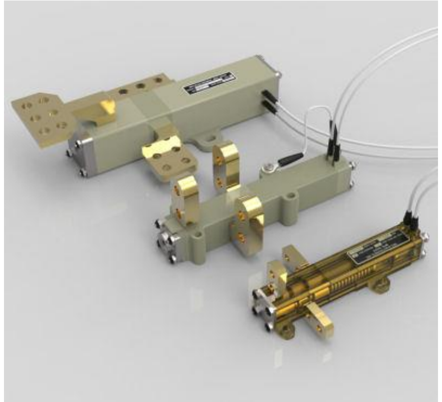


4)

Electrical configuration – bypass

- In p-s configuration no cell failure is acceptable, so if a cell malfunctions, this cell has to be bypassed to ensure the continuity of the battery configuration.

- Typically, switches are placed in series between battery cells and, when activated, bypass and isolate the failed cell from the battery assembly.



Space Battery Mechanical issues

- Reliability is driven mainly by mechanical and electrical robustness of internal connections and possible premature ageing
- For space, the main concerns are often the launch environment (vibration and shock), breaking internal connections. Seal integrity under vacuum is an issue as well
  - In most cell designs the jelly-roll is partly held in place by friction with the walls of the container. The jelly-roll swells irreversibly during cell formation so it can be introduced easily beforehand. It also reversibly swells during charge.
- Batteries are normally launched charged. There can be a problem if cells have to withstand severe vibration or shock when they are not fully charged like for atmospheric entry or landing of probes on planetary surfaces.

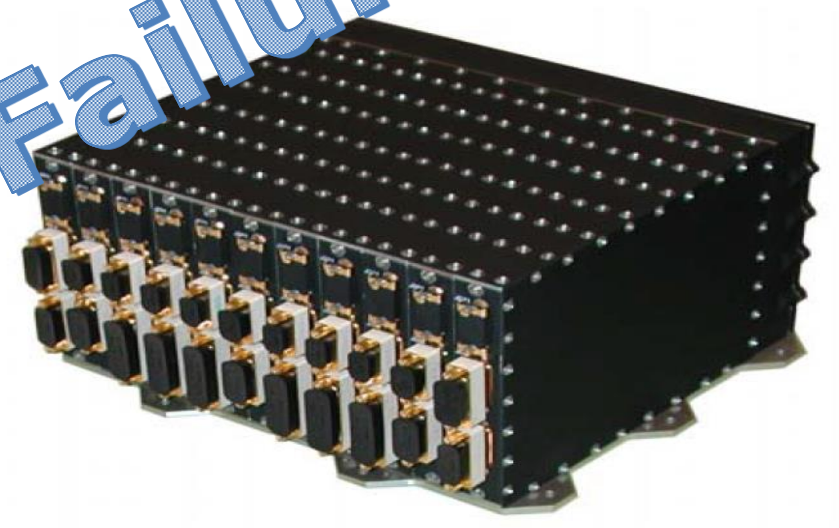
Space Battery thermal management

- Battery performance depends on electrochemical processes whether charging or discharging.
- These reactions are dependent on temperature and SoC. In space nominal battery performance is usually specified for working temperatures in the 0 to +30°C range, however the actual performance can deviate substantially if the battery is operated outside this temperature range because of mission needs.
- Higher temperatures improve electron and ion mobility, reducing the cell's internal impedance and increasing its capacity.
- However, even during storage at higher temperatures unwanted and often irreversible chemical reactions occur, which can cause permanent damage or complete failure of the battery. This in turn sets an upper temperature operating limit for the battery.
- For mission the design capacity and conversion efficiency have to be traded of with irreversible capacity degradation.
- A practical system may therefore need both heating (for operations) and cooling (for storage during cruise) to keep it not just within the battery manufacturer's specified working limits, but within a more limited range to achieve optimal performance. This approach is in fact a management at constant battery reserve energy, to guarantee the capability to face contingency

Power management and distribution

- Primary conversion makes use of a single box concept:
  Power Control Distribution Unit, or PCDU

  - Modular system that includes:
    - SA conversion
    - Battery charge
    - Battery discharge
    - Distribution
    - Telemetry and Telecommand
    - Heater drive
    - Deployment system drivers

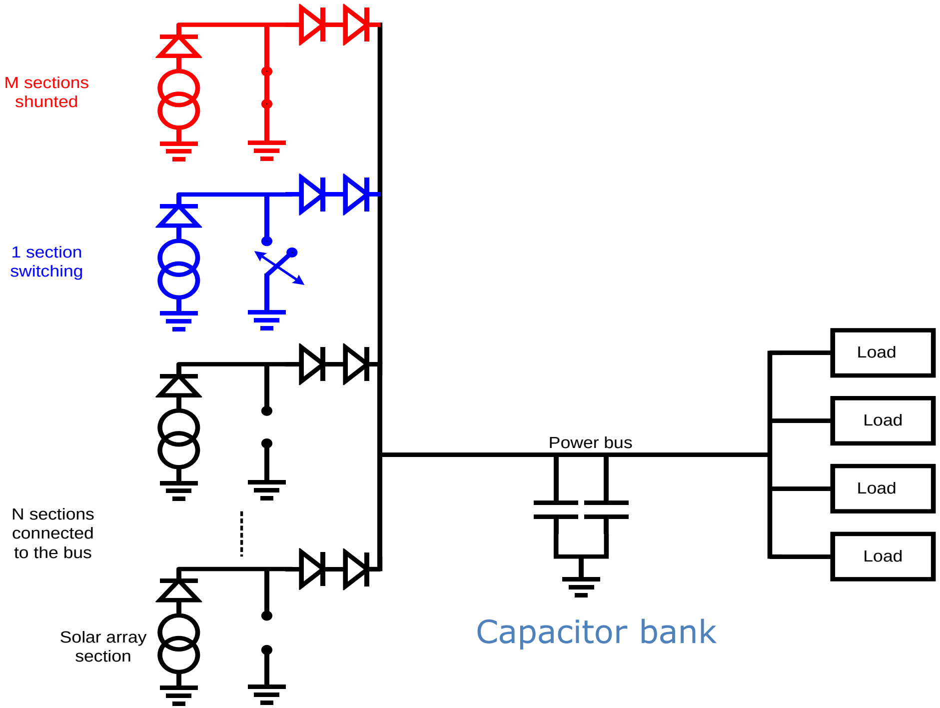


Single Point Failure Free

Power system architectures… but first…

- Solar array regulation is a key aspect
  - The two most common solutions:
    - Serial Shunt Switching Regulators (S^3R)
    - Pulse Width Modulation (PWM) Regulators with Maximum Power Point Tracking (MPPT) control

Sequential Switching Shunt Regulator (S3R)



It is a very simple concept

Only one section is switching

It operates in the current side of the SA

ESA invention 1977

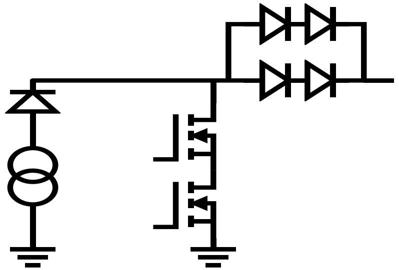
Equips the vast majority of satellites worldwide

It is also called Direct Energy Transfer device (DET)

Sequential Switching Shunt Regulator (S3R)

Specific redundancy!

Reliable sections: any single failure does not degrade the power available

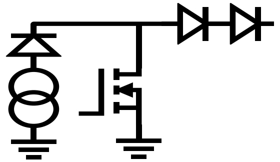


However, if a MOSFET fails in open, the section is permanently connected to the bus

It might require to have a dumping resistor to consume this power

Quad diode for full redundancy.

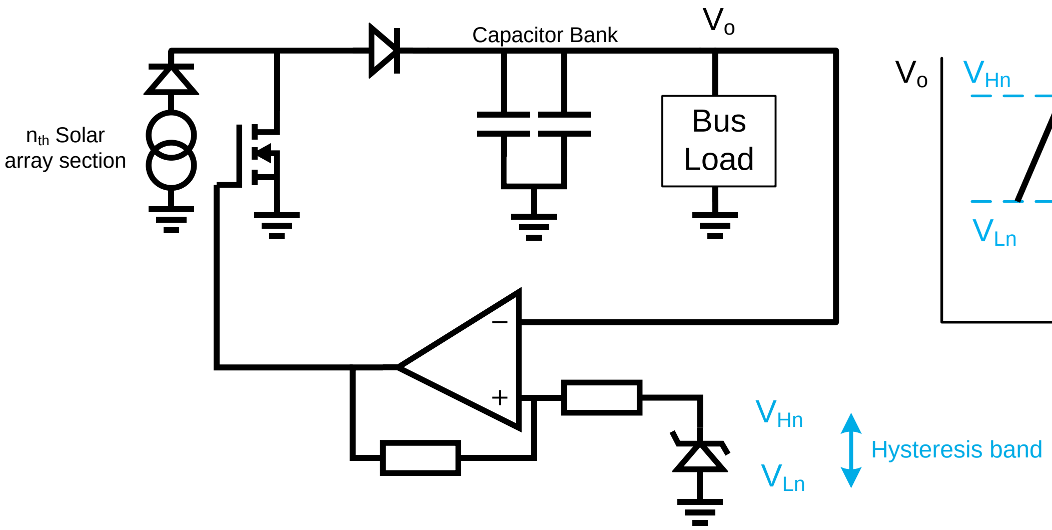
Non reliable sections: they can fail but the problem is solved at system level

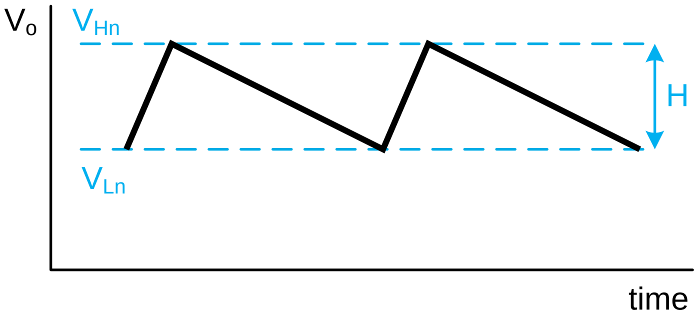


If the MOSFET fails in short or a diode in open, the power of this section is lost

An extra section is needed to cope with this failure

S3R control technique

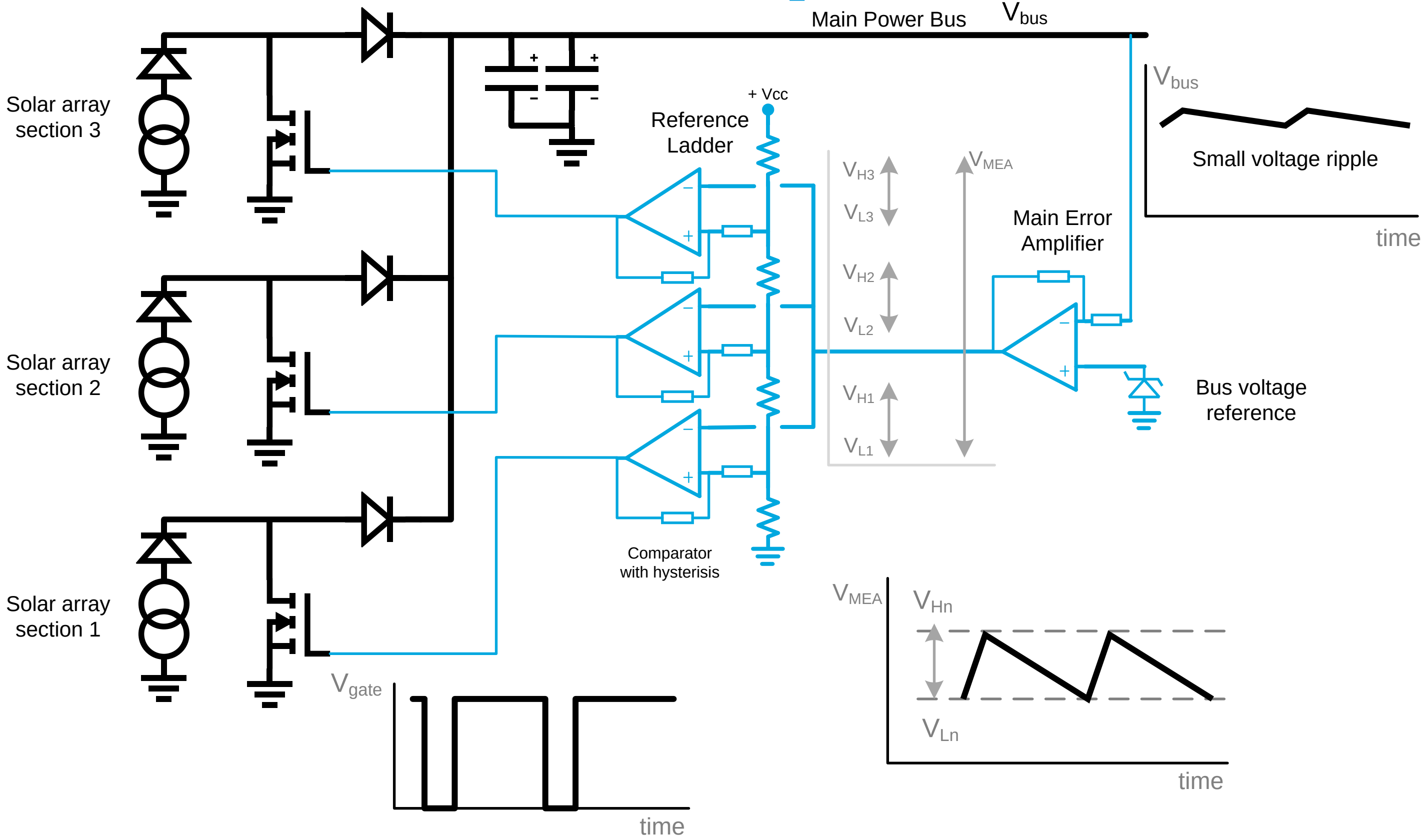




Variable Switching frequency. It depends on the load.

