ELECTRICAL POWER SUB-SYSTEM

PRESENTED BY : PIERRICK IGOT

SLIDES ORIGINALLY WRITTEN BY: VINCENT LEMPEREUR

INTRODUCTION

A THALES ALENIA SPACE

B THALES ALENIA SPACE IN BELGIUM

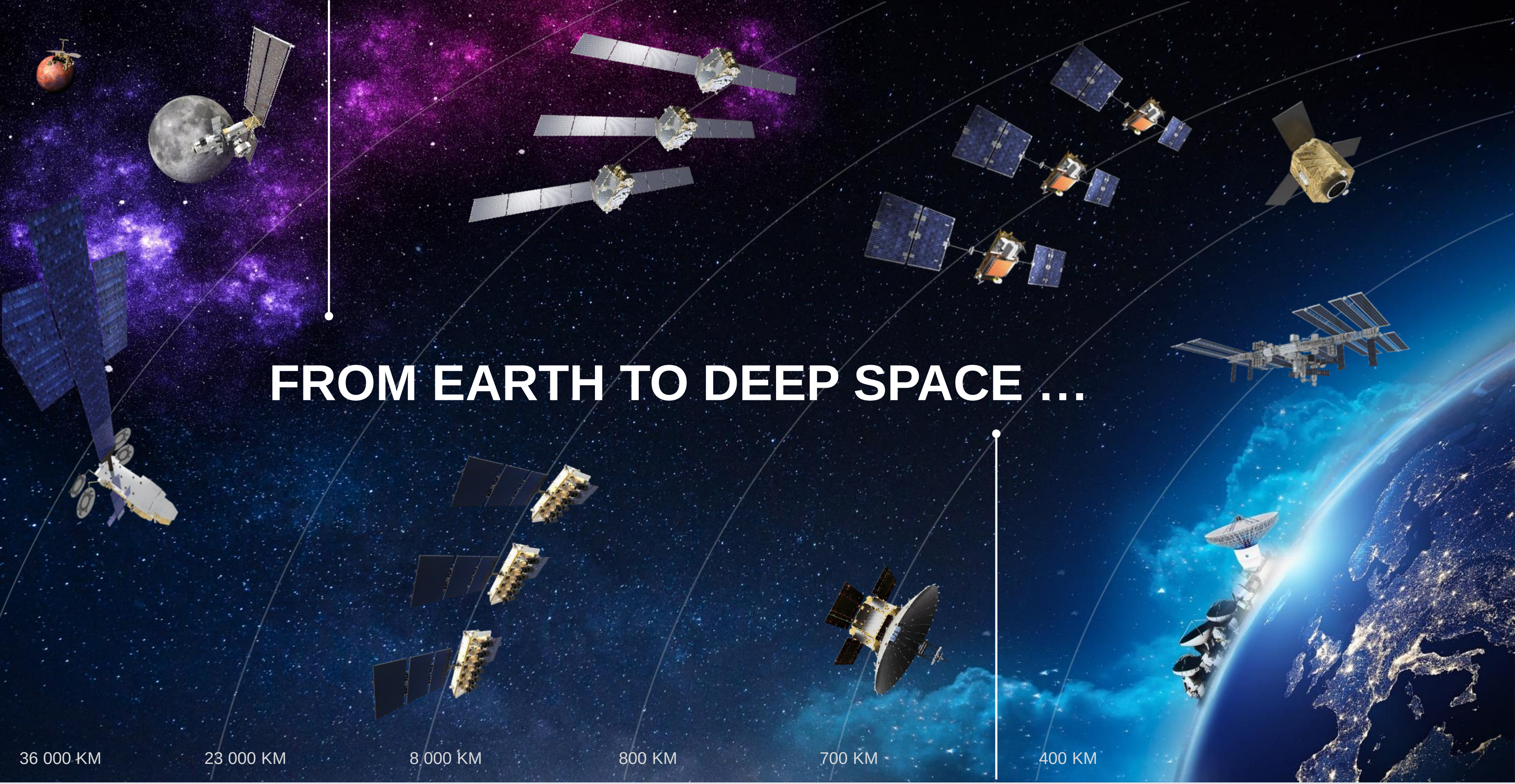
C Presentation of myself

D EPS: general informations

INTRODUCTION

A THALES ALENIA SPACE

FROM EARTH TO DEEP SPACE …



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SPACE FOR LIFE ///

SPACE TO CONNECT

THALES ALENIA SPACE IN 2022

2,2
BN € SALES

8,500
EMPLOYEES

18 SITES
WORLDWIDE

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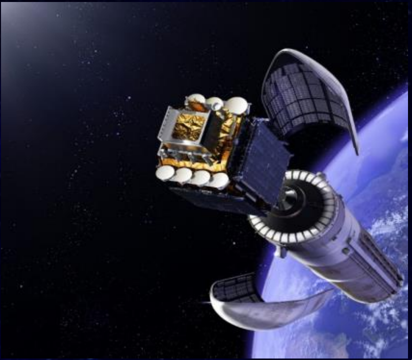
INTRODUCTION

B THALES ALENIA SPACE IN BELGIUM

THALES ALENIA SPACE IN BELGIUM



A world leader in power electronics for satellites and launchers

More than 750

3 sites

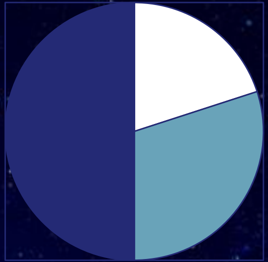
Charleroi
Leuven
Hasselt

Belgian leader in the space sector

>60 years of experience in Space

Operators ~20%

Engineers & managers ~50%



Support functions ~30%

~250 units / year

NEW SPACE IN THALES ALENIA SPACE IN BELGIUM

Software-defined **Long term sustainability**
Electrical propulsion leader
Federate High power/High voltage integration
Automotive components **Micro-solutions**
Agile Techno-push **New ways of working**
System approach Automatised manufacturing **Partnership**
Open-innovation Explore

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A Fully Flexible Software-Defined Satellite

Full Reconfiguration in Orbit

Unrivaled Mission Performances

Extensive Hosting Capacity

Smart Operations

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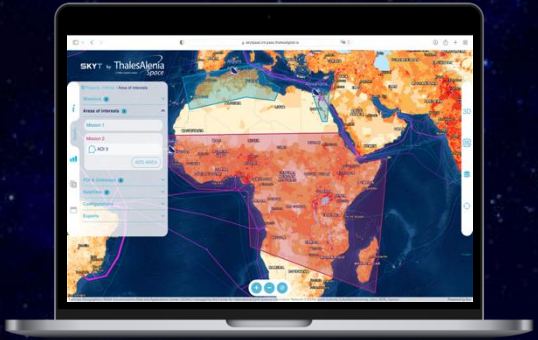
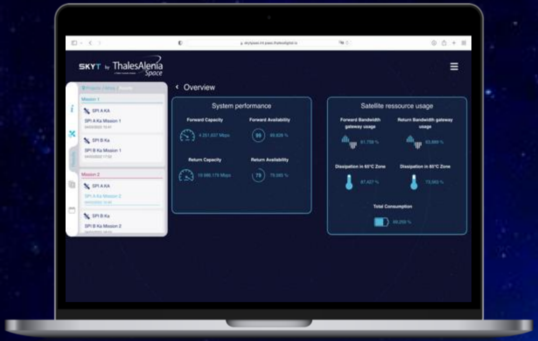


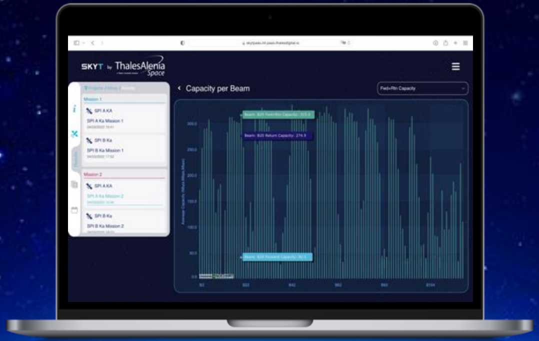
Create Areas of Interests

Configure your satellite

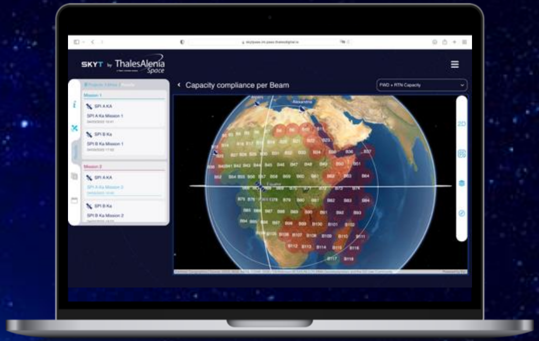
Configure your coverage



Visualise the system budget



Visualise results with barcharts



Visualise results in 2D/3D Maps

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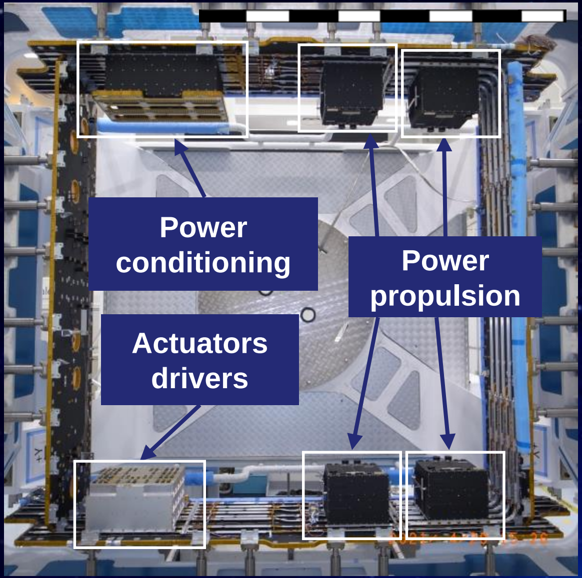
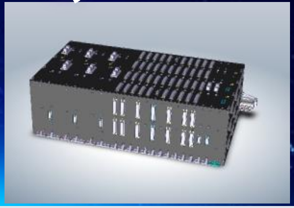
OUR SOLUTIONS IN BELGIUM

A COMPLETE NEW GENERATION FOR **AVIONICS SUBSYSTEM**

2020 Satellite



Space Inspire



same scale

- + More power
- + More interfaces
- + More functions
- + More flexible
- + Less mass
- + Automotive components

- + **Much** more faster to build

+ **Cost** efficiency

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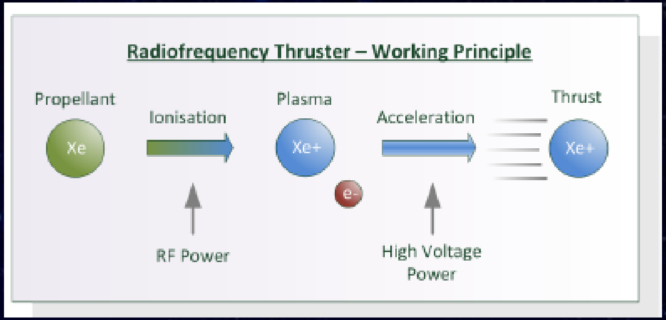
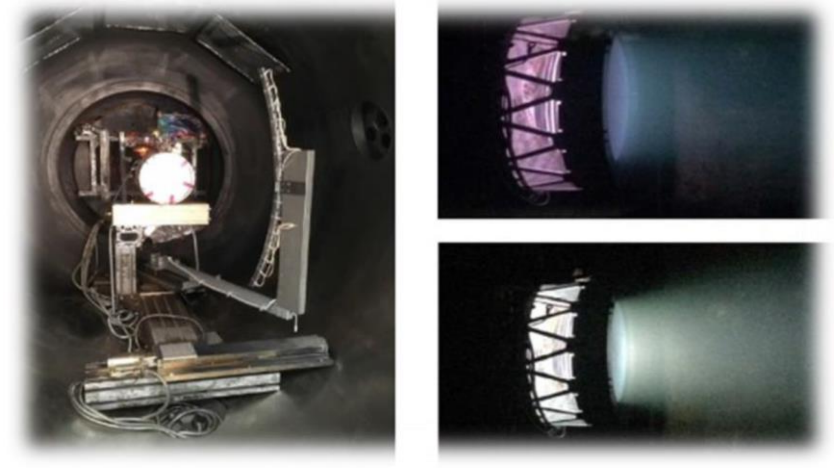
LEADER IN ELECTRICAL PROPULSION

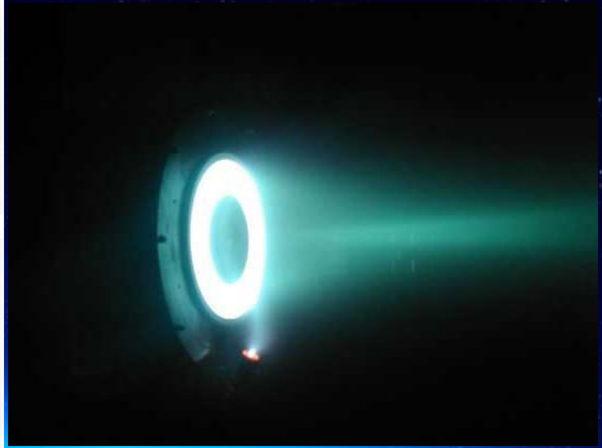
From **Chemical** propulsion to full **Electrical** propulsion with **G**rid **I**on **E**ngine technology

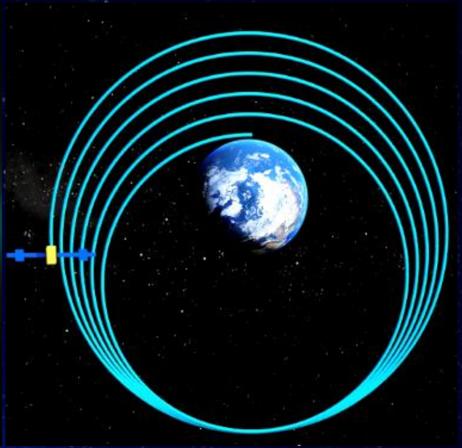


We provide **power for thrusters**

Orbit raising | Station keeping







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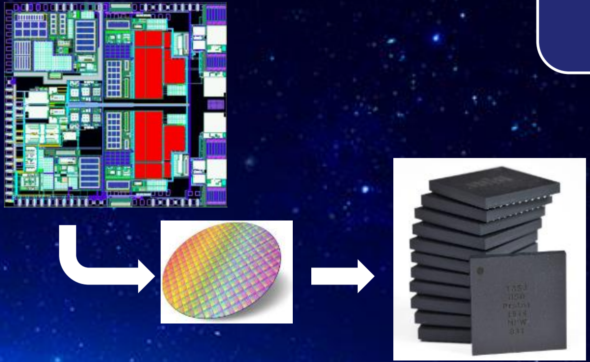
SOFTWARE INSIDE & MICRO-SOLUTION

Proprietary micro-controllers
in all products
with **adapted software solutions**

Software defined solutions – Flexibility – Live Reconfiguration

Strong **partnerships** with leading-edge companies

Proprietary GaN driver
for all power switches



High performance – High integration

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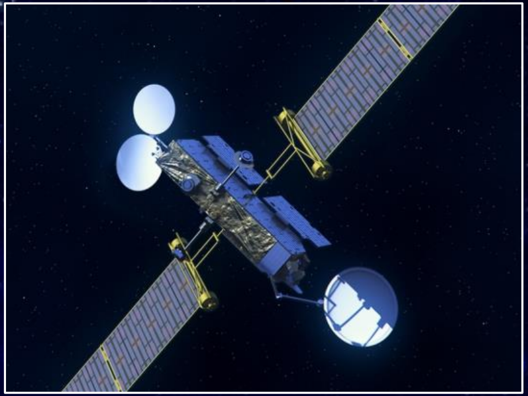
ThalesAlenia Space