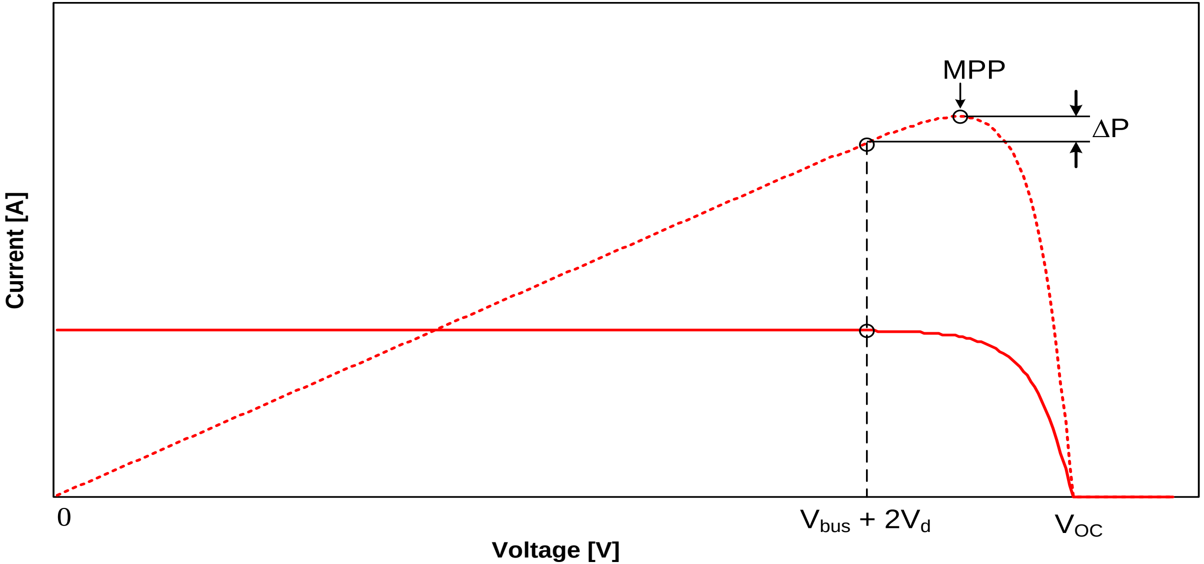
Very fast dynamic response.

S3R control technique

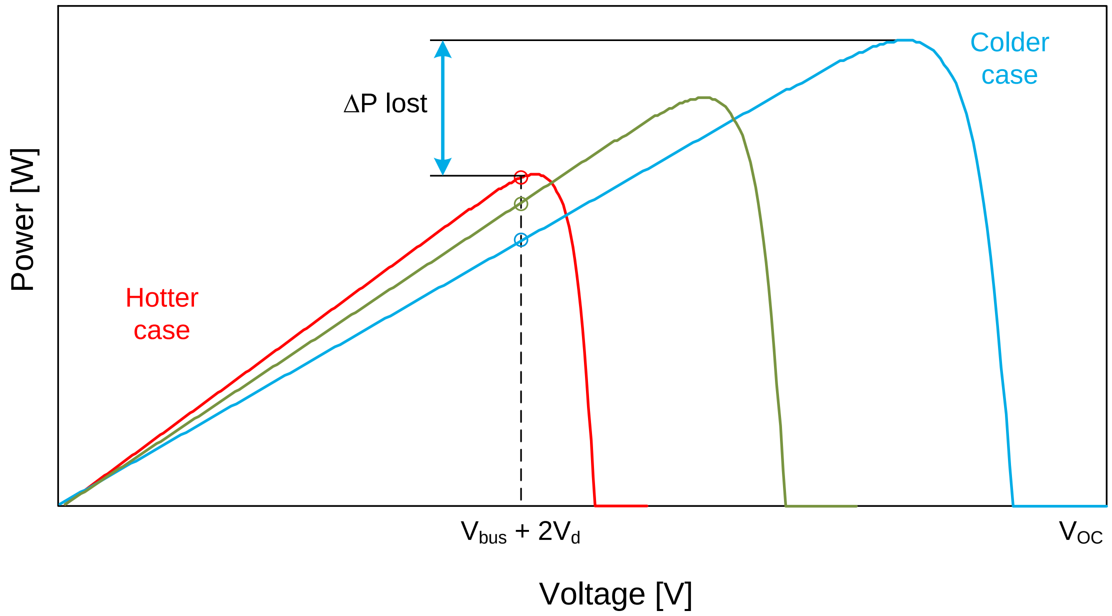


Sequential Switching Shunt Regulator (S3R)

- It is the most efficient topology.
- But, it operates the SA at a fixed voltage
- It is not able to make use of all the power available in the SA



S3R: fixed operating point



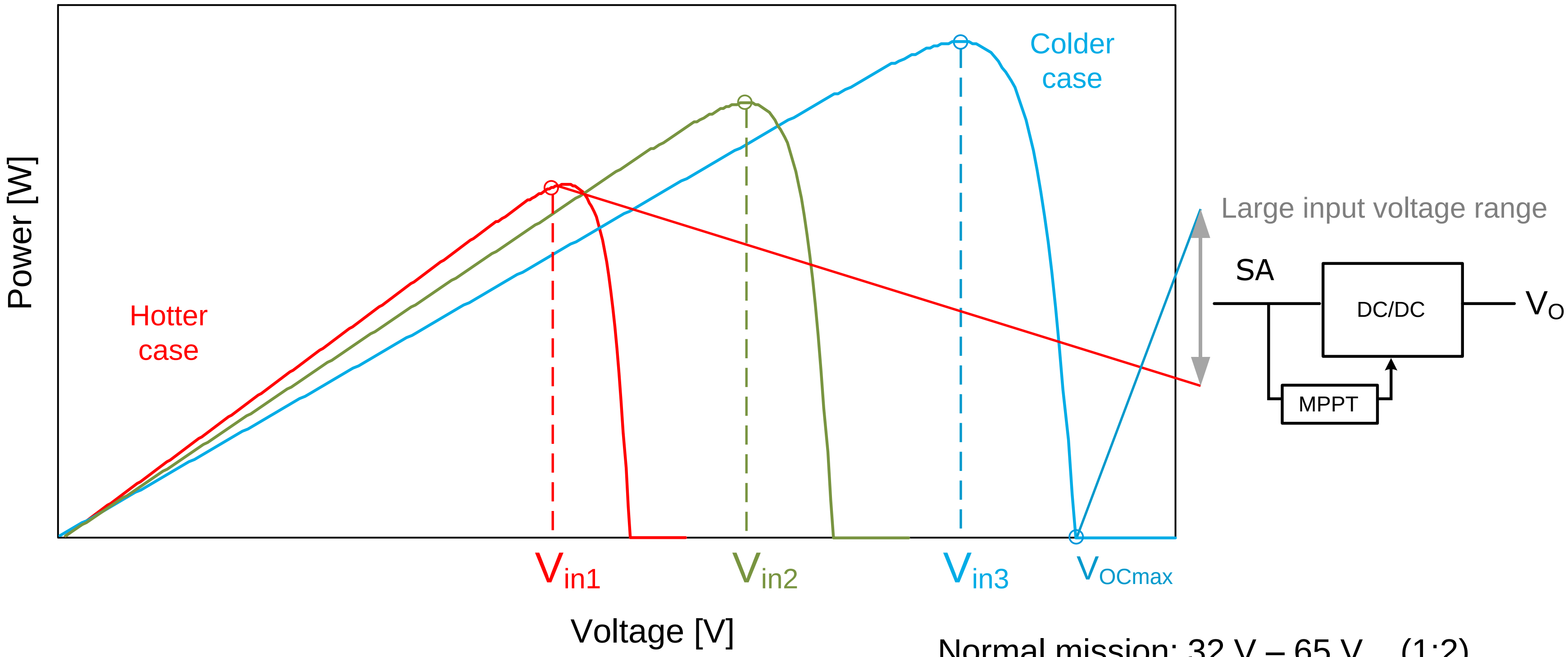
- In case the SA temperature (and illumination) conditions vary appreciably during the mission phases, then a power system based on MPPT might be preferred

MPPT

- Maximum Power Point Trackers are relatively new in Space
- A DC/DC converter is used for the power regulation
- If the maximum power available is needed, a circuit able to track the MPP is taking over the control of the converters.
- This circuit commands now the DC/DC converter so that the operation point is set at the SA MPP

The efficiency is slightly lower than S3R but the maximum power is available

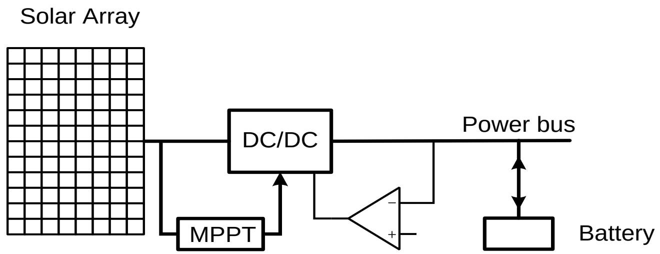
MPPT operating points



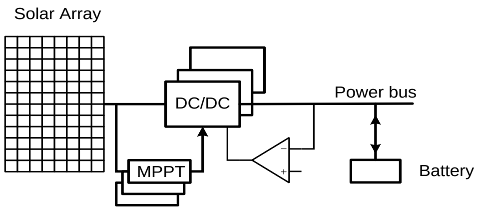
Normal mission: 32 V – 65 V (1:2)

Exploration: 32 V – 170 V (1:5)