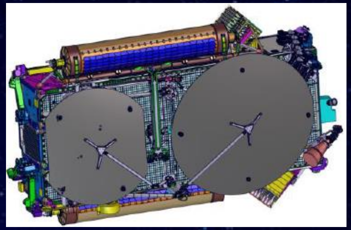
OUR SOLUTION

A COMPLETE NEW GENERATION FOR **SOLAR GENERATOR**

Flexible Solar Array

Smaller footprint in launcher
for **bigger solar cells surface**

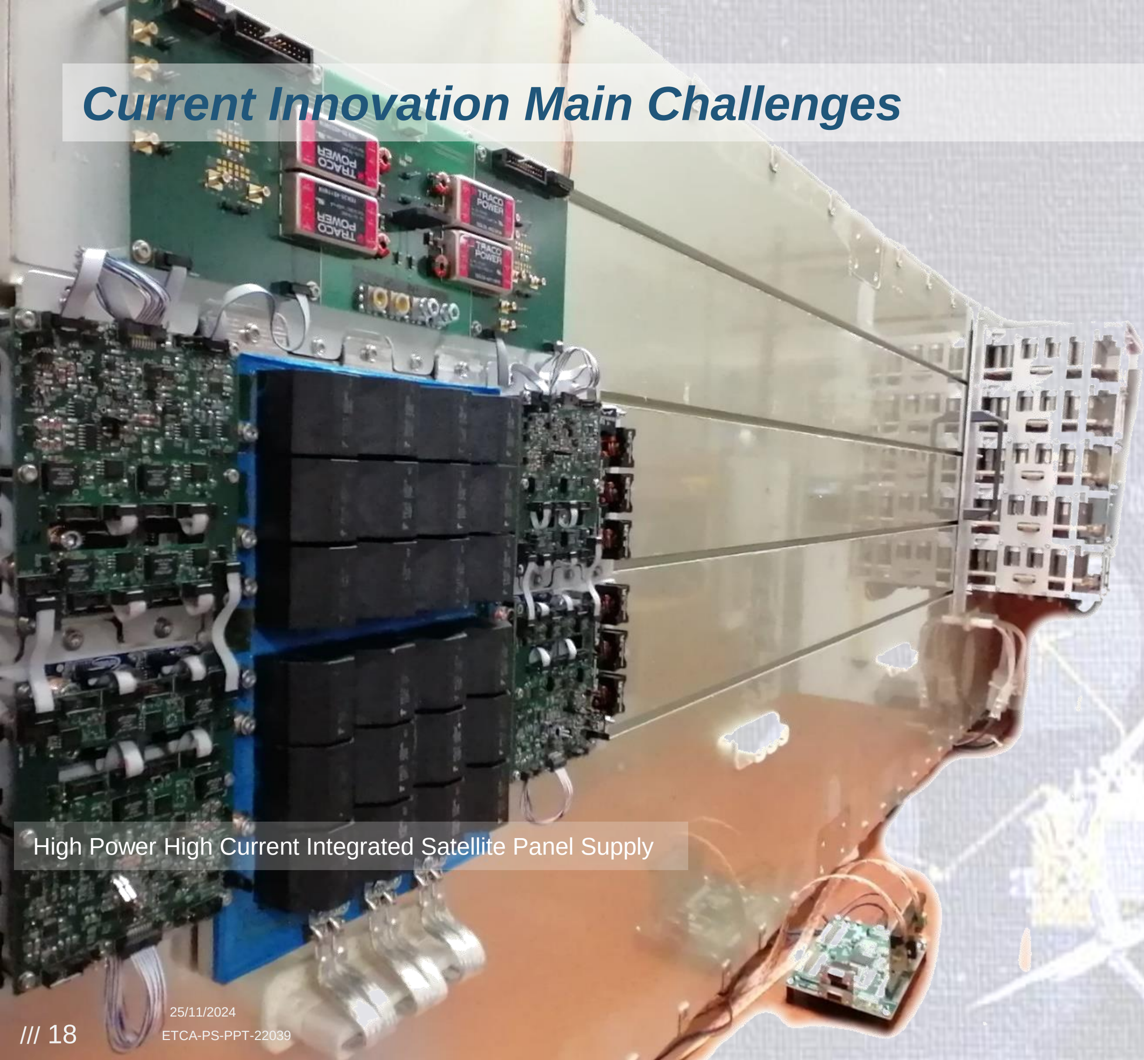




AND OTHER NEW TECHNOLOGIES FOR HIGH VOLTAGE AND HIGH POWER SOLUTIONS…

- Break-through technologies developed since 5 to 10 years, becoming mature now
- Matured in fruitful partnerships with Belgian and European SMEs, in other industries
- Successfully combining « open innovation » in a « fierce competition » environment

Current Innovation Main Challenges

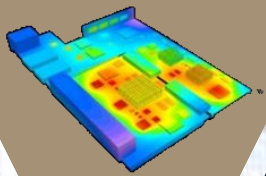


High Power High Current Integrated Satellite Panel Supply

Mass Reduction

Hgh Integration

Thermal Management



Embedded AI

WANNA JOIN THE TEAM ?

/// Internship / Jobs

- Feel free to apply !
- For an internship, don't forget to precise the period and if a Master's thesis is included
- https://careers.thalesgroup.com/fr/fr/home

INTRODUCTION

C Presentation of myself

INTRODUCTION

/// Training

/ GRADUATED AS ELECTROMECHANICS ENGINEER FROM UCLOUVAIN

/// Professional career in TAS-Belgium at Charleroi

- Specialized in hardware design in Electronics for satellite platform equipments

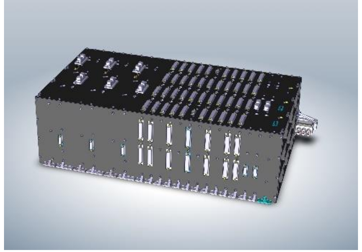
/ ELECTRONIC DESIGNER

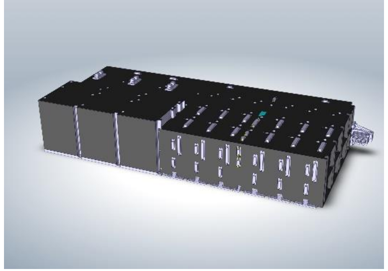
- Power conditioning and distribution (PCU/PCDU) Exomars & SWOT project

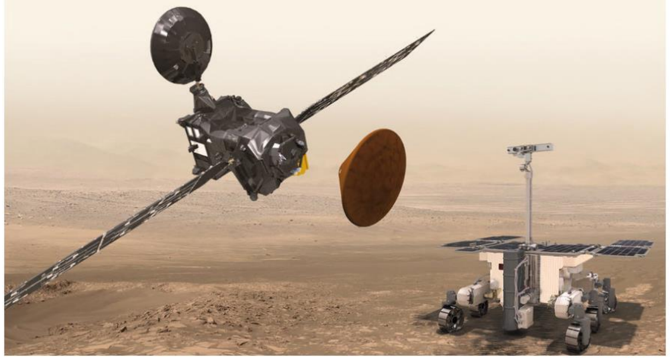
/ PROJECT DESIGN AUTHORITY

- PCDU modules for Space Inspire (whole HPU + module in ACE)

/ POWER CONDITIONING PRODUCT LINE ARCHITECT

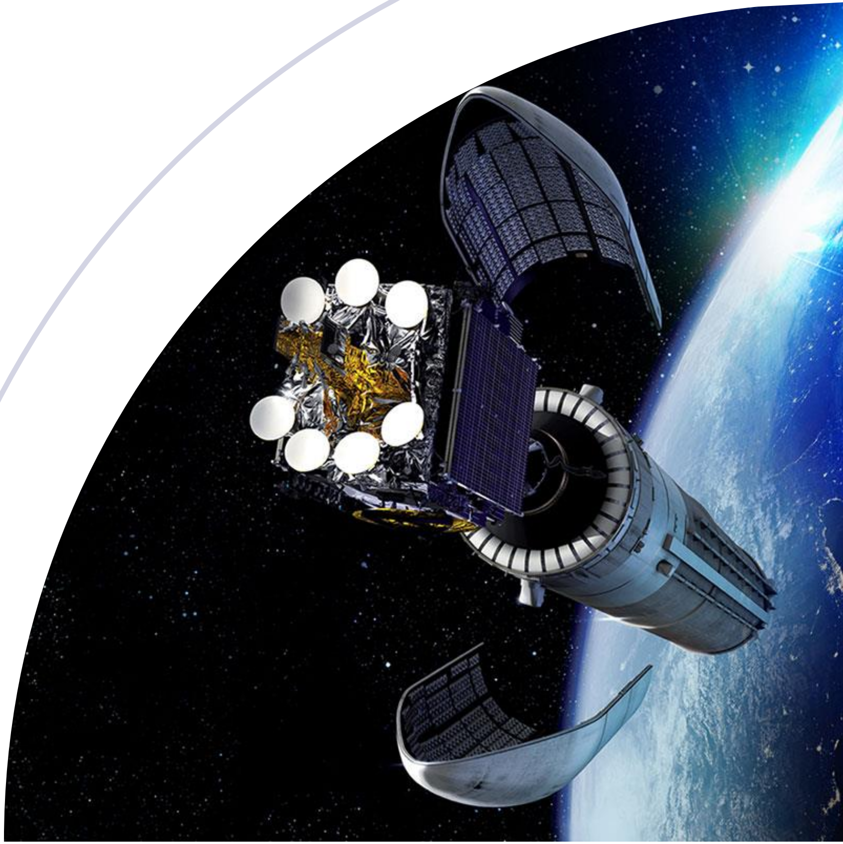




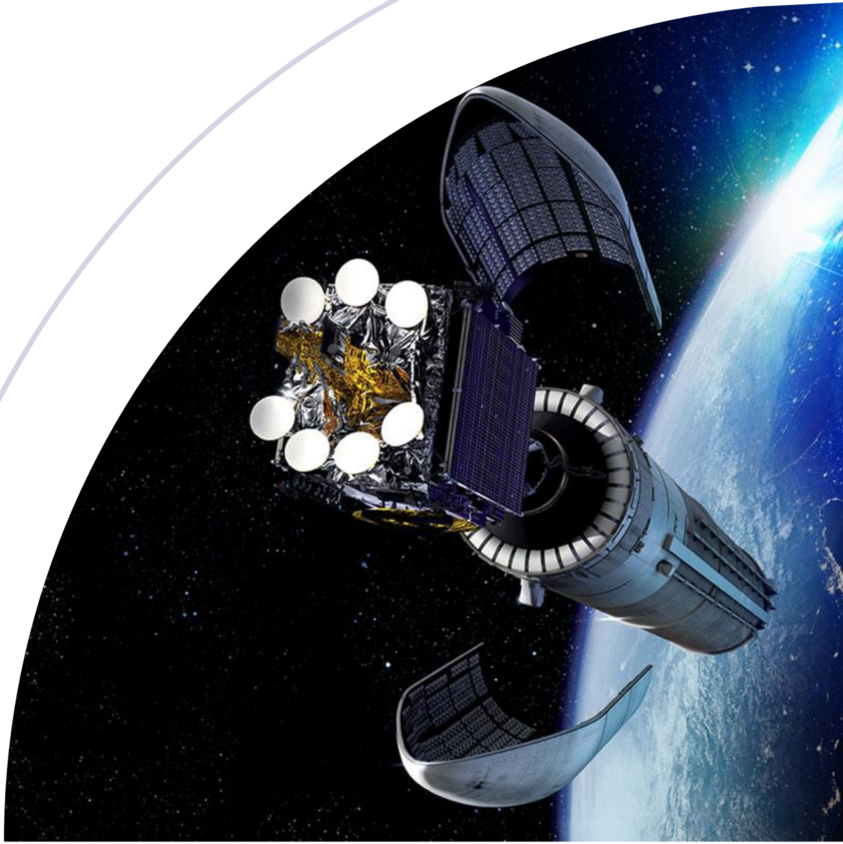


Source: Le Temps

AGENDA



AGENDA



INTRODUCTION / EPS GENERAL INFORMATION

A satellite is made of…

P/F (Platform)

- MECHANICAL & THERMAL STRUCTURE
- ELECTRICAL SYSTEM, AVIONIC, PROPULSION
- ON-BOARD COMPUTER, SOFTWARE, REMOTE CONTROL
- ENERGY SOURCES: SOLAR, BATTERIES, FUEL

P/L (Payload)

- ANTENNAS, TWTA, …
- CAMERA, ALTIMETER, RADAR, DETECTORS, …
- CLOCK, SCIENTIFIC INSTRUMENTS, …

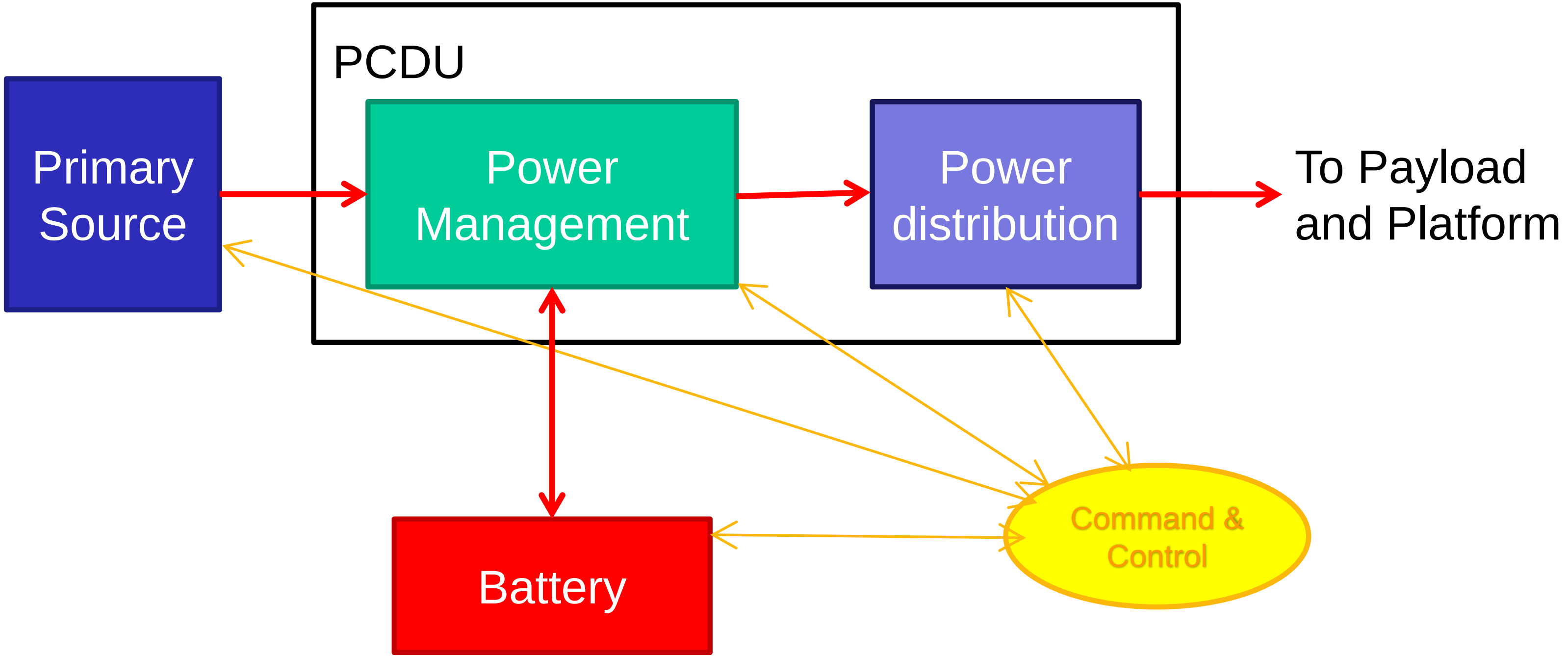
ELECTRICAL POWER SYSTEMS

Satellite's Electrical Power Subsystem (EPS) shall

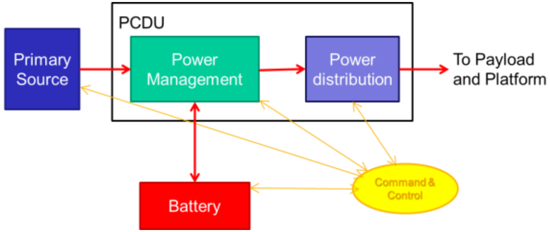
- provide electrical power all satellite's units
- Storage energy to power units in case of orbital night phases, transient phases and peak power demand
- autonomously manage the available power in order supply units and charge the battery
- fulfill some distribution requirements providing ON/OFF protected power lines, heater supply (for S/C thermal control needs) and commanding pyro lines (e.g. SA and antenna deployment)
- Note: power system failure means the loss of mission