MPPT

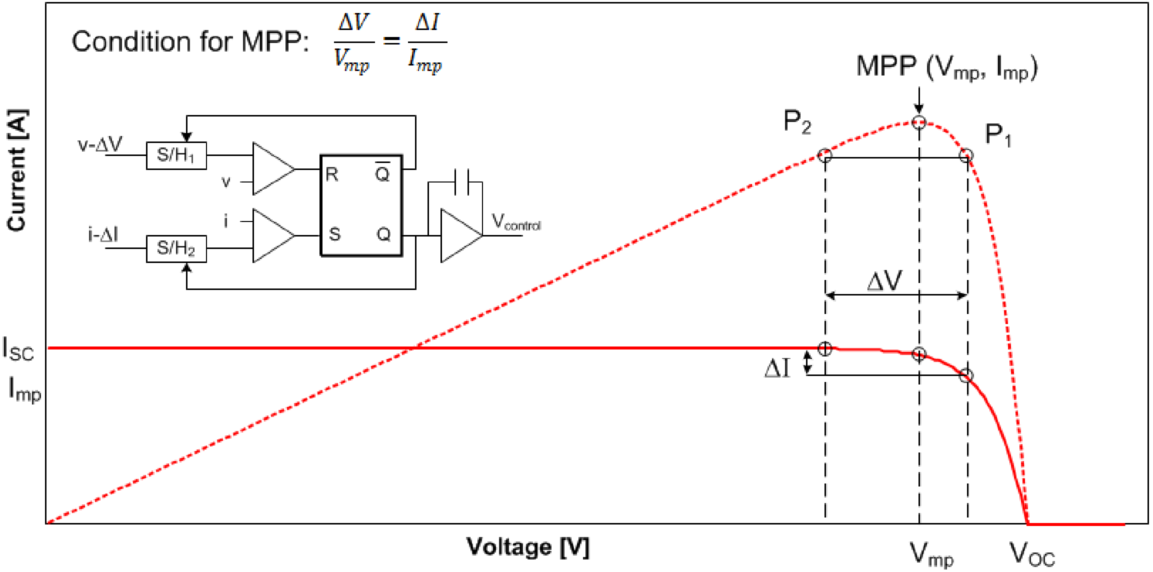


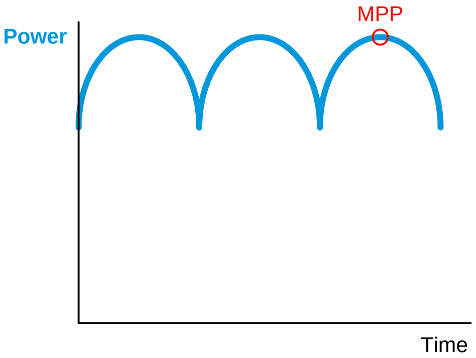
- The efficiencies of the DC/DC converter are around 95% - 96%
- The MPPT accuracy is around 1%
- The MPPT can be fully analog
- Redundancy is needed in all the circuits



MPPT practical implementation

- In 2003, Rosetta/ Mars Express used an MPPT for the first time in ESA
- Very simple analog circuit.
- Low frequency oscillation around the MPP. Typically < 100 Hz.





Basic comparison between S3R and MPPT

	S3R	MPPT
Efficiency	94% - 98.5%	90% - 95%
Loss wrt MPP	2% - 15%	1%
Complexity	Lower	Higher
Mass	Lower	Higher
SA impact	Depends..	Depends..

Low impact at PCDU level

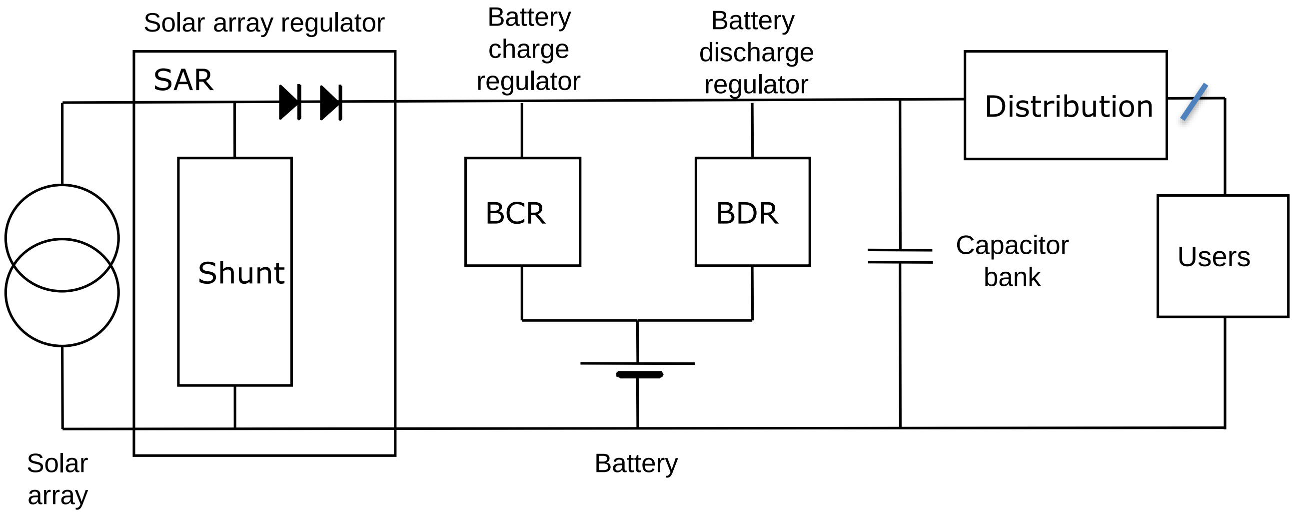
Power system architectures

- Regulated bus with S3R
- Regulated bus with MPPT
- Hybrid bus with S3R
- Unregulated bus
- Examples

Regulated bus

S3R version

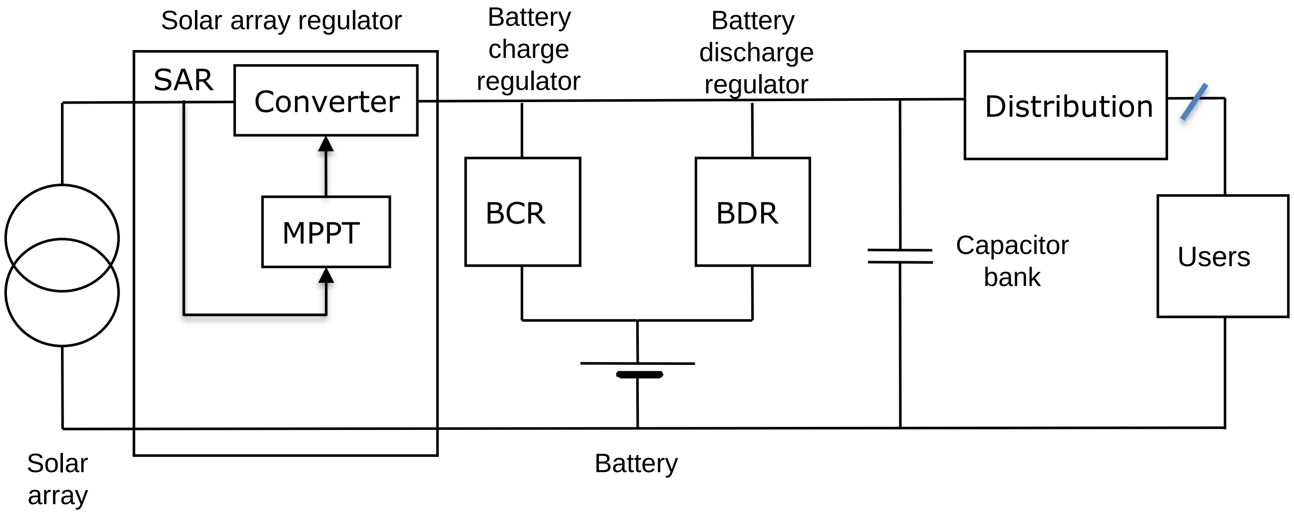
- Mostly used in telecom



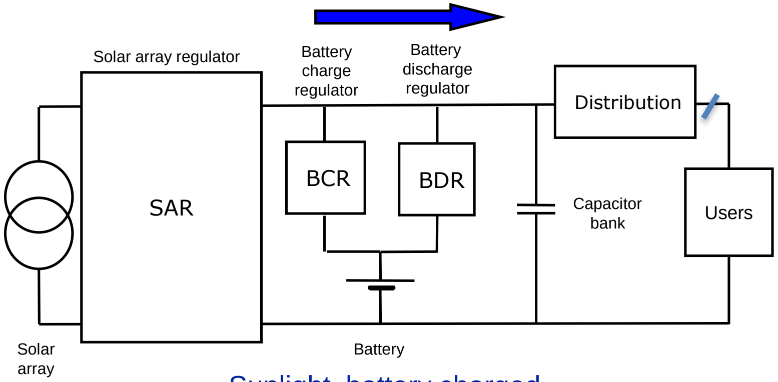
Regulated bus

MPPT version

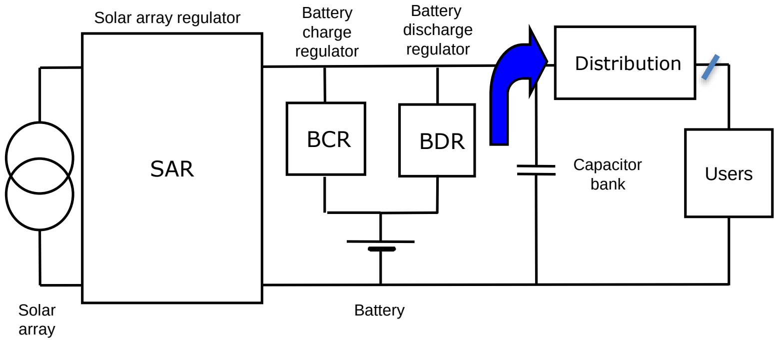
- Mostly used for science



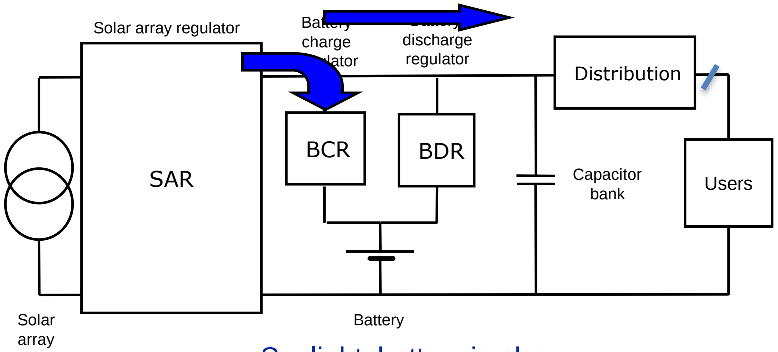
Regulated bus, how does it work?