INTRODUCTION / EPS GENERAL INFORMATION

General functional block diagram



INTRODUCTION / EPS GENERAL INFORMATION



Functions (1/2)

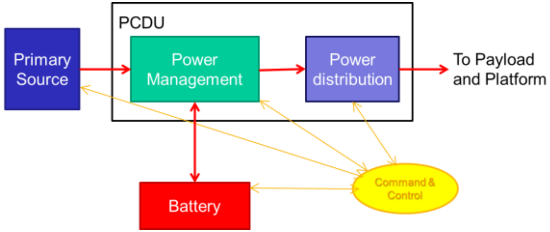
- POWER GENERATION
  - The power is generated from different sources ('fuel') or combination of them: the Solar radiant energy (solar cells via photovoltaic effect), Chemical (piles – fuel cells), nuclear (RTG), mechanical (reaction wheels), …
  - **Primary sources** convert 'fuel' into electrical power

- ENERGY STORAGE
  - The energy is generally stored under an electro-mechanical form and retrieved under an electrical form
  - The storage of the energy is done by a **secondary source**, when the primary system's energy is not available or insufficient

- CONDITIONING AND REGULATION
  - This function covers everything which is required to adapt the primary sources to the need of users 'equipment' (constant voltage, battery charge…)

- DISTRIBUTION
  - To distribute the conditioned power to users
  - DC/DC voltage converters
  - ON/OFF switches (sometimes
  - Does not include the harness

INTRODUCTION / EPS GENERAL INFORMATION



Functions (2/2)

- **PROTECTION**
  - To avoid a propagation of failures or any Single Point Failure
  - Protections against short-circuits
    - Fuses
    - Circuit breakers

- **CONTROL**
  - Observing parameters
    - Current, voltages, temperatures, status, …
  - Information are transmitted to the Ground by telemetry for mid-term and long-term monitoring
  - Information are transmitted to the On-Board Computer for real-time monitoring

- **COMMAND**
  - Configuration setting (nominal, safety, recovery, …)
  - Parameters
  - ON/OFF

AGENDA

1. Introduction

- EPS – GENERAL INFORMATION
- EPS DESIGN DRIVERS

2. Primary power sources

- SOLAR CELLS & SOLAR ARRAYS
- OTHERS

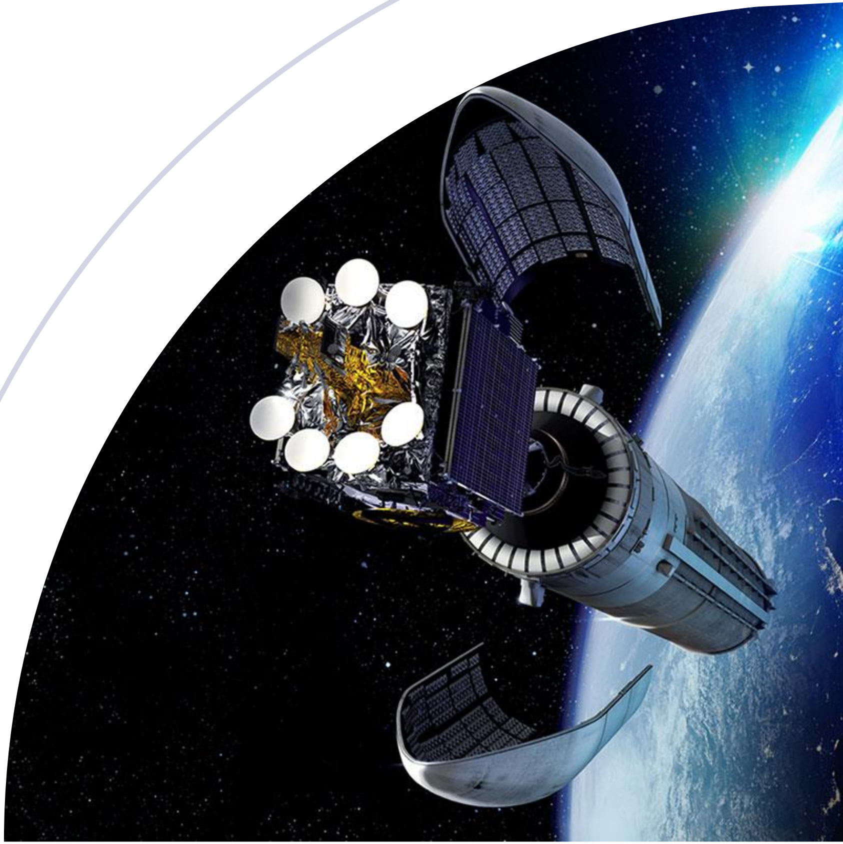
3. Secondary power sources - batteries

4. Power Management, Control & Distribution

- ARCHITECTURE
- PCU / PCDU EXAMPLES

5. Power budget - practical exercise

6. Conclusions



INTRODUCTION / EPS DESIGN DRIVERS

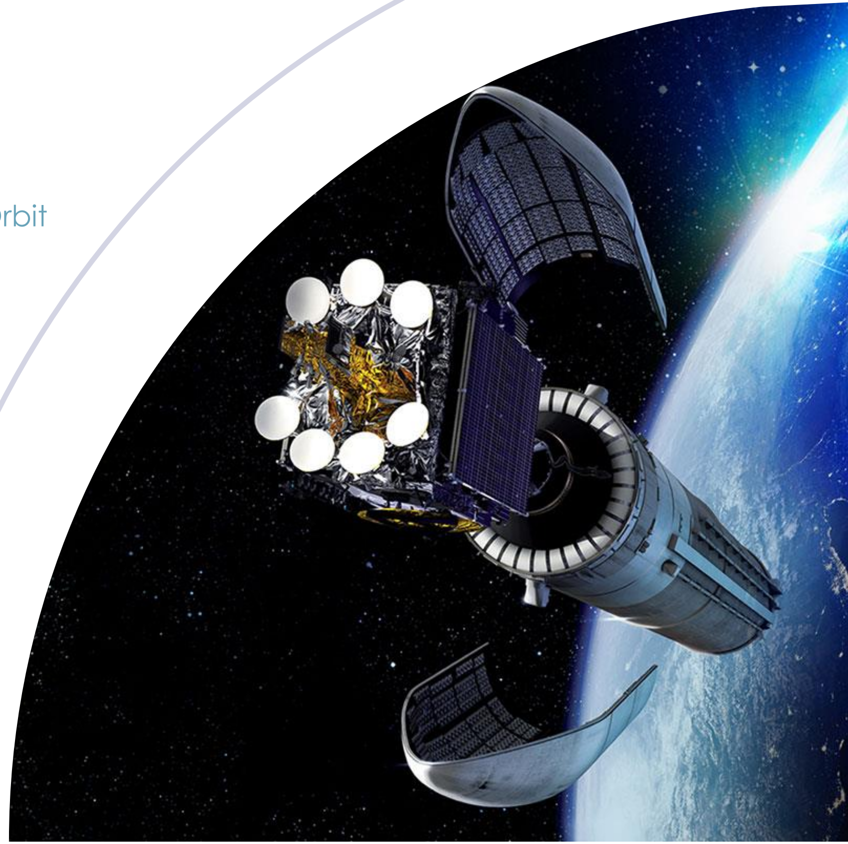
System drivers / Synthesis

The orbit

- Low Earth Orbit (LEO), geostationary (GEO), Mean Earth Orbit (MEO), Sun Synchronous Orbit (SSO), Sun Centric (Interplanetary), …

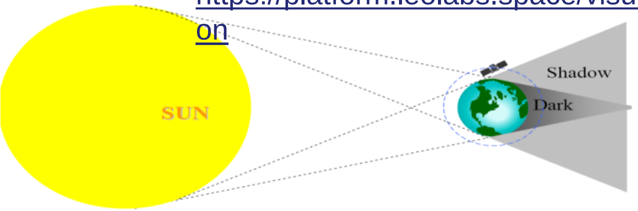
The mission

- (Life) duration
- Energy budget
  - Mission profiles
  - Payload needs
  - Max and Mean power
  - Orientation (attitude) of the satellite
- Reliability requirements



INTRODUCTION / EPS DESIGN DRIVERS

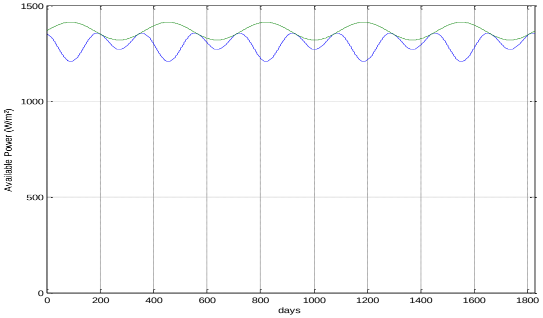
System drivers / Orbits



https://platform.leolabs.space/visualization

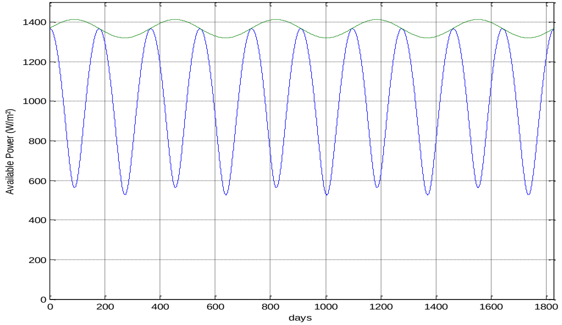
LEO (Low Earth Orbit) / Scientific applications – Earth observation

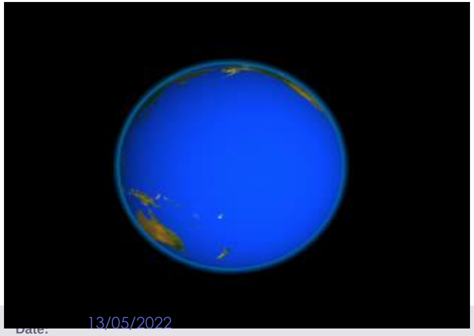
- **ORBIT**
  - Altitude: between 350 and 1000 km
  - Duration:~2 hours
  - Low sensitivity to radiations
- **ECLIPSES**
  - High variability versus the orbit selection
  - Up to 40 % of eclipse duration
  - Thousands of cycles along mission duration

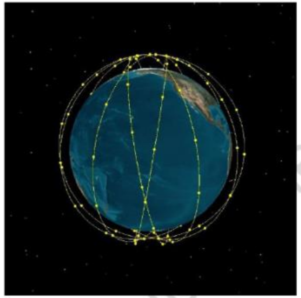


SA flux (sun-synchronous orbit)
- > Min SA flux = 1220 W

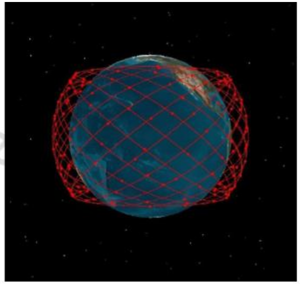
SA flux (polar orbit)
- > Min SA flux = 520 W







Polar Orbit

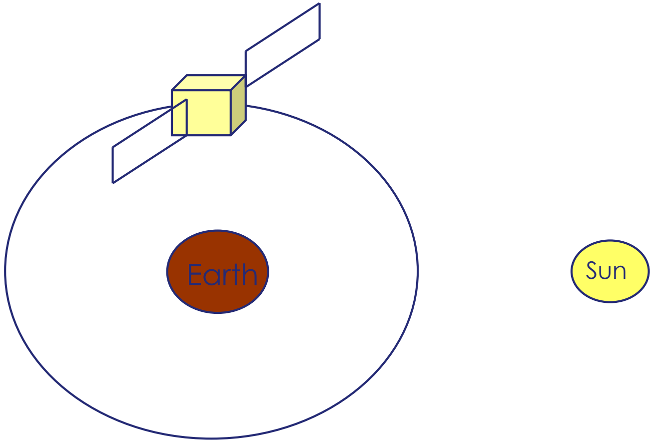


Inclined Orbit

INTRODUCTION / EPS DESIGN DRIVERS

SYSTEM DRIVERS / ORBITS

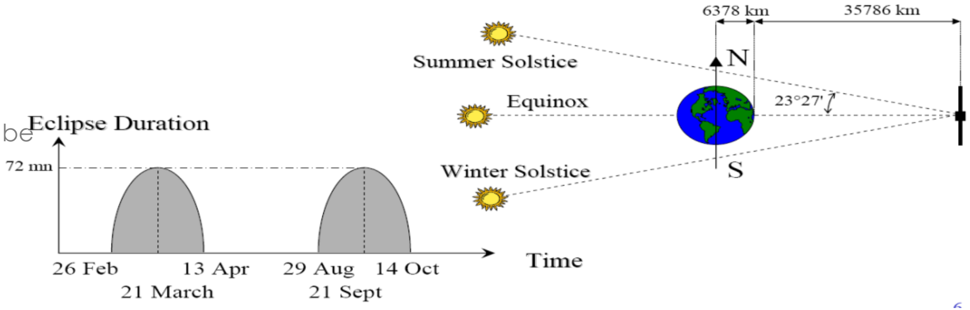
GEO (Geostationary Orbit): Telecom applications



- **ORBIT**
  - Type: Circular
  - Altitude: 35786 km
  - Duration: 24 hours
  - Medium sensitivity to radiations
- **ECLIPSES**
  - Less then 1% of mission duration
  - Only during equinoctial periods
  - From few to 72 min max

… DURING LIFETIME … BUT UP TO 6 MONTHS EOR WITH ELECTRICAL PROPULSION DRASTICALLY MODIFY THE SITUATION

- Increased number of longer eclipses
  - Thermal cycling more severe
  - Ratio charge / discharge impacted
  - Higher battery DoD (especially if thrust has to be performed in night mode)
- More stringent radiative environment