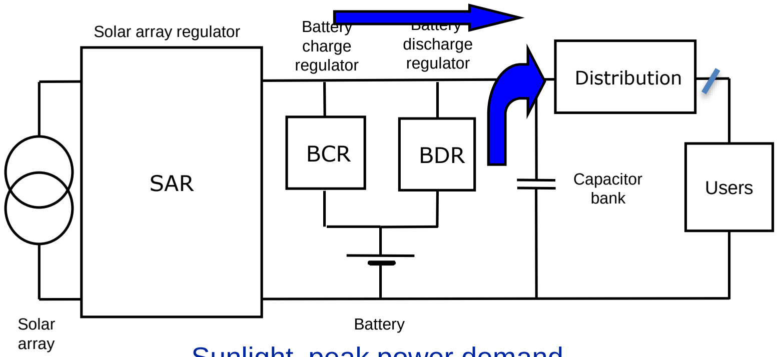
Sunlight, battery charged



Eclipse



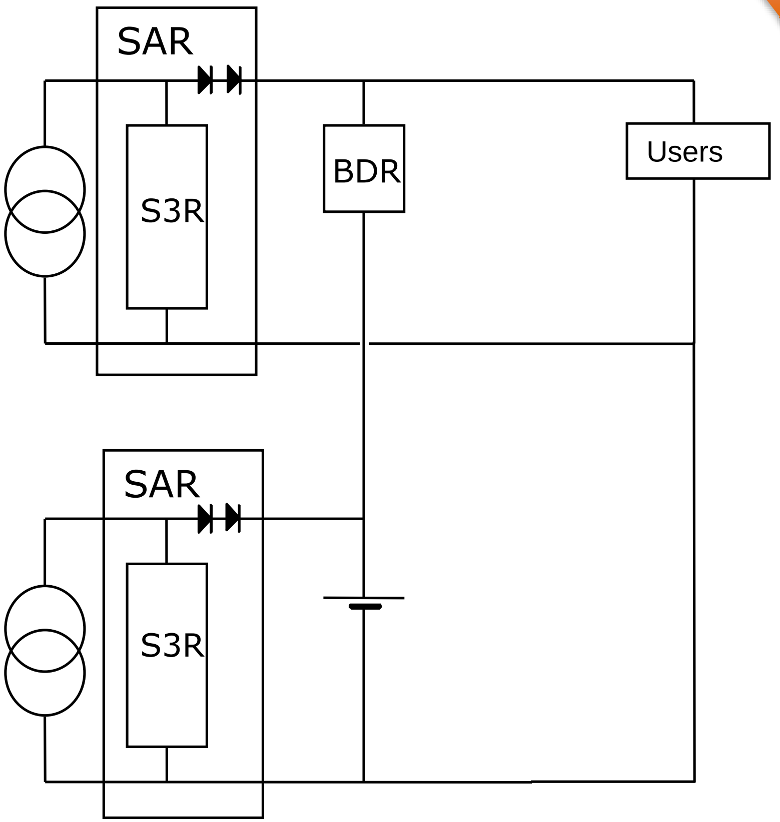
Sunlight, battery in charge



Sunlight, peak power demand

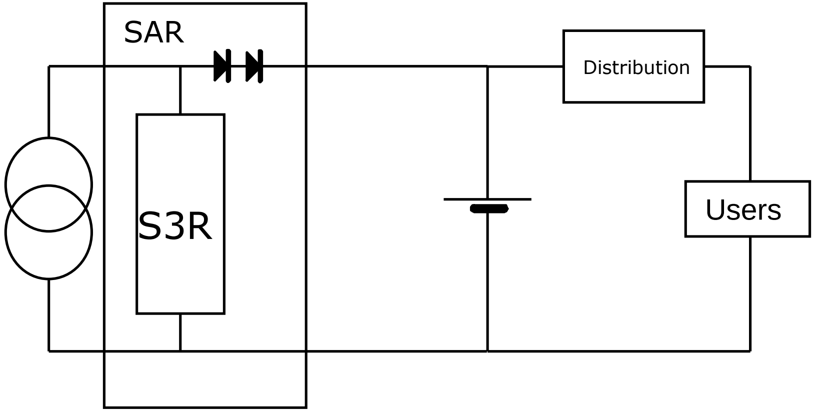
Hybrid bus

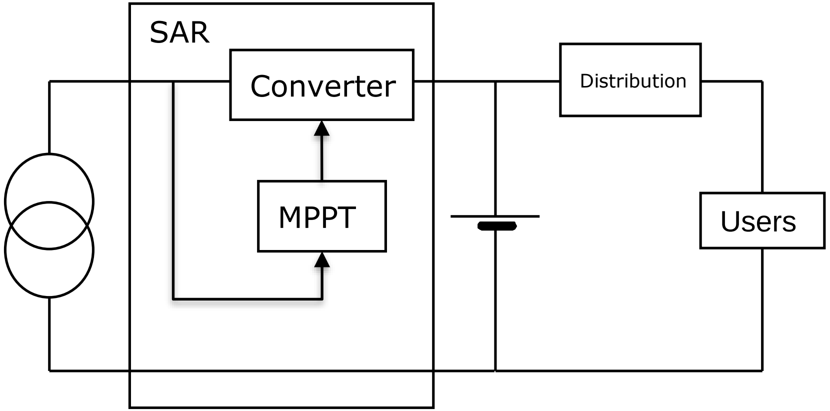
- Many other architectures are possible to improve the overall performance
  - LEO “Hybrid” topology
    - ☺ Full bus regulation is still ensured
    - ☺ Bulky BCR is replaced by an S3R
    - ☹ Peak load demand exceeding the main array capability is provided by the charge array at the expense of the BDR inefficiency



Battery bus (the "Unregulated bus")

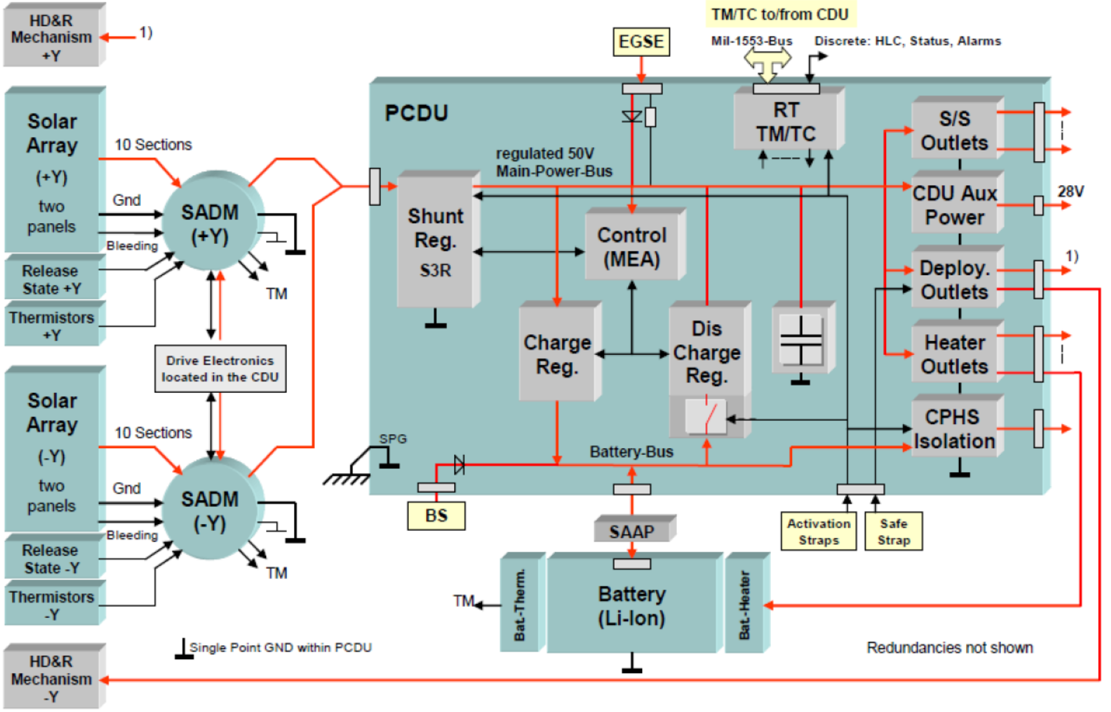
- Even the BDR alone can be too much
  - The battery bus in two versions
    - ☺ Bulky BCR and BDR are removed
    - 😐 Bus regulation is no longer ensured but battery voltage excursion is limited, especially with Li-ion in the S-P configuration
    - ☺ Peak load demand is coped with "naturally"
    - Both DET and MPPT configurations are feasible
      - the MPPT may even slightly improve the overall energy budget by taking profit of the high eclipse exit SA peak power





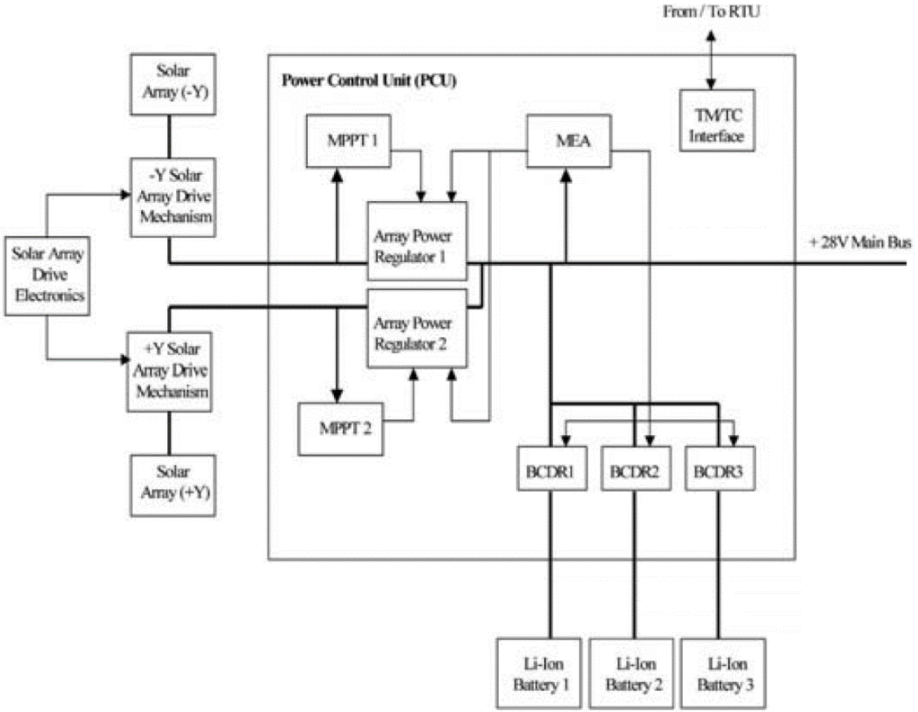
Examples of actual topologies

- Galileo FOC, fully regulated bus



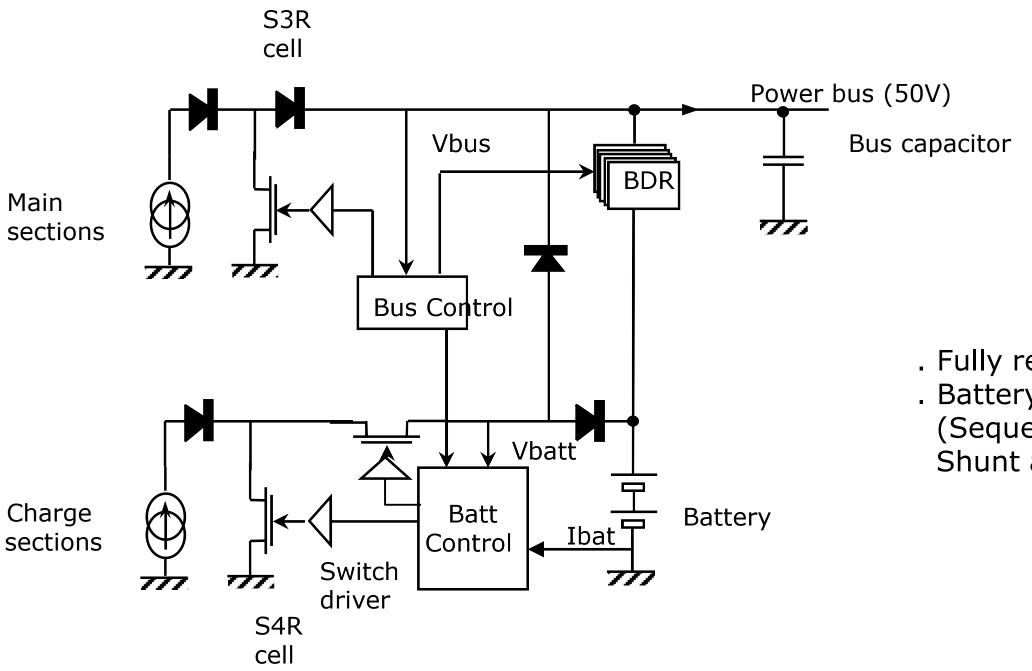
Examples of actual topologies

- Venus Express: the fully regulated bus with a MPPT



Examples of actual topologies

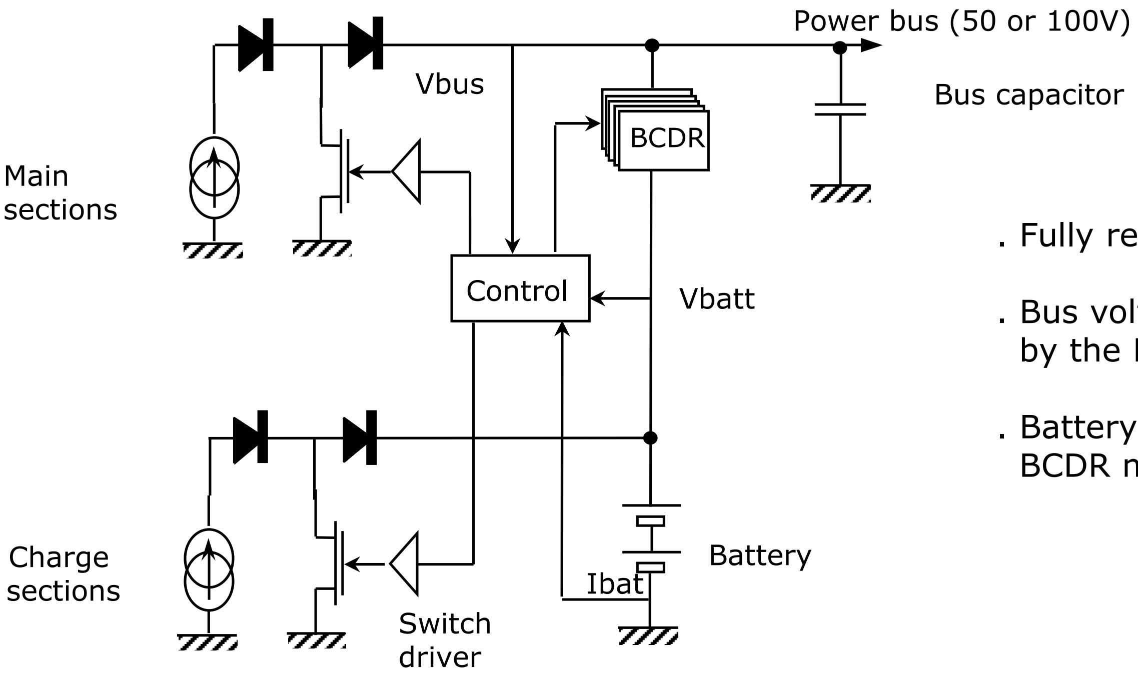
- Galileo IOV: the fully regulated bus almost by the book



. Fully regulated 50V
. Battery charge by S4R
(Sequential Switching
Shunt and Series Reg.)