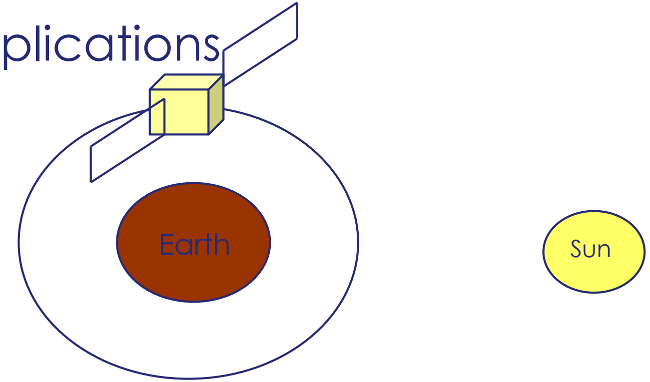


- **MISSION DURATION**
  - 15 years
- **EXAMPLE(S): SPACEBUS BASED SATELLITES, …**

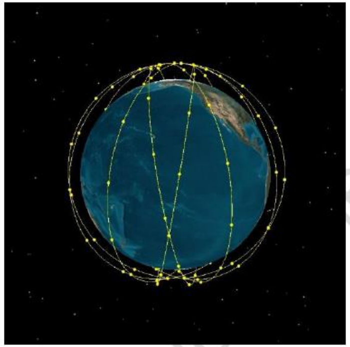
INTRODUCTION / EPS DESIGN DRIVERS

SYSTEM DRIVERS / ORBITS

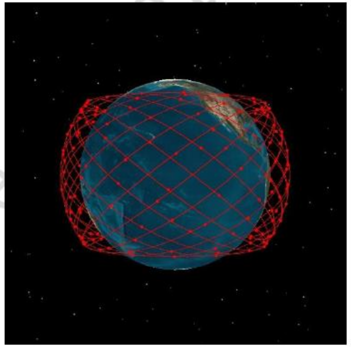
MEO (Medium Earth Orbit): GNSS/TELECOM applications



- **ORBIT**
  - Type: Circular
  - Altitude: 1000 to 20000 km
  - Duration: 12 hours
  - Medium to high sensitivity to radiations (according to orbit height)
- **ECLIPSES**
  - Duration: up to 1 hour
- **MISSION DURATION**
  - Up to 15 years
- **EXAMPLE(S): GLOBALSTAR, GALILEO, IRIDIUM, …**



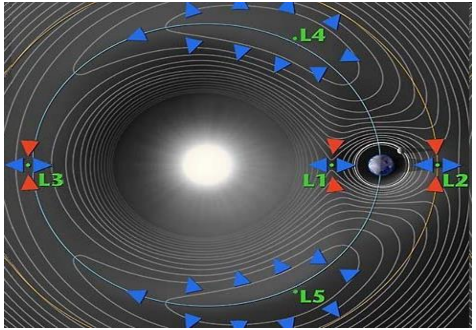
Polar Orbit



Inclined Orbit

INTRODUCTION / EPS DESIGN DRIVERS

SYSTEM DRIVERS / ORBITS



Lagrange Point: Scientific applications - ESA

- **POINTS WHERE THE COMBINED GRAVITATIONAL PULL OF TWO LARGE MASSES PRECISELY COMPENSATE THE CENTRIPETAL FORCE REQUIRED TO ROTATE WITH THEM (ANALOGY WITH THE GEOSTATIONARY ORBIT)**
  - Distance from earth for L1,L2: 1.5*106 km
- **ECLIPSES**
  - None
- **MISSION DURATION**
  - 3 years
- **EXAMPLE(S): HERSCHEL (L2), PLANCK(L2), GAIA(L2),…**

Interplanetary

- Challenge : management of solar flux, which decreases

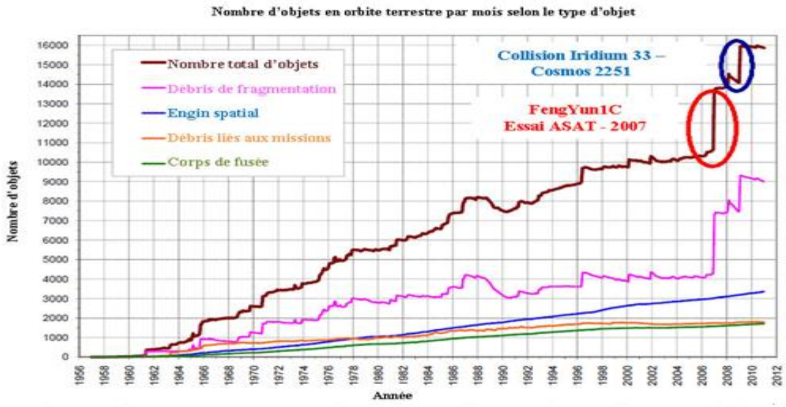
with the square of the distance to the sun

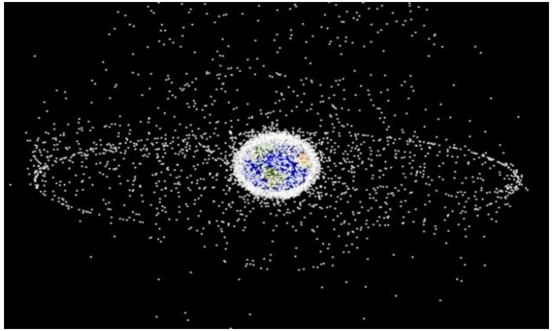
	Distance (AU)	Solar fluw (W/m²)
Mercury	0.39	$9.3\ 10^3$
Earth	1.0	$1.36\ 10^3$
Mars	1.5	582
Jupiter	5.2	48.7
Saturn	9.5	13.5

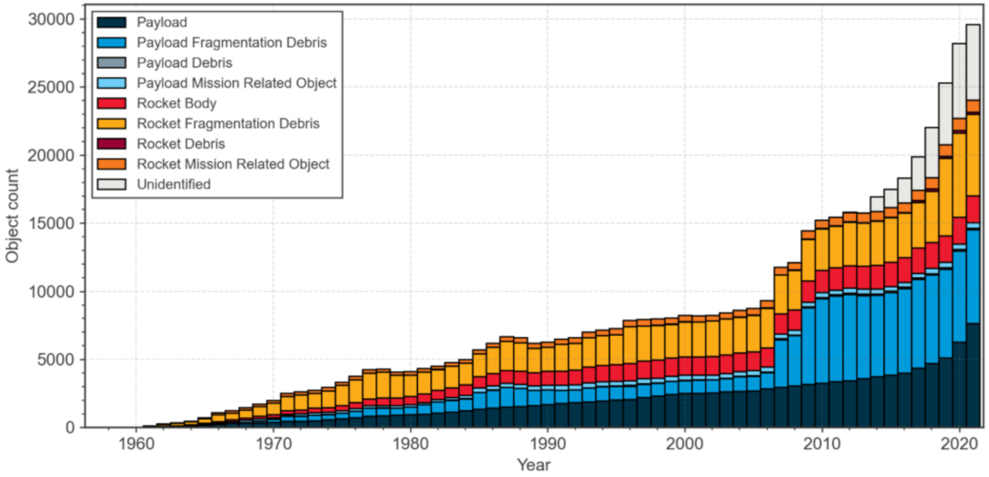
INTRODUCTION / EPS DESIGN DRIVERS

System drivers / Orbits

M-METEORITE & DEBRIS



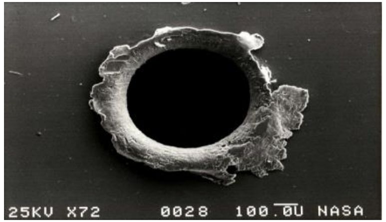




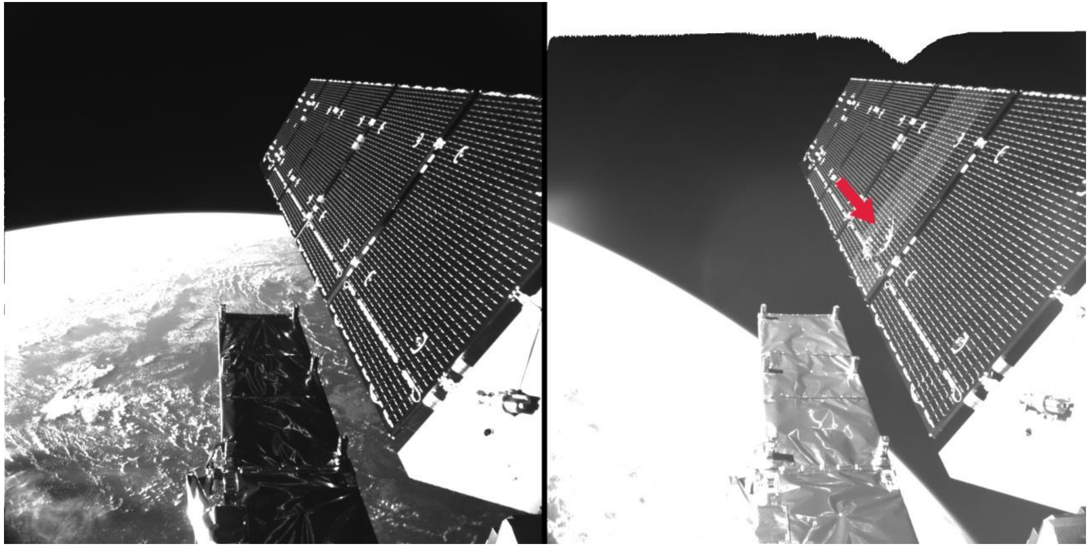
- 29 000 parts > 10cm
- 670 000 parts > 1 cm
- >170 000 000 > 1mm
- Large concentration between 700 & 1000 km

INTRODUCTION / EPS DESIGN DRIVERS

System drivers / Orbits



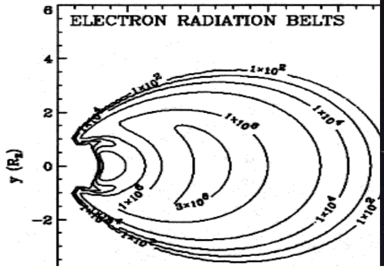
- **LOS (FRENCH RULE) TO AVOID GENERATION OF NEW DEBRIS**
  - Controlled desorbitation or
  - Parking in specific orbit with complete (propulsion and electronic) passivation (25 years in LEO, 100 years in GEO)

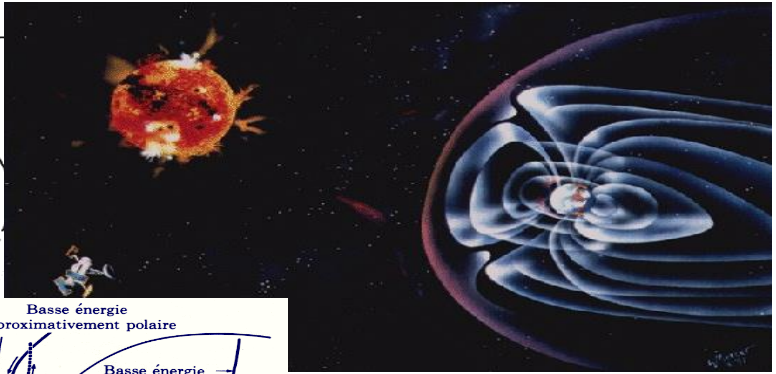


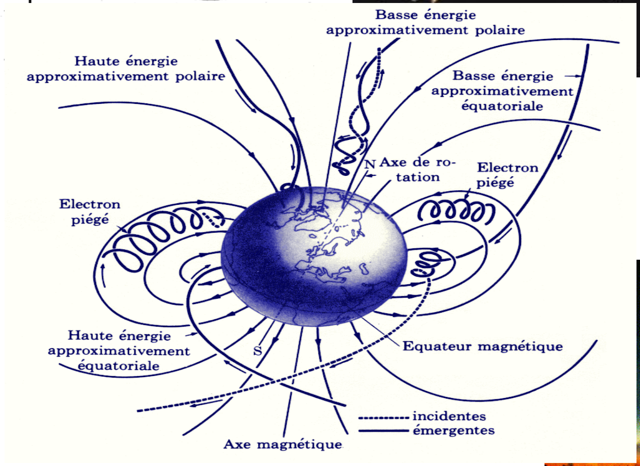
INTRODUCTION / EPS DESIGN DRIVERS

System drivers / Orbits

- **RADIATION SOURCES**
  - Trapped electrons
    Van Allen belts
  - Trapped protons
    Van Allen belts
  - Sun protons
    Sun eruptions
  - Space heavy ions
    Cosmic rays



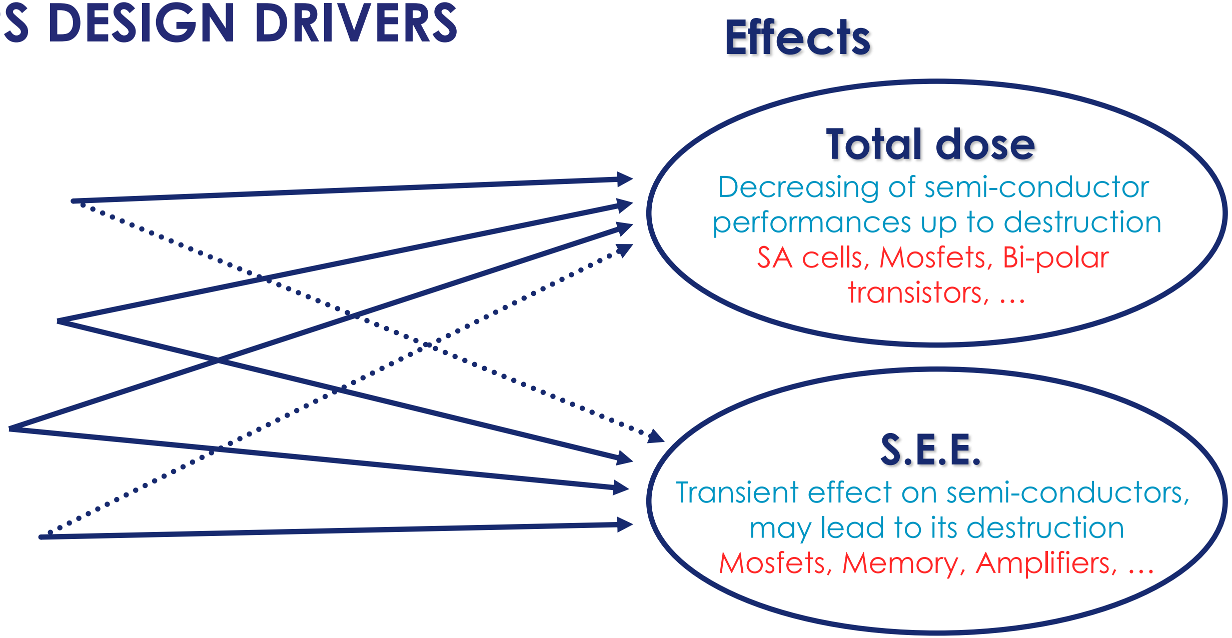




INTRODUCTION / EPS DESIGN DRIVERS

System drivers / Orbits

- **RADIATION SOURCES**
  - Trapped electrons
    Van Allen belts
  - Trapped protons
    Van Allen belts
  - Sun protons
    Sun eruptions
  - Space heavy ions
    Cosmic rays



-> The radiation environment has a direct impact on the definition & sizing of EPS

INTRODUCTION / EPS DESIGN DRIVERS

SYSTEM DRIVERS / MISSIONS

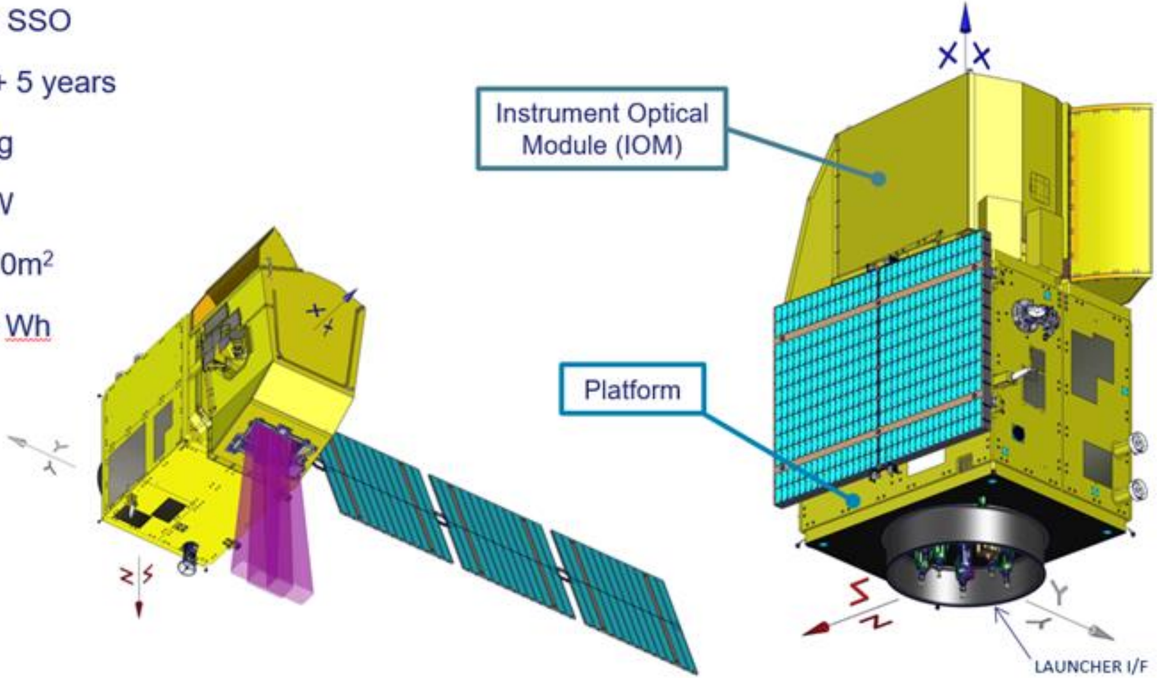
- (LIFE) DURATION
  - From few minutes (launchers) to 15 years (Geo)
  - Ageing drifts shall be assessed on each EPS constituent / Even some manufacturers may not be qualified for long term missions (e.g. ABSL batteries)
  - Impact on total radiation dose & nb of thermal cycles

- ENERGY BUDGET
  - Mission profiles
  - Regulated or Not regulated bus VS payload
  - Payload needs
    - TV broadcasting points a zone of the Earth
    - Science satellites may point any zone of the sky
    - Military satellites may point any zone of the earth and shall be very agile
  - Max and Mean power (in sunlight and in eclipse)
  - Orientation (attitude) of the satellite. The attitude constraints directly drive the sizing of the primary and secondary sources: impacts on
    - Eclipse duration
    - SA flux
    - Payload power available (in sunlight and in eclipse)
    - Definition of recovery / safety attitudes of the S/C
    - Thermal control
    - Bus quality
    - …

INTRODUCTION / EPS DESIGN DRIVERS

CHIME

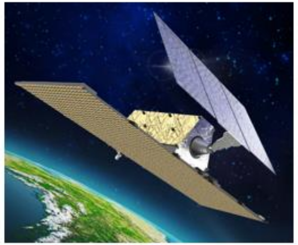
- /// Orbit: 632 km SSO
- /// Lifetime: 7.5 + 5 years
- /// Mass: 1800 kg
- /// Power: 2.2 kW
- /// Solar Array: $10 m^2$
- /// Battery: 4608 Wh

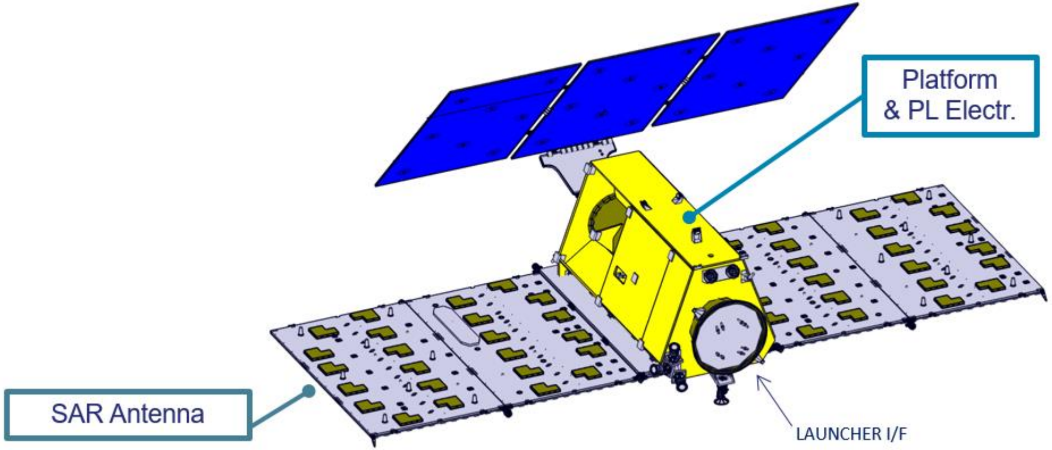


INTRODUCTION / EPS DESIGN DRIVERS

ROSE-L

/// Orbit:	693 km SSO
/// Lifetime:	7.5 + 5 years
/// Mass:	2130 kg
/// Power:	≈ 6.3 kW
/// Solar Array:	≈ 26 m^2
/// Battery:	10860 Wh

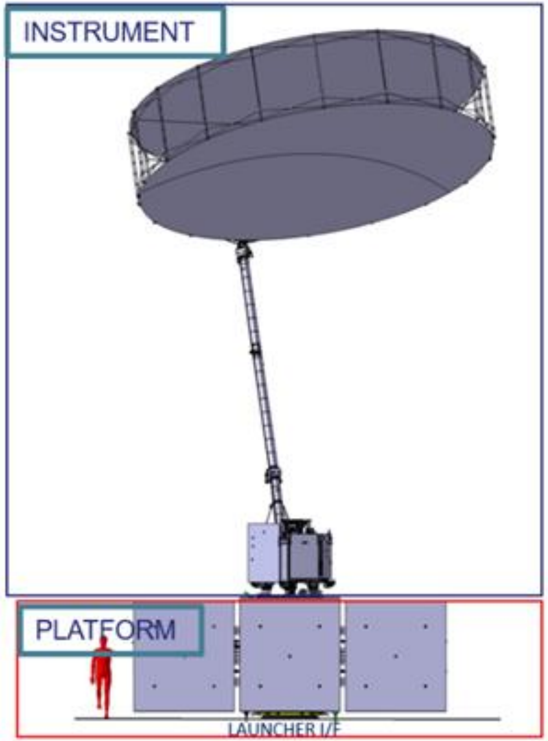




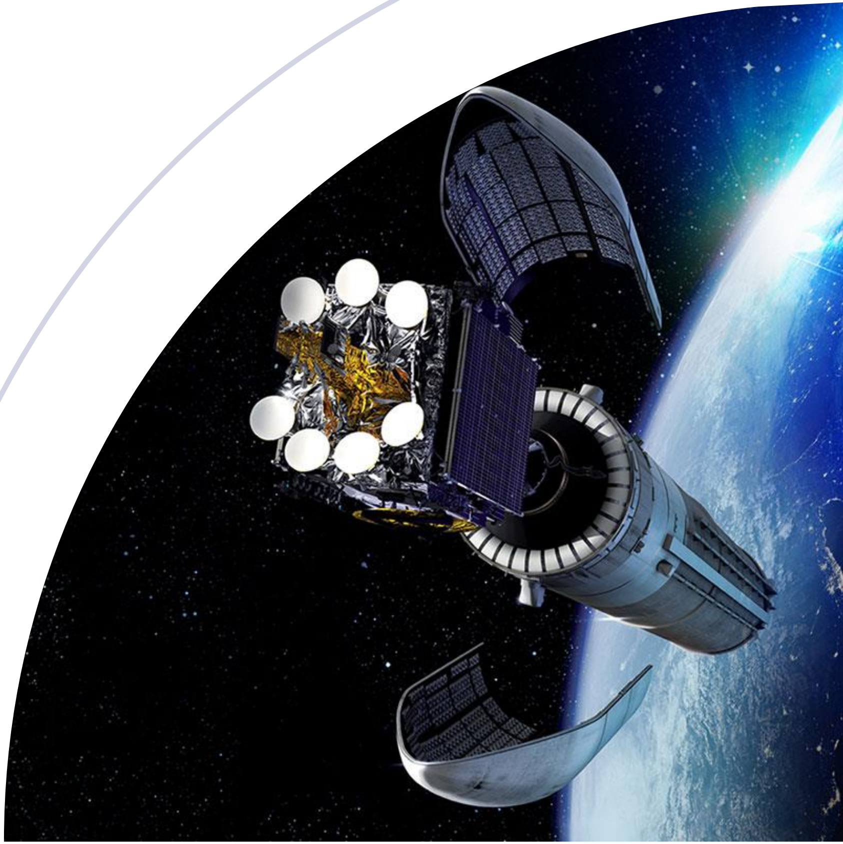
INTRODUCTION / EPS DESIGN DRIVERS

CIMR

- /// Orbit: 817 km SSO
- /// Lifetime: 7.5 + 5 years
- /// Mass: 1709 kg (dry)
- /// Power: ≈ 1.96 kW
- /// Solar Array: ≈ 14,57 m^2
- /// Battery: 3620 Wh

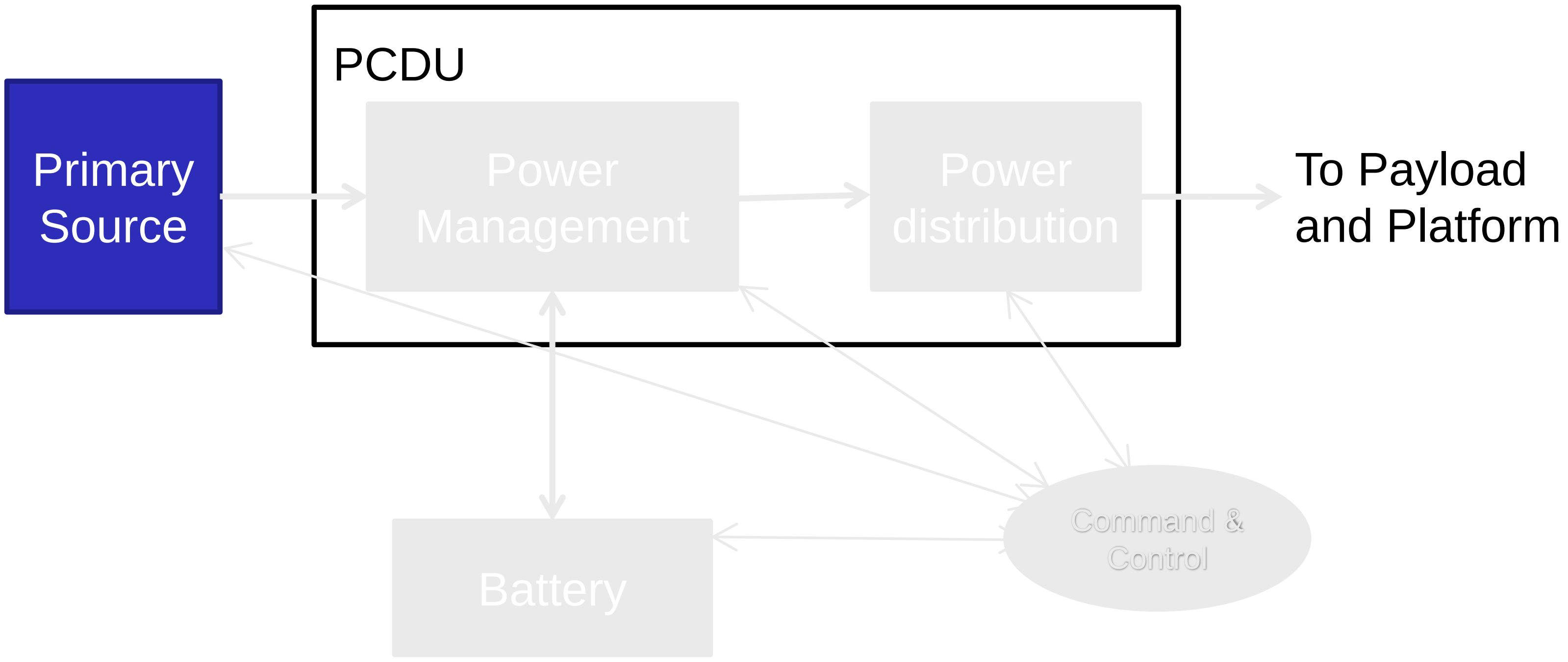


AGENDA



INTRODUCTION / EPS GENERAL INFORMATION

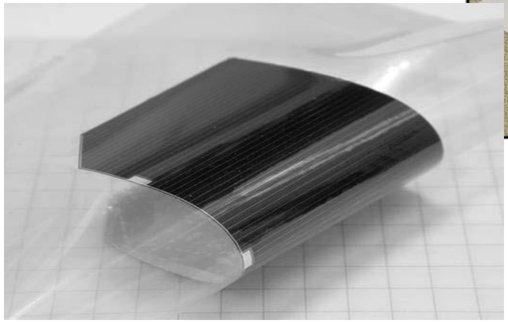
General functional block diagram



PRIMARY POWER SOURCES / SOLAR CELLS & ARRAYS

SA cells

- **A SOLAR CELL IS COMPOSED OF A SEMI-CONDUCTOR MATERIAL AND CONVERTS PHOTONS TO ELECTRONS**
- **PHOTOVOLTAIC EFFECT**
  - The solar flux is reflected, absorbed by the solar cell or crosses it
  - Every absorbed photon whose energy is greater than semi-conductor gap is going to release an electron and to create a positive « hole » (lack of electron). This electron is part of the crystalline network
  - Photons with excess energy dissipate it as heat in the cell, leading to reduced efficiency
  - An electrical field is introduced in the cell in order to separate this pair of opposite charges