Examples of actual topologies

- Eurostar 3000: the fully regulated bus with a trick



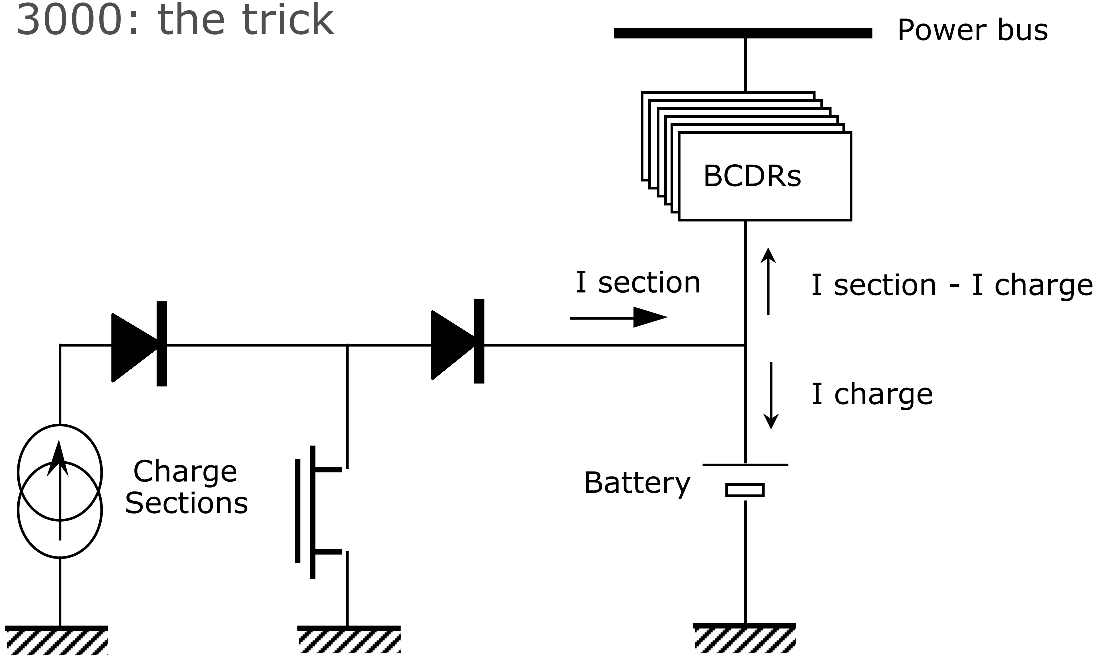
. Fully regulated bus (50 or 100V)

. Bus voltage regulation ensured by the Power Supply Regulator (PSR)

. Battery charge regulation ensured by BCDR modules (bypass control charging)

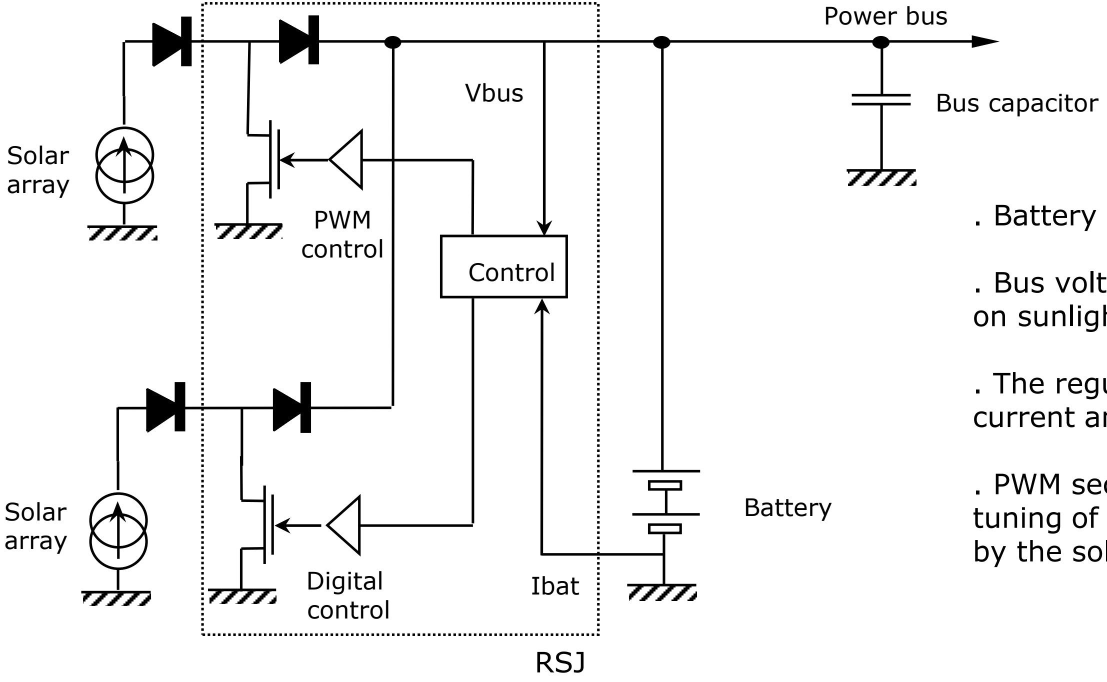
Examples of actual topologies

- Eurostar 3000: the trick



Examples of actual topologies

- Envisat: the battery bus almost by the book



. Battery follower (22V-37V)

. Bus voltage = battery voltage on sunlight and on eclipse

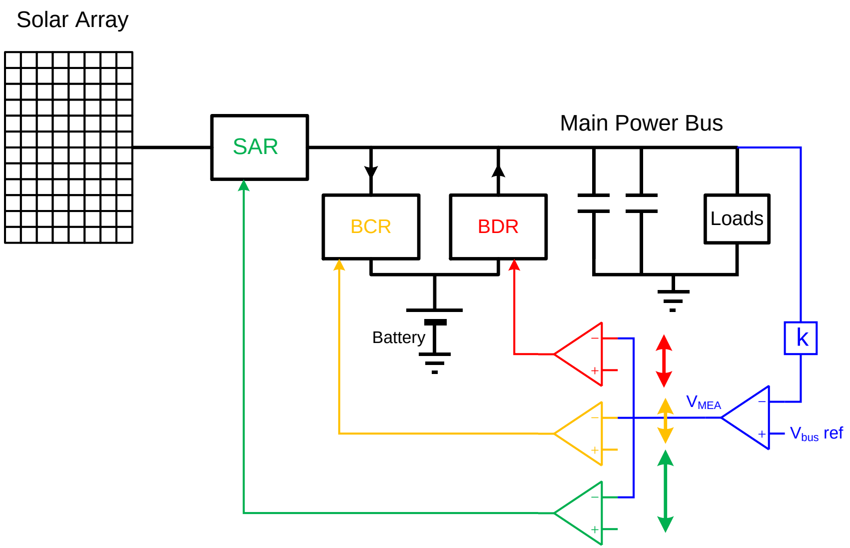
. The regulator only limits battery current and voltage

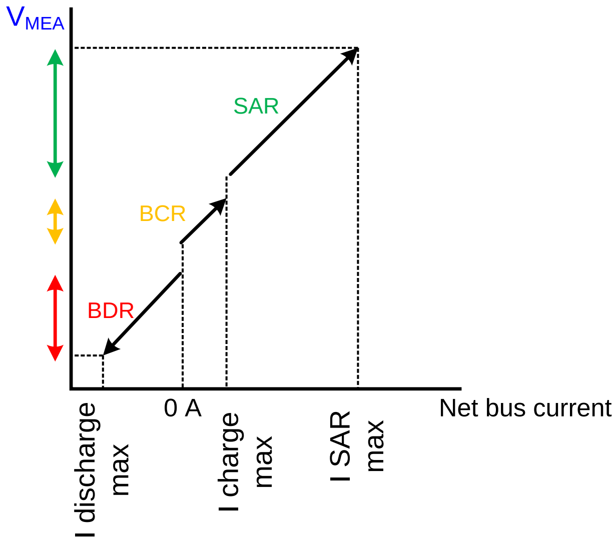
. PWM sections provide a fine tuning of the current delivered by the solar array.

Other key elements of the Power system

- Typical control scheme
- Battery Charge Regulators
- Battery Discharge Regulators
- Distribution & protection elements

Fully regulated bus – 3 domain control

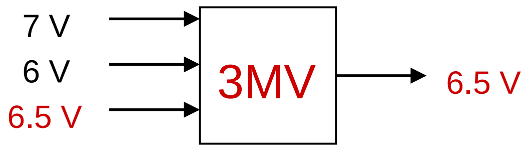




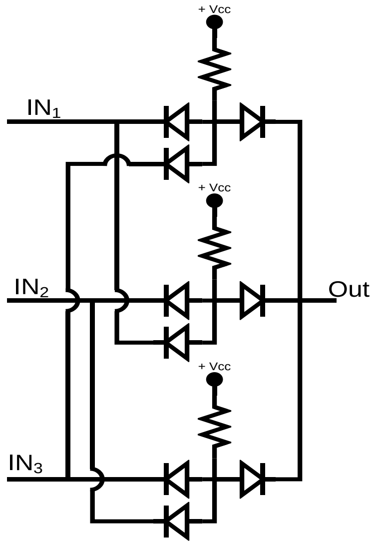
- Each regulator (SAR, BCR, BDR) is configured as a voltage commanded current source.
- The control voltage levels of the three systems (SAR, BCR and BDR) are located in a logical sequence
- There is a common Main Error Amplifier (MEA) driving the three domains

MEA, triple majority voting

- For the MEA, a Triple Majority Voter (3MV) system is implemented
- Three signals enter into a circuit that chooses the one which value is in the middle



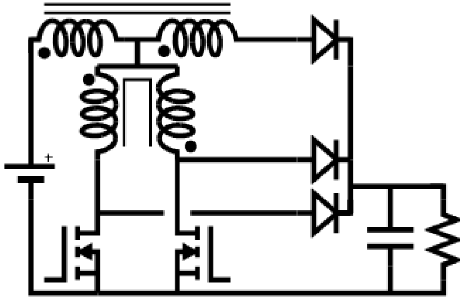
A simple implementation



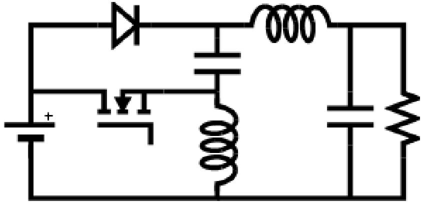
Battery Discharge Regulators

- The most common case is to use step-up (boost) regulators
- Superboost / Weinberg are typical topologies
- Variable input voltage range (e.g. 20 V - 33.6 V for 28V bus)
- The BDR has to implement:
  - Protections to avoid battery overdischarge
  - Maximum current discharge control
  - High output dynamics
- The BDR power level is the max S/C power (in eclipse)

Weinberg



Superboost



Battery Charge Regulators

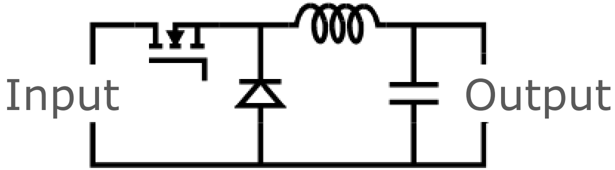
- The battery voltage could be:
  - Lower than the bus (most common)
  - Higher than the bus