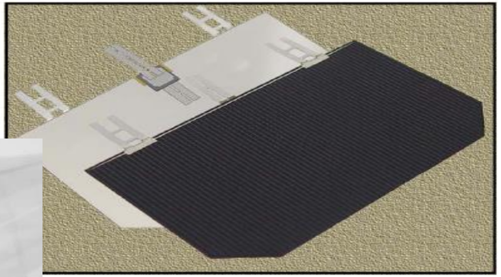
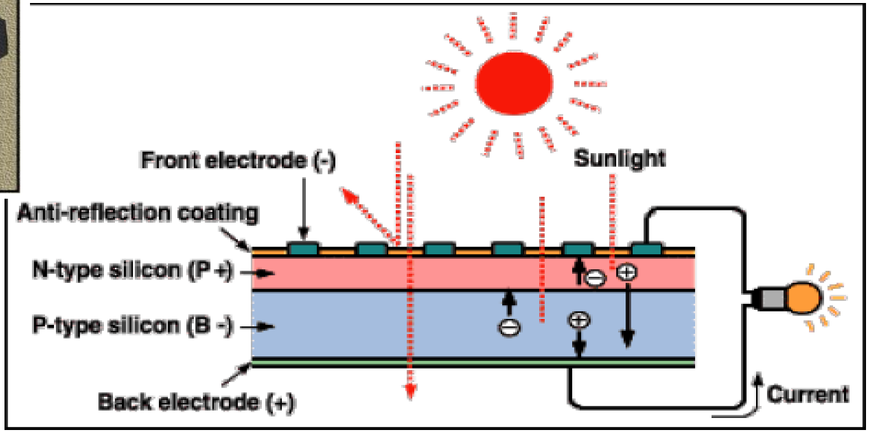
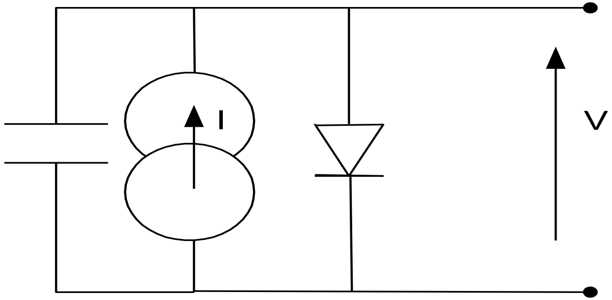


Illustration SPECTROLAB



Equivalent circuit diagram



- **EACH SOLAR CELL IS EQUIVALENT TO**
  - a current source in parallel with
  - a capacitor (variable) and
  - a diode

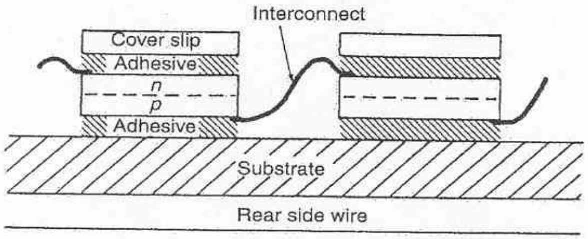


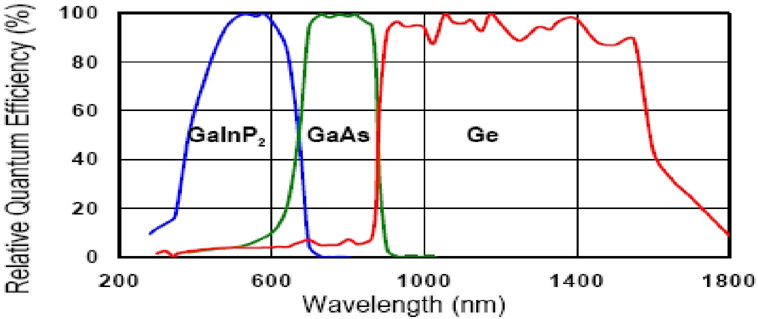
Figure 10.3 Schematic of a typical solar cell assembly

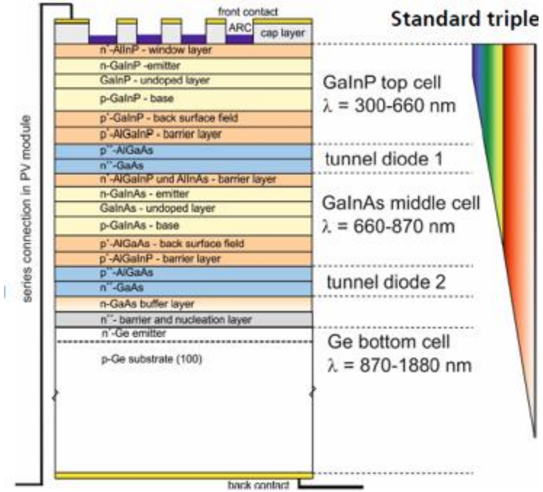
PRIMARY POWER SOURCES / SOLAR CELLS & ARRAYS

Solar arrays – performances

- **TYPICAL PERFORMANCES AFTER 15 YEARS IN GEO**
  - Silicium: 100 W / m2
  - High efficiency silicium: 130 W / m2
  - AsGa (mono junction): 170 W / m2
  - AsGa double junction: 200 W / m2
  - AsGa triple junction: 240 W / m2

- **POWER / KG:**
  - Silicium or AsGa/Ge: 40-50 W/kg
  - Multi junctions: 50-60 W/kg





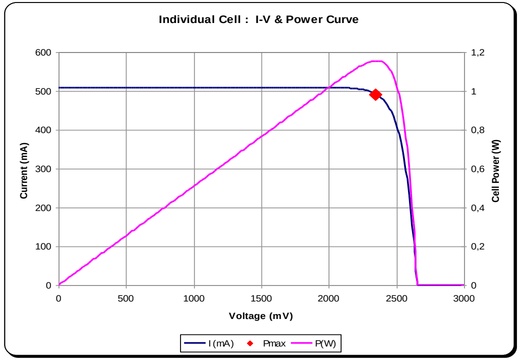
PRIMARY POWER SOURCES / SOLAR CELLS & ARRAYS

SA cell efficiency & characteristics

Electrical Data

		BOL	2,5E14	5E14	1E15
Average Open Circuit V_{oc}	[mV]	2700	2616	2564	2522
Average Short Circuit I_{sc}	[mA]	520.2	518.5	514.0	501.9
Voltage at max. Power V_{mp}	[mV]	2411	2345	2290	2246
Current at max. Power I_{mp}	[mA]	504.4	503.2	500.6	486.6
Average Efficiency η_{bare} (1367 W/m²)	[%]	29.5	28.6	27.8	26.5
Average Efficiency η_{bare} (1353 W/m²)	[%]	29.8	28.9	28.1	26.8

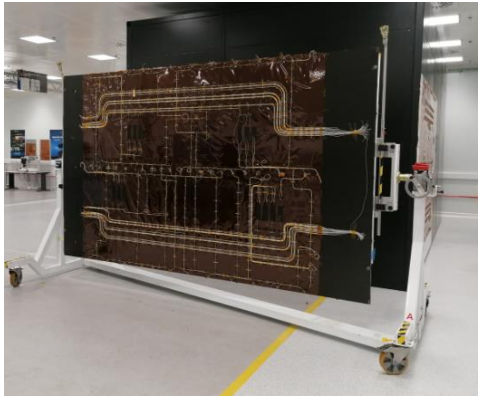
Standard: CASOLBA 2005 (05-20MV1, etc); Spectrum: AMO WRC = 1367 W/m²; T = 28 °C @fluence 1MeV [e/cm²]

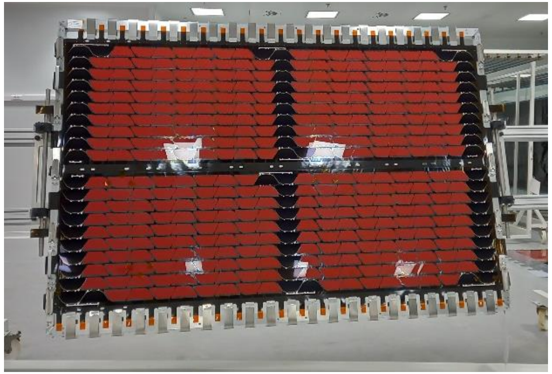


PRIMARY POWER SOURCES / SOLAR CELLS & ARRAYS

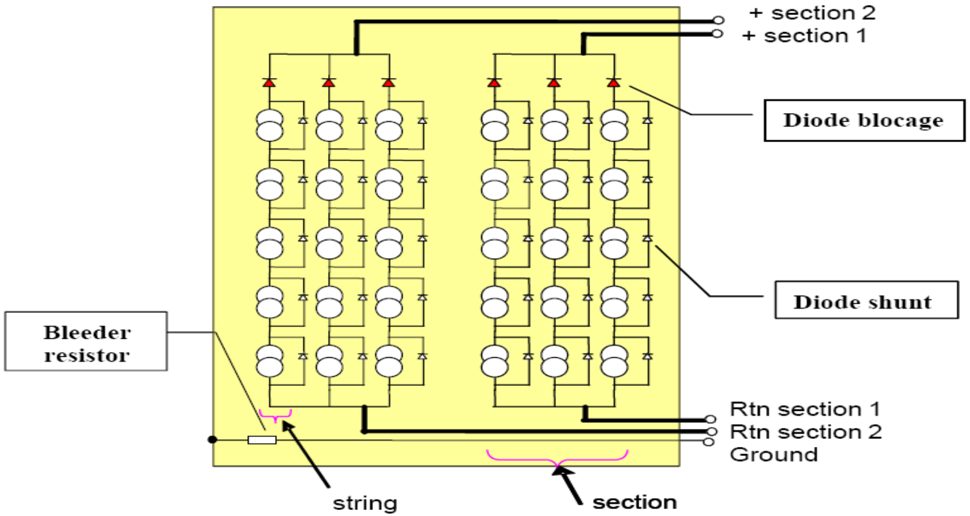
Solar arrays

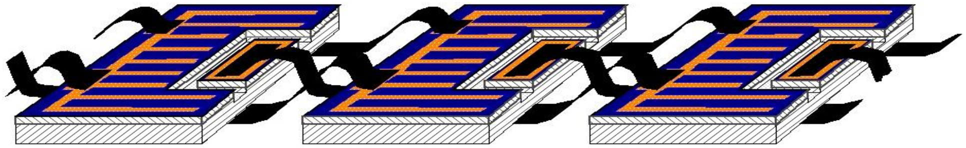
- A SOLAR CELL PRODUCES SOME HUNDREDS OF MILLIWATTS
- A SOLAR ARRAY (SA) IS COMPOSED OF THOUSANDS CELLS ASSEMBLED IN SERIES AND IN PARALLEL
  - The network = cells + interconnections + cabling + diodes
  - A string = assembling of cells in series to obtain the desired voltage
  - A section = strings in parallel to obtain the desired current
- SECTIONS ARE INDEPENDENT

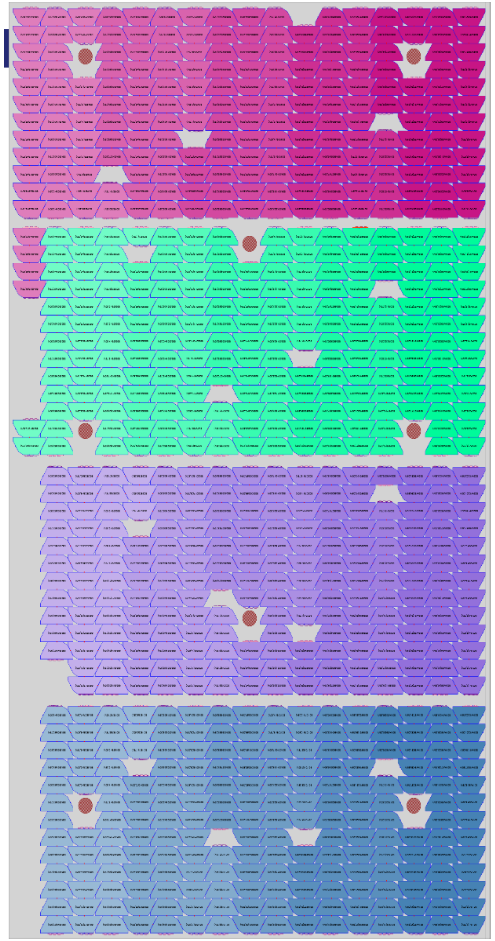




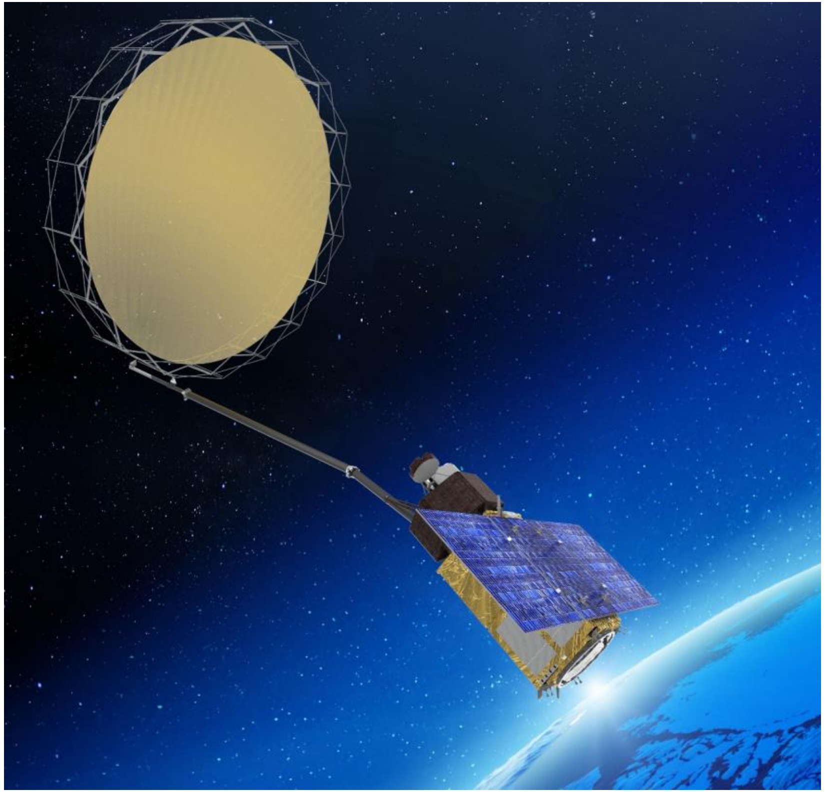
PRIMARY POWER SOURCES / SOLAR CELLS & ARI

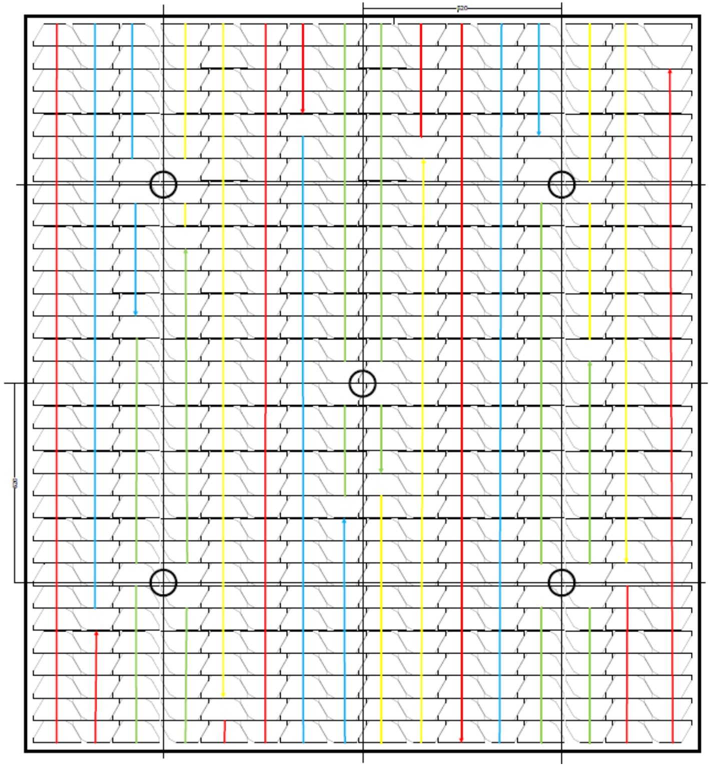






PRIMARY POWER SOURCES / SOLAR CELLS & ARRAYS





Copernicus CIMR Wing panel 37s 13p
2290 mm x 2100 mm dimension

PRIMARY POWER SOURCES / SOLAR CELLS & ARRAYS

Panel, glue, coverglass, …

- **SUBSTRATE**
  - Kapton with glass – or carbon- reinforcement

- **GLUE, ADHESIVE**
  - Fix SA cell on SA panel
  - Fix the coverglass on the cell
  - Ensure electrical & thermal conductivity

- **PANEL (HONEYCOMB)**
  - Support SA cells
  - Transfer heat to bottom side
  - Face high thermal gradient
  - Be compatible with deployment and orientation mechanisms

- **COVERGLASS**
  - Protect SA cell against ATOX
  - Protect SA cell against radiation
  - Limit the UV flux to the adhesive layer and to the cell by allowing suitable wavelength selection, via a good optical coupling (between free-space and glass & between glass and adhesive)

PRIMARY POWER SOURCES / SOLAR CELLS & ARRAYS

Efficiency degradation factors

- CELLS MISTMATCH & CALIBRATION
- MISSION LIFETIME
  - Loss of power: 1% to 2% every year (depends of the orbit)
- RADIATION EFFECTS

Radiation Degradation
(Fluence 1MeV Electrons/cm²)

Parameters	1x10¹⁴	5x10¹⁴	1x10¹⁵
Imp/Imp₀	0.99	0.98	0.96
Vmp/Vmp₀	0.94	0.91	0.89
Pmp/Pmp₀	0.93	0.89	0.86

- UV
- METEORITE IMPACT
- ATOX DENSITY
  - Aggressive and corrosive environment (tied to the LEO) on cover glass protection and on exposed interconnection (oxidation of silver and then increase of resistivity)

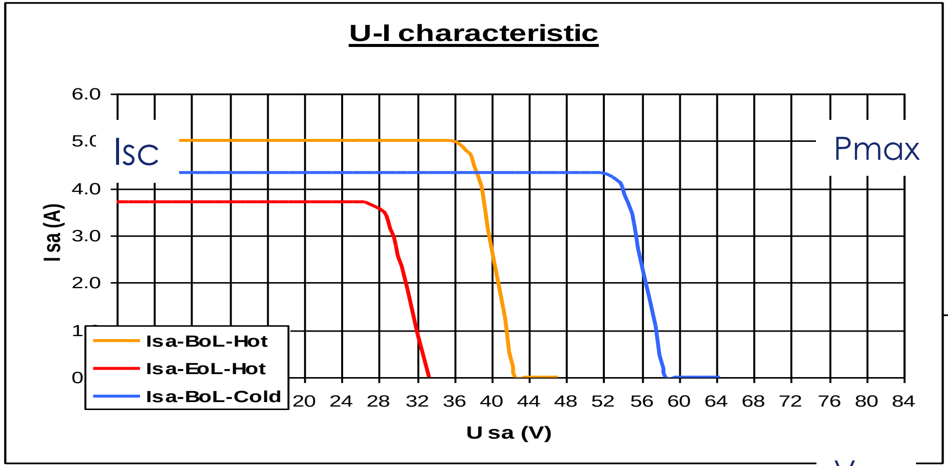
	BOL	EOL SS	EOL WS	Isc Max	Voc Max
Duration	0,0	12,5	12,5	0,0	0,0
Cell Mismatch	0,990	0,990	0,990	1,010	1,010
Cell Calibration	0,970	0,970	0,970	1,000	1,000
RSS	0,968	0,968	0,968	1,010	1,010
CVG Loss	0,982	0,982	0,982	0,995	0,995
UV + µM	1,000	0,969	0,969	1,000	1,000
ATOX	1,000	1,000	1,000	1,000	1,000
Dataset Uncert.	1,000	1,000	1,000	1,020	1,000
Pointing Error	0,9998	0,9998	0,9998	0,9998	0,9998
Life Loss	0,982	0,951	0,951	1,015	0,995
String //	13	13	13	6	1
Cell serie	37	37	37	37	37
V bus	70,0	70,0	70,0	0,0	0,0
Delta V	2,8	2,8	2,8	2,8	2,8
V fluence (EOL)	0,00E+00	2,11E+14	2,11E+14	0,00E+00	0,00E+00
I fluence (EOL)	0,00E+00	9,89E+13	9,89E+13	0,00E+00	0,00E+00
Solar flux	1323	1323	1413	1413	1413
Declinaison	0,00	32,00	0,00	0,00	0,00

Temp NOP	99,4°C	95,3°C	99,3°C	122,0°C	NA
Temp OP	80,0°C	80,0°C	80,0°C	121,3°C	-130,0°C
Isc	17,4	14,2	17,9	9,23	
Iop	16,80	13,07	16,70	0	0
Voc	80,6	76,4	76,3	76,1	129,34
Vmp	73,2	68,9	69,6		
Imp	16,31	13,33	16,80		
Power @ Vmp	1194	918	1170		
Power @ Vbus	1176	915	1169	0	0
Power @ Vbus 1Str Failed	1085	844	1079		

Datas of Copernicus CIMR Wing panel 37s 13p

PRIMARY POWER SOURCES / SOLAR CELLS & ARRAYS

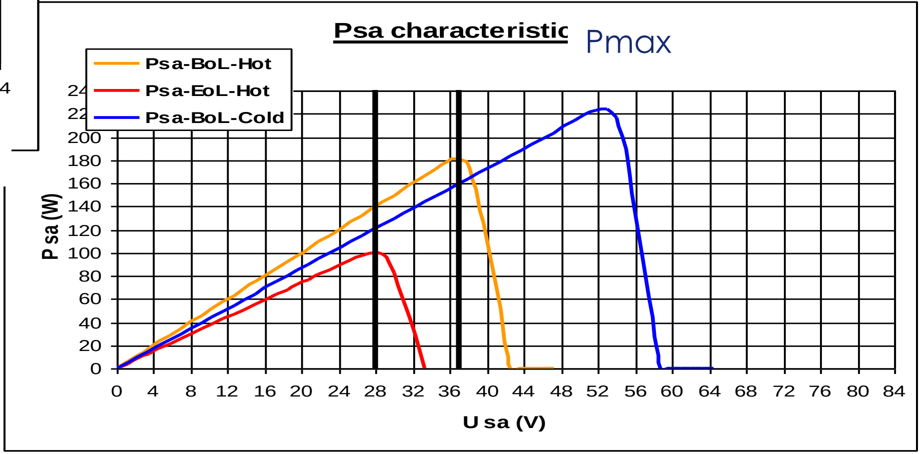
Optimal working point – at max. available power



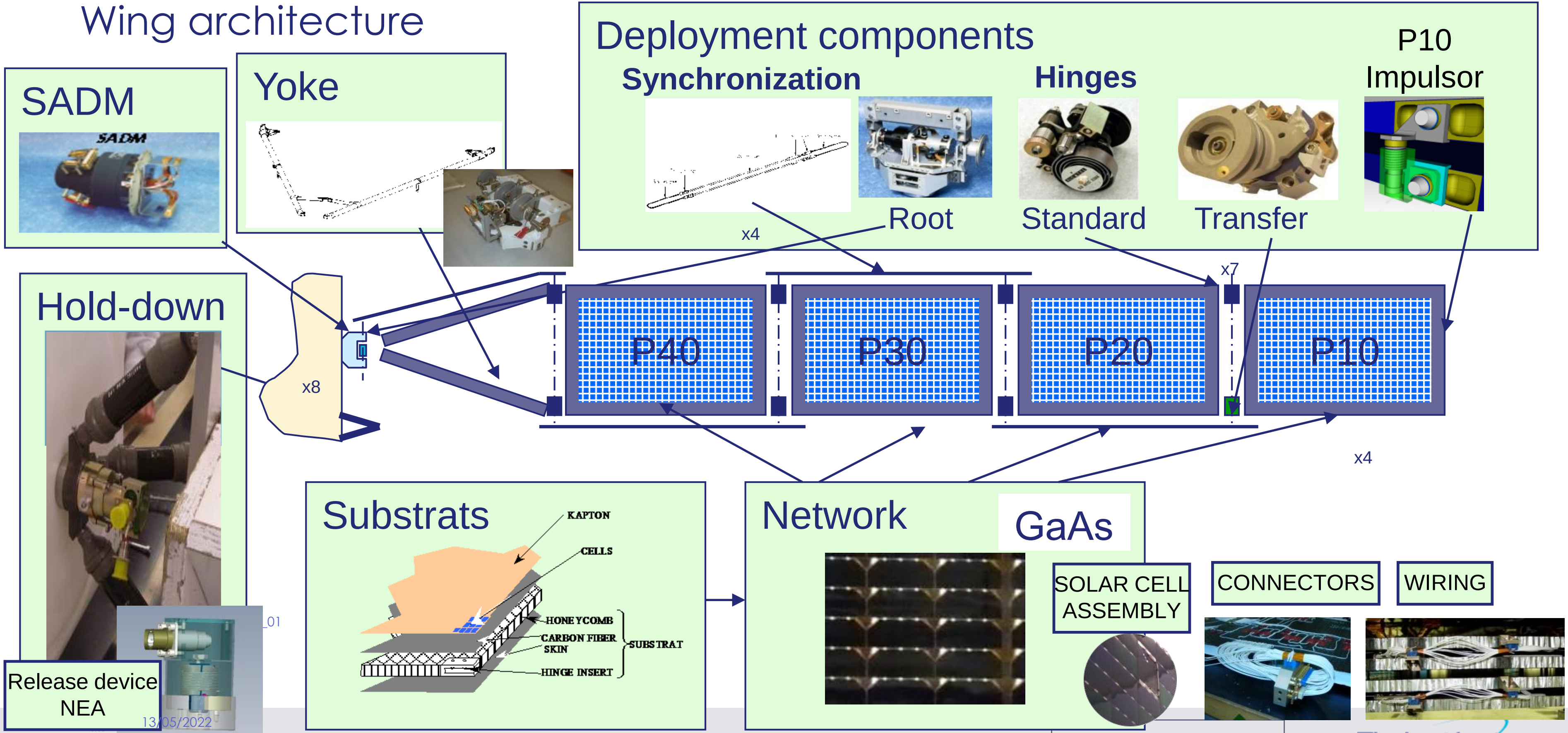
Pmax is largely depending of temperature & ageing

Temperature is linked to

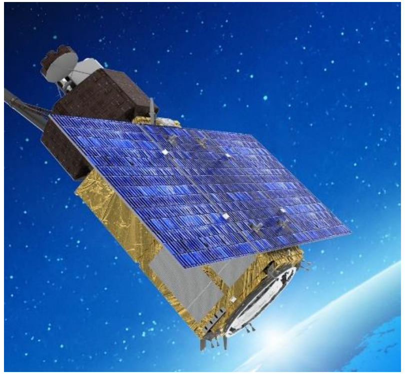
- The incoming flux
  - Direct solar flux
  - Albedo
  - IR flux of the earth
- The outcoming flux
  - Flux reflected by the cells
  - Power delivered to the satellite
  - IR flux of the front and rear part of the SA



PRIMARY POWER SOURCES / SOLAR CELLS & ARRAYS



PRIMARY POWER SOURCES / SOLAR CELLS & ARRAYS



CIMR

Solar arrays – Types

- FIXED
  - Solar cells are glued on the structure of the satellite
  - The power is limited by the surface of the satellite
- DEPLOYABLE (FIXED)
  - Solar cells are glued on flaps (folded at launch and deployed in orbit)
  - Difficult to manage the attitude constraints
- DEPLOYABLE AND MOBILE
  - 1-degree of freedom

LADEE (Lunar Atmosphere and Dust Environment Explorer)

Space Inspire

PRIMARY POWER SOURCES / FUEL CELLS

Electromechanical devices performing a controlled chemical reaction (oxidation) to derive electrical energy (rather than heat energy)

- **ADVANTAGES**
  - Minimal thermal changes
  - Compact and flexible solution
  - Production of water (manned mission)

- **DRAWBACKS**
  - Need of fuels: hydrogen & oxygen yielding water as the reaction product

- **USED FOR SHUTTLE ORBITER, LUNAR ROVER, …**

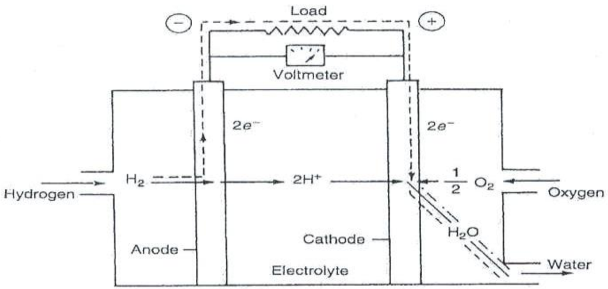


Figure 10.10 Schematic of a hydrogen/oxygen fuel cell. At the anode–electrolyte interface, hydrogen dissociates into hydrogen ions and electrons. The hydrogen ions migrate through the electrolyte to the cathode interface where they combine with the electrons that have traversed the load [2] (From Angrist, S. W. (1982) *Direct Energy Conversion*, 4th edn, Copyright Allyn and Bacon, New York)

PRIMARY POWER SOURCES / FUEL CELLS

Typical current-voltage curve for a hydrogen/oxygen fuel cell

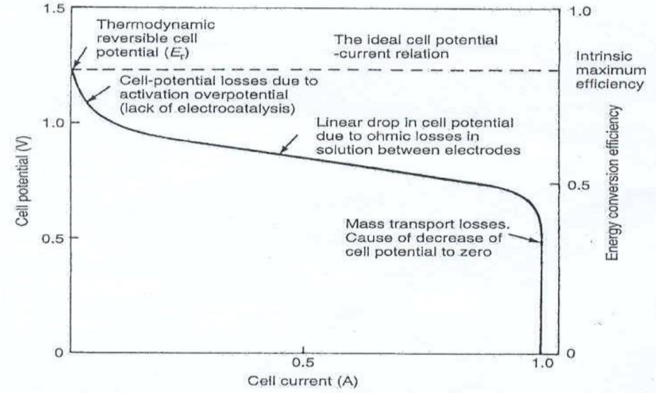


Figure 10.11 Typical cell potential and efficiency–current relation of an electrochemical electricity producer showing regions of major influence of various types of overpotential losses (Source [10])

Performance summary of fuel

System	Specific power (W/kg)	Operation	Comment
Gemini	33	240 h	Not drinking water
Apollo	25		Operated at 505 K 24 h start-up / 17 h shutdown
Shuttle	275	2500 h	15 min start-up / instantaneous shutdown
SPE technology	110 – 146	> 40000 h	
Alkaline technology	367	> 3000 h	
Alkaline technology	110	> 40 000 h	
Goal (lightweight cell)	550		

PRIMARY POWER SOURCES / FUEL CELLS

Use of fuel cell as « secondary power source »

- **REGENERATIVE FUEL CELLS (100 KW SYSTEM POWER) ELECTROLYZE OF WATER IS PERFORMED DURING THE 'CHARGE' CYCLE THANKS TO PRIMARY SOURCE POWER**

- **ADVANTAGE**
  - Lower SA power need thanks to judicious sizing of the fuel

- **DRAWBACK**
  - Lower efficiency (50 – 60 %) than battery

- **INTERESTING FOR LEO OPERATIONS WHERE ATMOSPHERIC DRAG IS IMPORTANT (VERY LOW ORBITS) -> REDUCTION OF PROPELLANT USED FOR ORBIT CONTROL**