- Buck / Superbuck are typical topologies

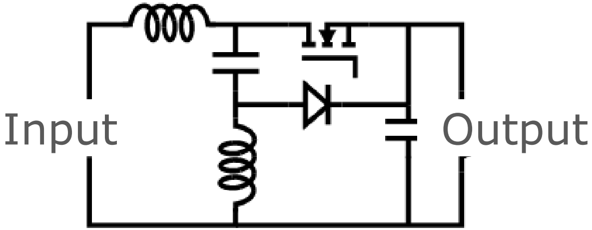
- The BCR has to implement:
  - Protections to avoid battery overcharge
  - Maximum current charge control
  - Configurable End of Charge (EoC) settings

- Typically, the BCR power level is lower than the max S/C power

Buck



Superbuck

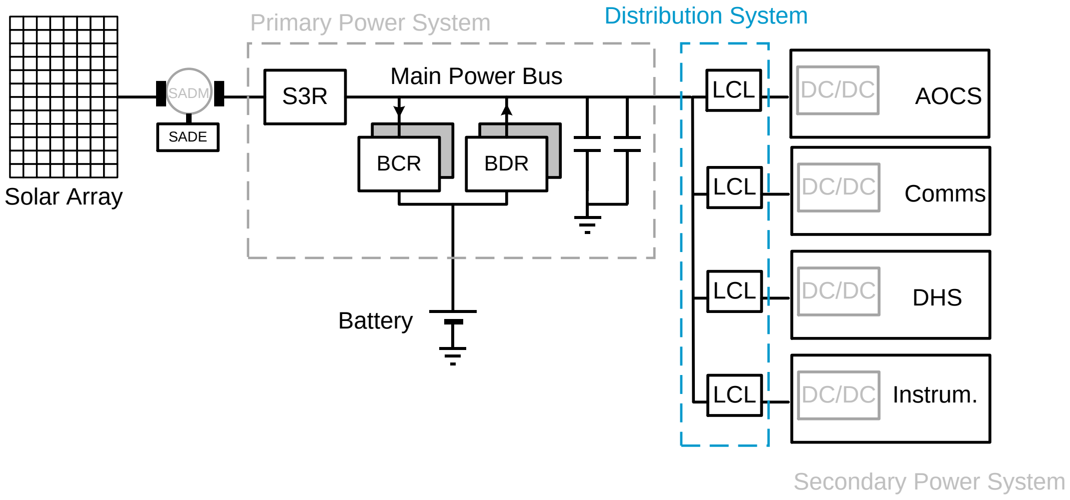


Distribution system

- The main bus shall be protected for ensuring spacecraft survival

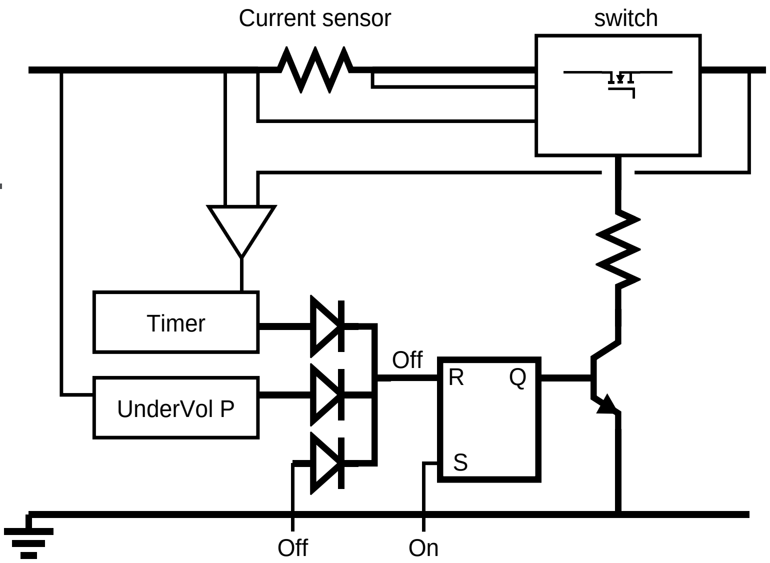
- Failures at load level shall therefore be isolated and disconnected from the bus

- The protection devices can be
  - Fuses
  - Latching Current Limiters (LCL)
  - Retriggerable LCL (RLCL)



Distribution system – the LCL -

- The LCL is a solid state switch provided with current limitation (from a fraction of an Amp to several Amps).
- It normally works in ohmic mode (e.g. it presents a small resistance in series with the load current).
- The voltage drop is kept typically within 1% of the nominal Main Bus voltage level.
- In case of an overload on the line, the current limitation feature enters quickly into action, and the line is opened if the overload duration exceeds the trip off time (some ms to tens of ms).
- The LCLs are usually protecting the non essential loads (otherwise the RLCL is used).



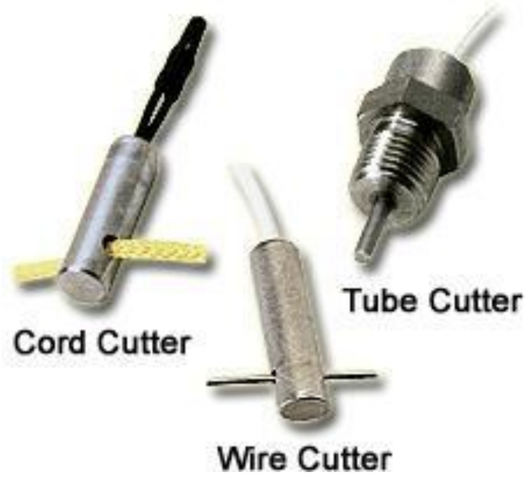
Thermal Knives

- Thermal knives are used to deploy Solar Arrays
- They need a specific driver and it is usually located in the PCDU
- 20.5 V applied during 60s

Pyro Actuators

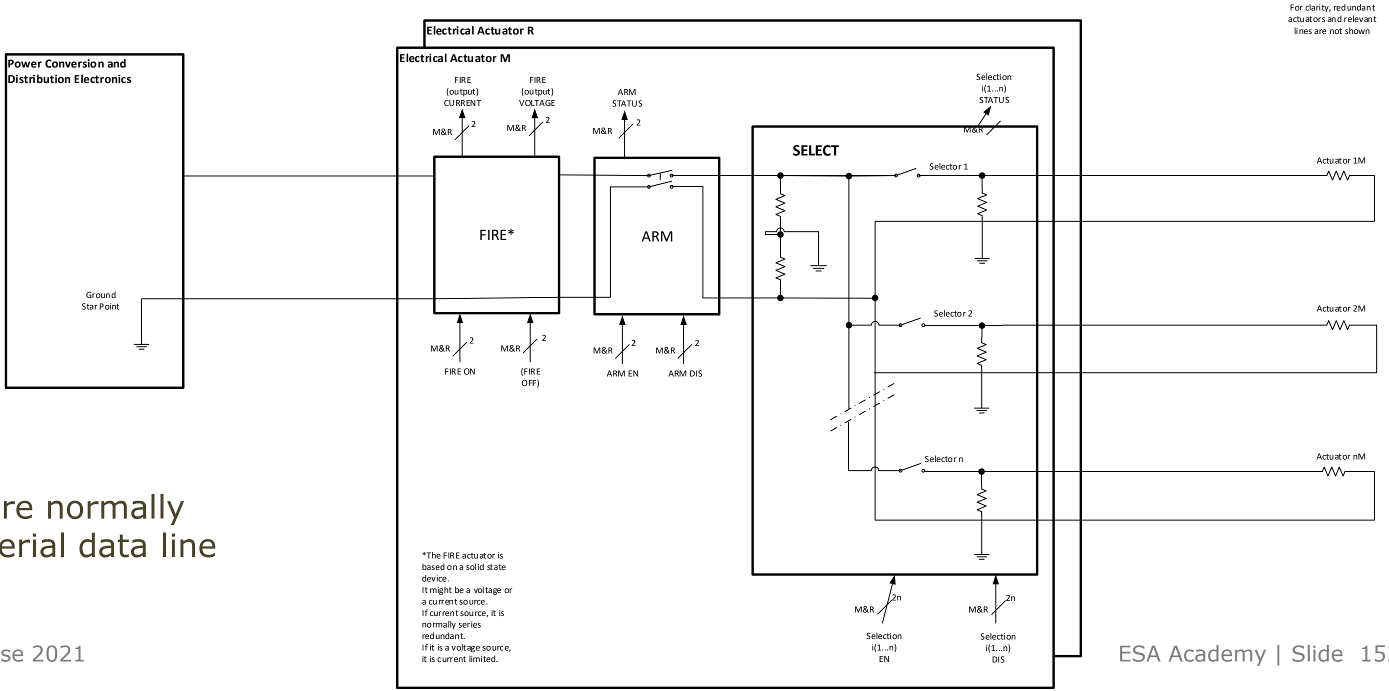
- These are also used to deploy appendages
- Electrically initiated chemical energy "fireworks"

- Complicated electrical loads. They might end up in short circuit.
- Standard resistance for a pyro is 1 Ohm.
- 5 to 6A are needed to activate them.



Thermal Knives / Pyro Drivers

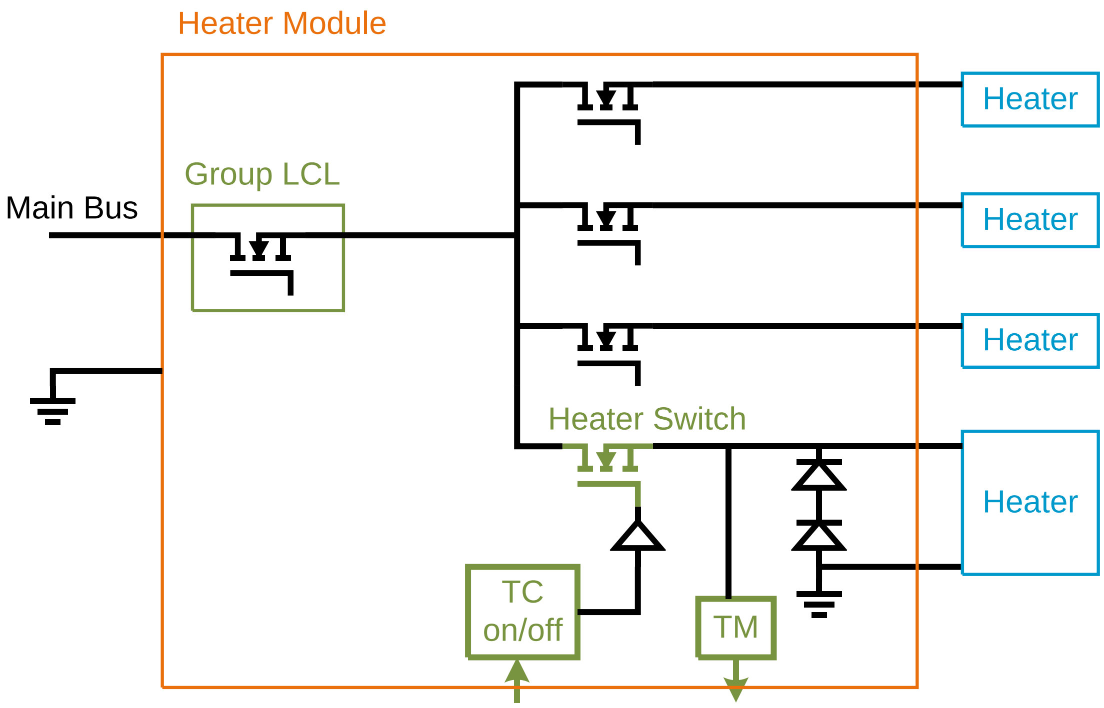
- 3 barriers to avoid inadvertent deployment
  - ARM (typically a latching relay, driven by a dedicated command line)
  - SELECT (relay of MOSFETs to deliver the actuation pulse to the correct device)
  - FIRE (linear or PWM converter to deliver the correct voltage/current to the initiator)



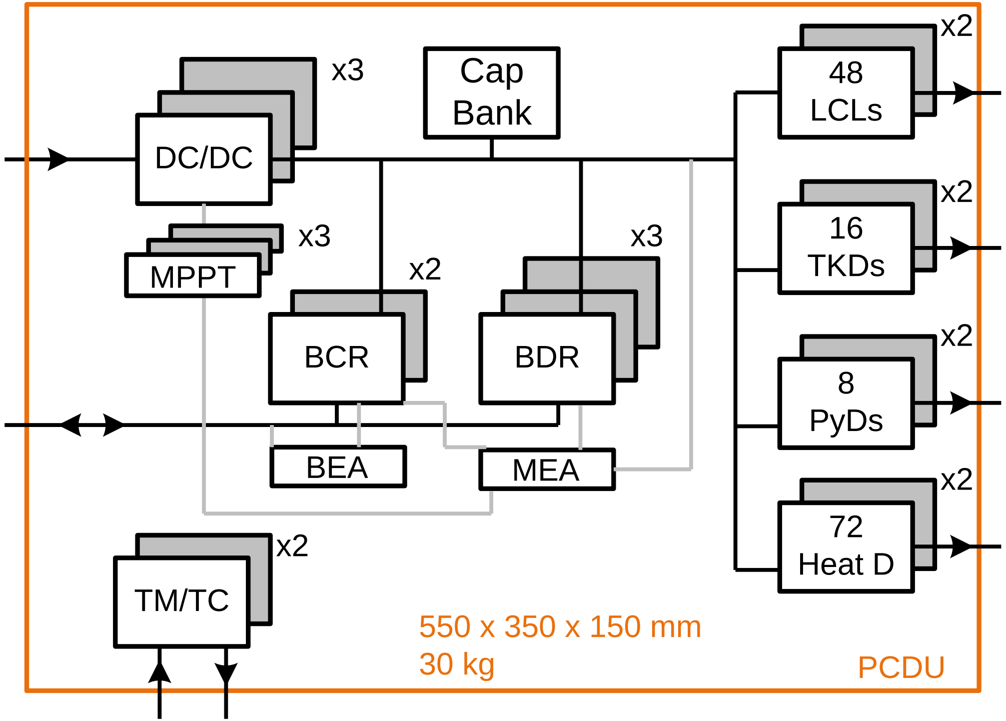
SELECT and FIRE are normally commanded by a serial data line

Heater Drivers

- Also inside the PCDU
- Provide power to the Satellite Heaters
- Controlled by the Thermal System
- In general, there are many of them



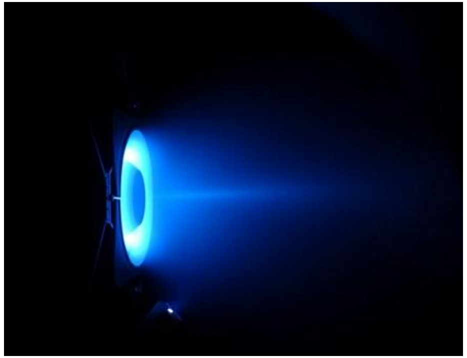
PCDU configuration example

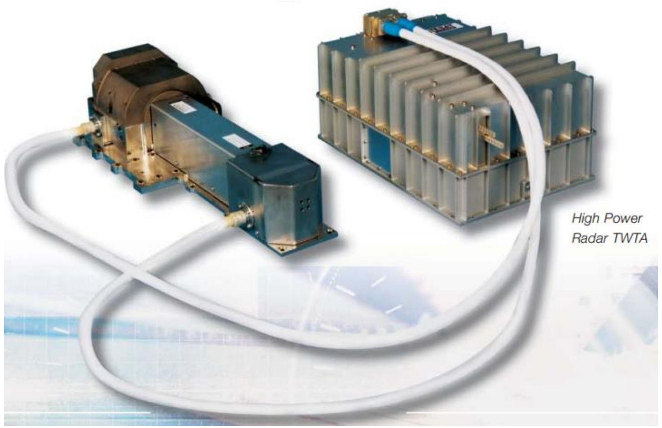


Nominal operation
after one failure

High Voltage Applications

- In a satellite there are also systems that need high voltage
  - Propulsion: Power supplies for Electric Thrusters (left picture)
  - Telecom: Power supplies for Travelling Wave Tubes (right picture)
- These applications need voltages between 1 kV and 30 kV
- Very specific techniques are needed in this case

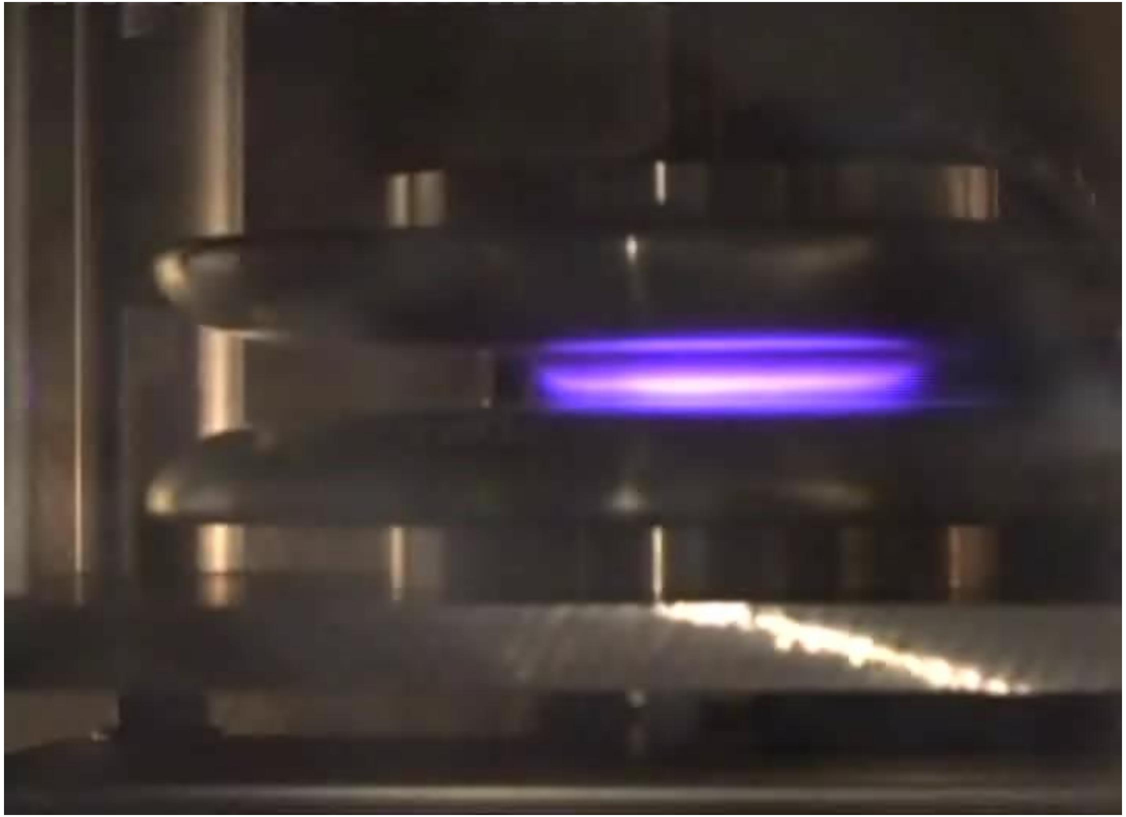




High Voltage

- What is considered high voltage in space?
- Which specific challenges are there to cope with?

⇒ The answers are in the embedded video:



Electro Magnetic Compatibility issues

- emission, susceptibility
- conduction, radiation
- typical issues together with the methods normally used to control them

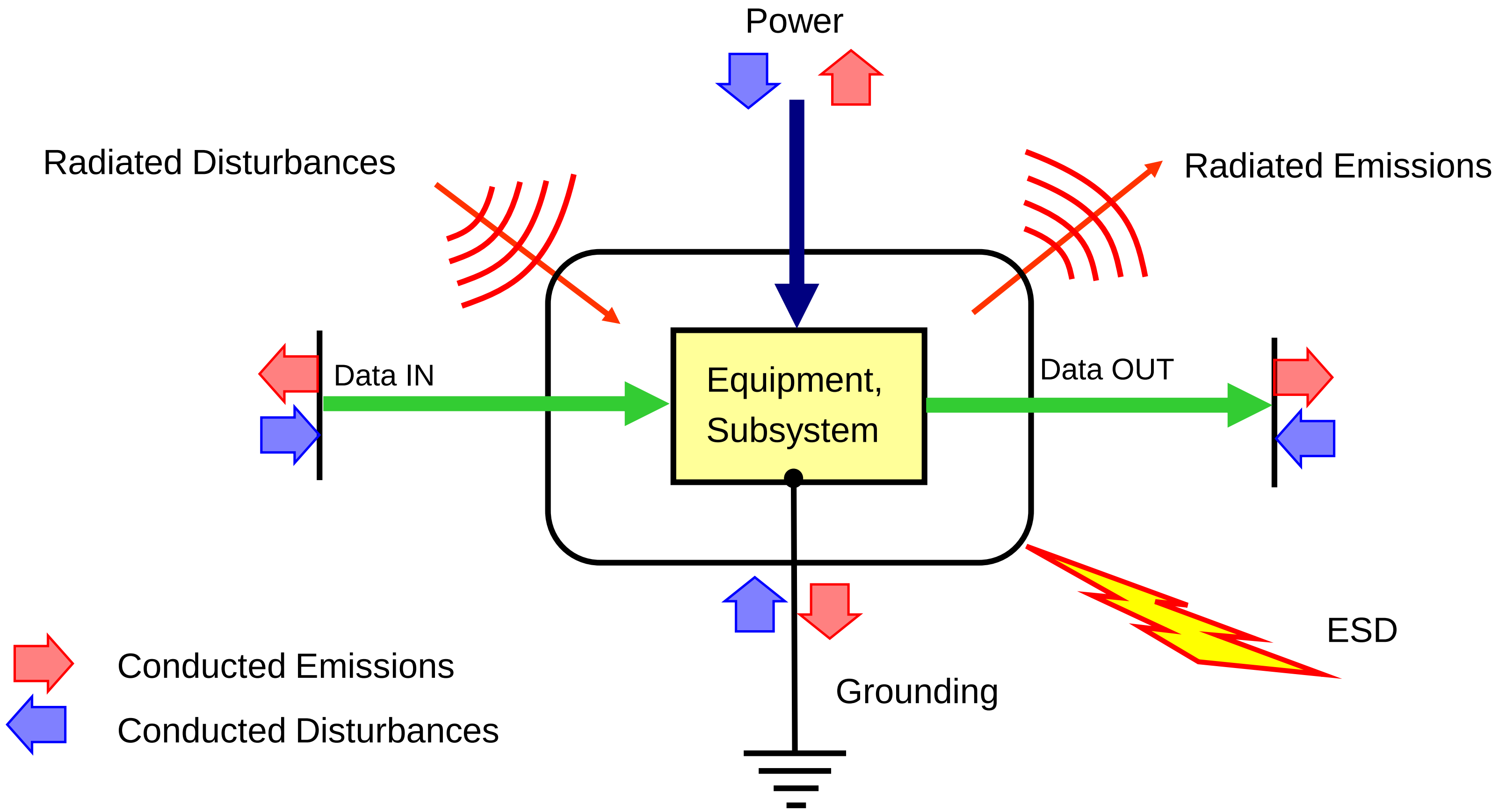
Definition of EMC

- Electromagnetic compatibility (EMC) of a space system or equipment is the ability to function satisfactorily in its electromagnetic environment without introducing intolerable electromagnetic disturbances to anything in that environment.

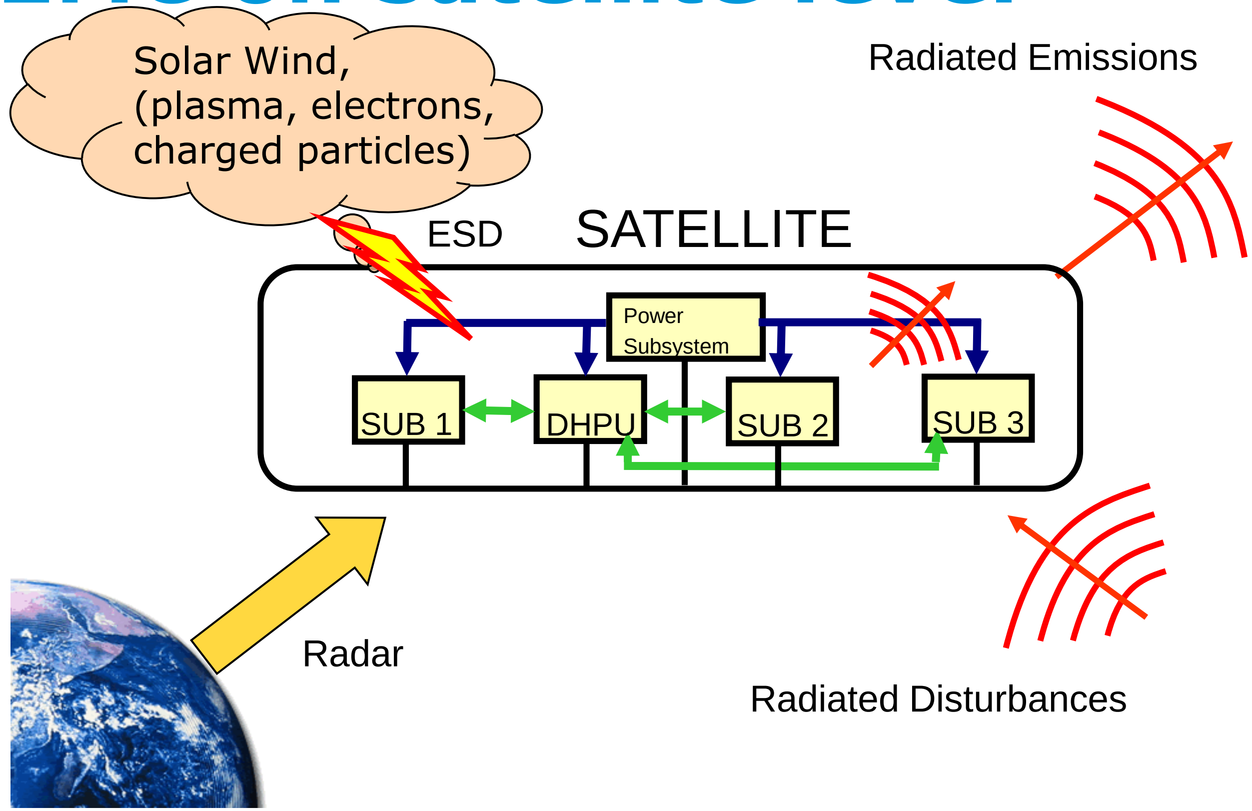
- Two main elements:
  - "to function satisfactorily" → immunity to electromagnetic noise
  - "without introducing intolerable disturbances" → electromagnetic emissions

- HOW EMC IS ACHIEVED
  - A certain immunity to electromagnetic noise is required → susceptibility test
  - Electromagnetic emissions need to be limited → emission measurements

EMC on equipment level



EMC on satellite level



LAUNCHER

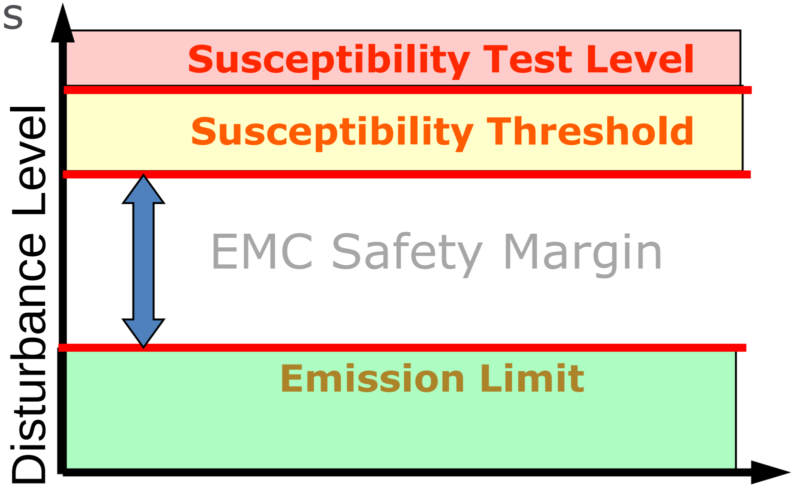
How to ensure EMC

- Define appropriate EMC Requirements and EMC safety margin
  - Conducted & Radiated Emission → Control of Emission limits
  - Conducted & Radiated Susceptibility → Control of Susceptibility levels

- Electrostatic and magnetostatic cleanliness

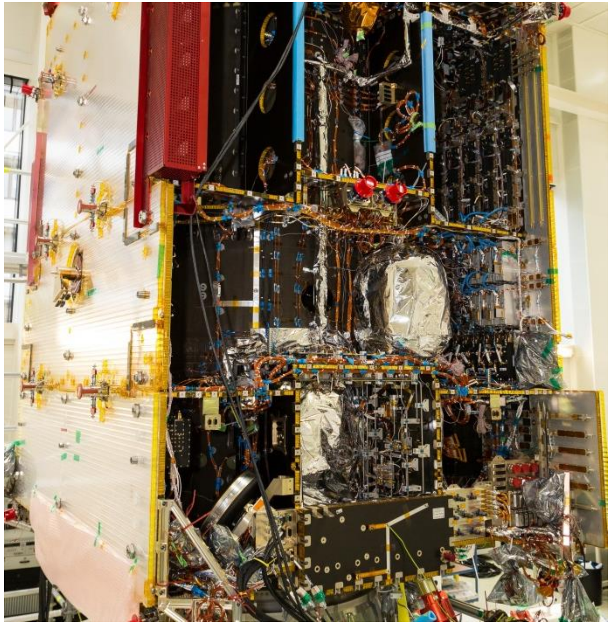
- Design, manufacture and integrate hardware accordingly and enforce good engineering solutions

- Verify the EMC Requirements: Analyses, Simulations and Testing



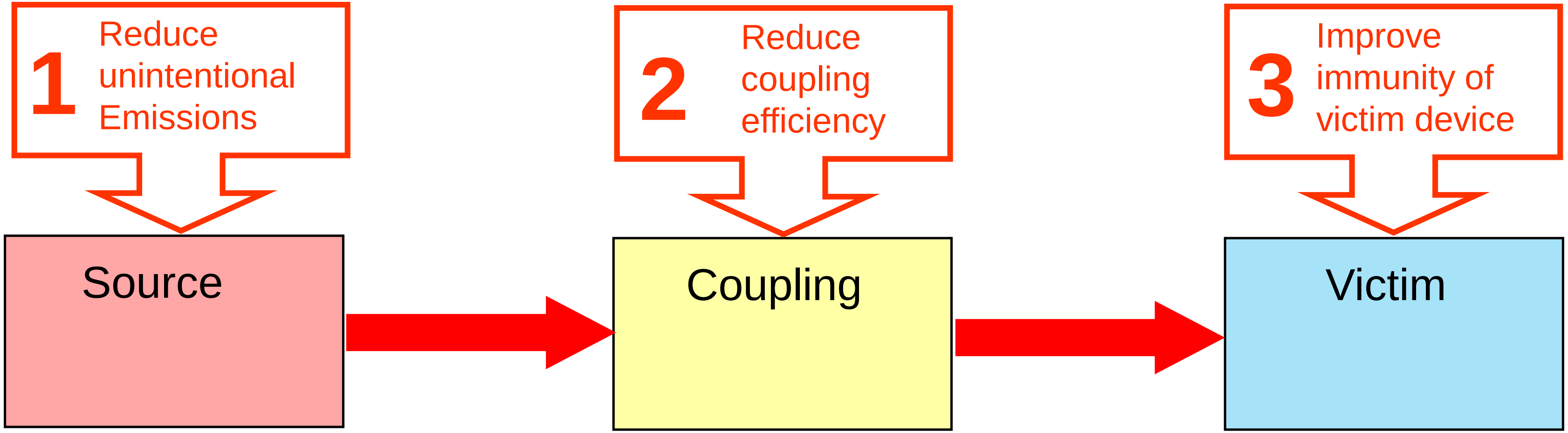
EMC Aspects

a) Intersystem EMC
(external sources,
launcher environment)



b) Intrasystem EMC (internal sources and victims)

The EMI coupling model



Device or Equipment generating disturbances (intentional or intentional)

Conducted

Radiated

Near field:

- capacitive (electric)
- inductive (magnetic)

Far field:

- non-ionizing radiation

Device or Equipment being disturbed (susceptible to the disturbance)

Grounding

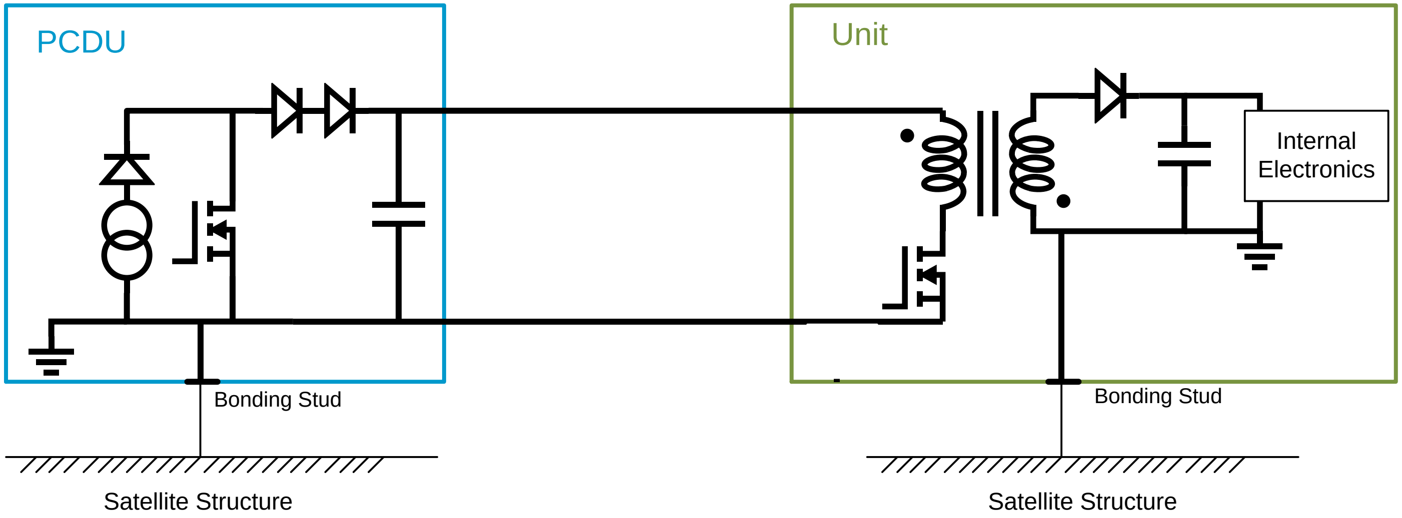
- A well-posed electrical grounding architecture is the primary ingredient for achieving Electromagnetic Compatibility at spacecraft level in a cost-effective way.

- Grounding provides a common voltage reference to spacecraft electronic equipment and subsystems while minimizing Electromagnetic Interference and unintentional interactions between them.

- Different grounding concepts have pros and cons, including reliability considerations that are the subject of trade-off studies since the conceptual phase of the spacecraft definition. A conclusion must be reached before the equipment EMC design takes place.

(One) Grounding concept

- Distributed single point grounding… one of the most common on ESA satellites



Introduction the ECSS (E-20) standards

- Codify the information and knowledge relevant to electrical and electronic into relevant requirements addressing
  - ***Reliability***
  - ***Functionality***
  - ***Performance***

- Note…
  - ***Performance*** dictate the necessary engineering/scientific targets for mission success… but their verification can hardly be forgotten
  - ***Functionality*** might cover non obvious features of the equipment/subsystem/system, and it might also be easy that proper verification is overlooked
  - ***Reliability*** requirements are <u>hidden</u> design drivers, and their verification might be <u>easily</u> overlooked!

Introduction the ECSS (E-20) standards

Topic	Standard (ST) / Handbook (HB)
General design and interface, EPS, power & analogue electronics, some EMC aspects (EMC control plan)	ECSS E ST 20 C Rev. 1
Standard Power interfaces (for the moment limited to Power Distribution by LCLs)	ECSS E ST 20 20C and ECSS E HB 20 20 A
Photovoltaic Assemblies and Components	ECSS E-ST-20-08C Rev. 1
Li-ion battery testing handbook	ECSS-E-HB-20-02A
High voltage engineering and design handbook	ECSS-E-HB-20-05A
EMC	ECSS-E-ST-20-07C, rev.1
EMC (Handbook)	ECSS-E-HB-20-07A

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- Geoff Dudley, Energy Storage
- Johannes Wolf, EMC
- Keith Stephenson, Nuclear Systems

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