PRIMARY POWER SOURCES / RTG

Deep-space missions (further than Mars) or Military use

- **LONG TIME MISSIONS, NOT-COMPATIBLE WITH FUEL CELLS**
- **FAR FROM SUN, NOT-COMPATIBLE WITH SA**
  - Decrease of SA flux partially compensated by increased of cell efficiency due to decrease of temperature (rE/rSC)1.5

-> Use of radioactive decay process, use of thermoelectric effect

Thermoelectric effect

- **GENERATION OF A VOLTAGE BETWEEN (SEMI-CONDUCTOR) MATERIALS MAINTAINING A TEMPERATURE DIFFERENCE. POWER FUNCTION OF:**
  - Absolute t° of hot junction
  - T° difference between materials
  - Properties of materials
- **LOW EFFICIENCY (< 10 %)**
  -> REMOVING WASTE HEAT MAY BE AN ISSUE
- **HEAT SOURCE: SPONTANEOUS DECAY OF A RADIOACTIVE MATERIAL, EMITTING HIGH-ENERGY PARTICLES, HEATING ABSORBING MATERIALS**

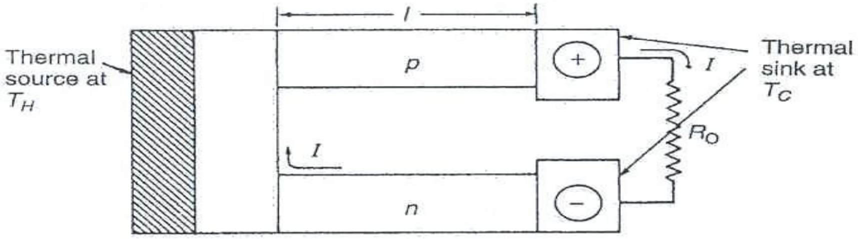


Figure 10.12 Schematic diagram of a semiconductor radioisotope generator (From Angrist, S. W. (1982) *Direct energy conversion*, 4th edn, Copyright Allyn and Bacon, New York)

PRIMARY POWER SOURCES / RTG

Advantages

- POWER PRODUCTION INDEPENDENT OF S/C ORIENTATION & SHADOWING
- INDEPENDENCE OF DISTANCE FROM SUN
- LOW POWER LEVEL MAY BE PROVIDED FOR LONG TIME PERIOD
- NOT SUSCEPTIBLE TO RADIATION DAMAGE
- COMPATIBLE WITH LONG ECLIPSE (E.G. LUNAR LANDERS)

Drawbacks

- Affect the radiation environment of S/C (deployment away from the main satellite bus)
- Radioactive source induce safety precautions in AIT
- High t° operation required -> impact thermal environment of S/C
- Interfere with plasma diagnostic equipment (scientific missions)
- Environmental risk in case of launch failure or S/C crash

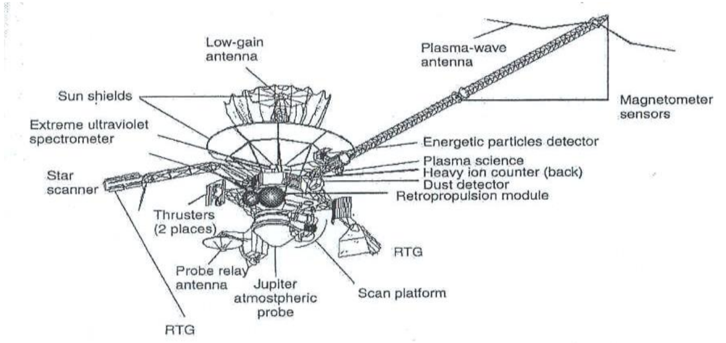


Figure 10.13 The Galileo spacecraft configuration, showing the position of the RTG sources (Courtesy of NASA/JPL/Caltech)

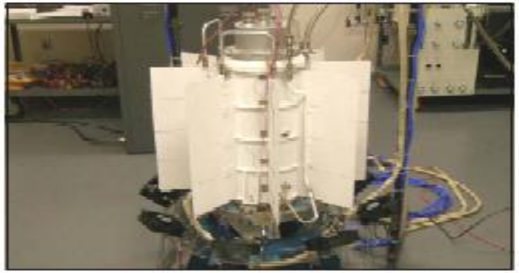
PRIMARY POWER SOURCES / RTG & OTHERS

Example of RTG

CASSINI (SATURN MISSION)	628 W	195 W/KG
GALILEO PROBE/ULYSSES	285 W	195 W/KG
NIMBUS/VIKING/PIONNER	35 W	457 W/KG
APPOLO LANDER	25 W	490 W/KG
MARS SCIENCE LABORATORY	120 W	416 W/KG

Nuclear fission

- FISSIBLE MATERIAL (E.G. URANIUM-235) USE OF NUCLEAR FISSION PROCESS (AS FOR TERRESTRIAL NUCLEAR POWER PLANT
- USED TO DRIVE THERMOELECTRIC CONVERTE



MMRTG Engineering Unit

PRIMARY POWER SOURCES / OTHERS

Solar heat

- USE OF SUN ENERGY TO DRIVE A HEAT ENGINE AND THEN A ROTARY CONVERTER TO ELECTRICITY OR A THERMOELECTRIC CONVERTER
- CONCEPT INTERESTING FOR SPACE STATION
  - Reduced drag (reducing area of SA panels)
  - Reduced maintenance effort

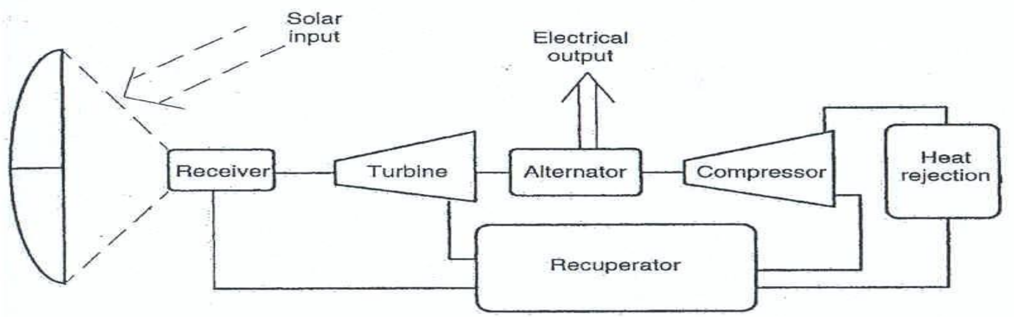


Figure 10.14 Solar dynamic Brayton cycle

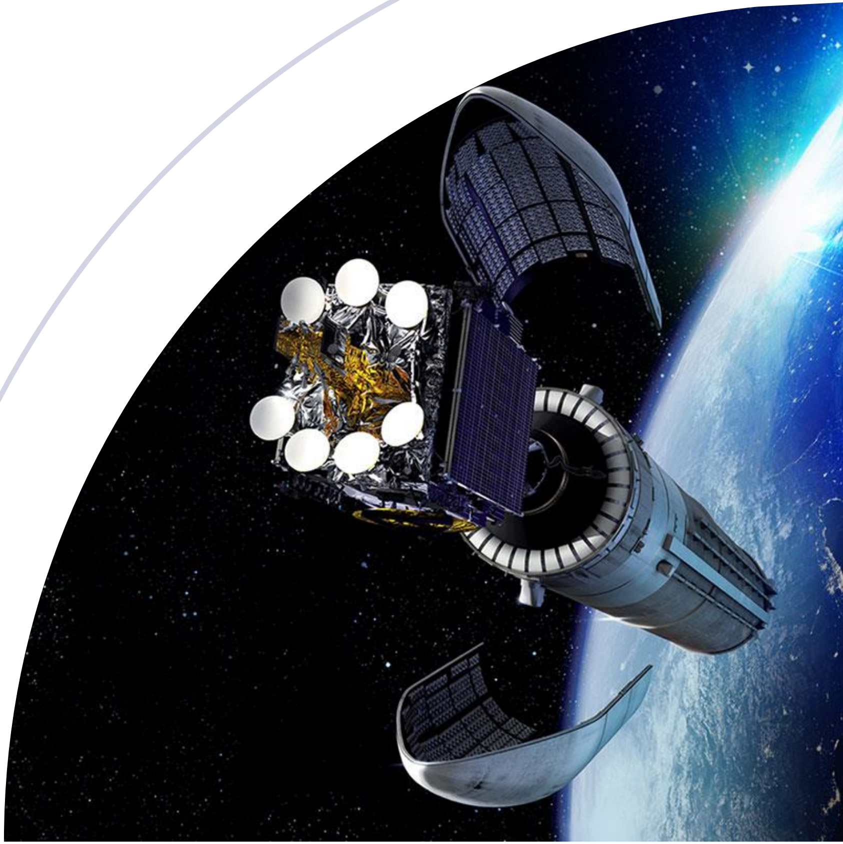
PRIMARY POWER SOURCES / OTHERS

Other sources

- Fuel Cell
  - Regenerative fuel cells (100 kW system power) electrolyze of water is performed during the ‘charge’ cycle thanks to primary source power
  - Interesting for very large mission, no application today

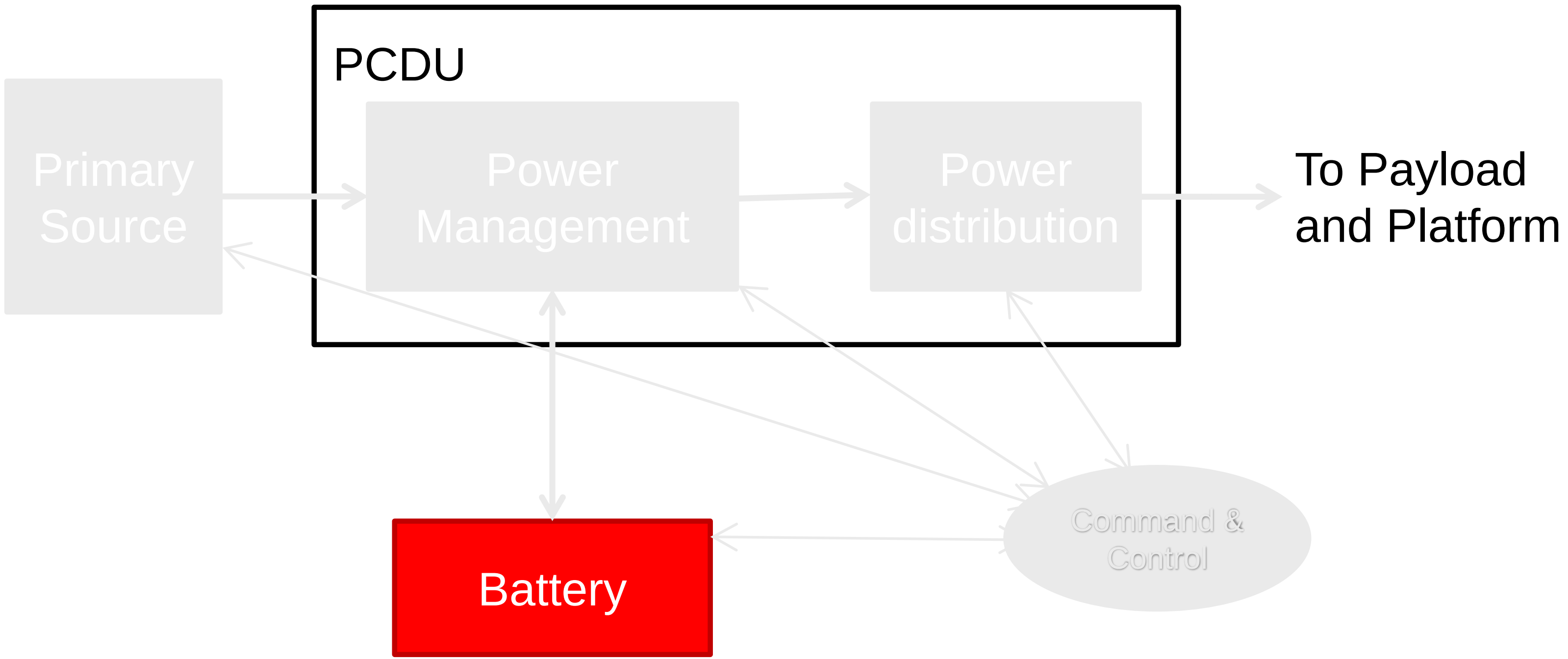
- RTG. RadioThermal Generator => e.g. Voyager 1 & 2
  - Generation of a voltage between (semi-conductor) materials maintaining a temperature difference.
  - Low efficiency (< 10 %)
  - Interesting for deep space missions (beyond Jupiter)

AGENDA



INTRODUCTION / EPS GENERAL INFORMATION

General functional block diagram



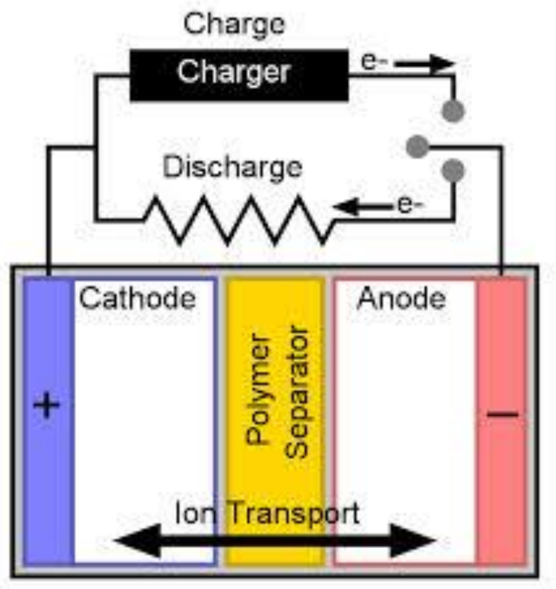
SECONDARY POWER SOURCES

Accumulators

- ELECTROMECHANICAL DEVICES PERFORMING A CONTROLLED CHEMICAL REACTION TO DERIVE ELECTRICAL ENERGY

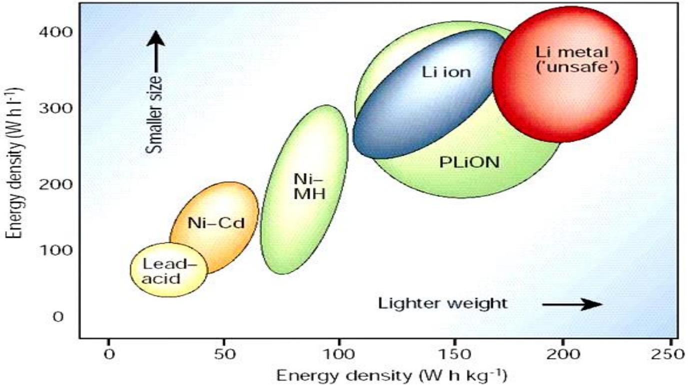
- DURING DISCHARGE, THE POSITIVE ACTIVE MATERIAL IS REDUCED, ABSORBING ELECTRONS, AND THE NEGATIVE MATERIAL IS OXIDIZED, RELEASING ELECTRONS. IONS ARE DISSOLVED INTO AN ELECTROLYTE AND TRANSFERRED THROUGH A SEPARATOR (WHICH IS AN ELECTRIC INSULATOR) TO EQUILIBRATE THE CHARGE.

- IF THE ELECTRODE MATERIALS ARE CHOSEN SO THAT THESE REACTIONS ARE REVERSIBLE, THE CELL CAN BE RECHARGED. IT IS CALLED SECONDARY (I.E. RECHARGEABLE).



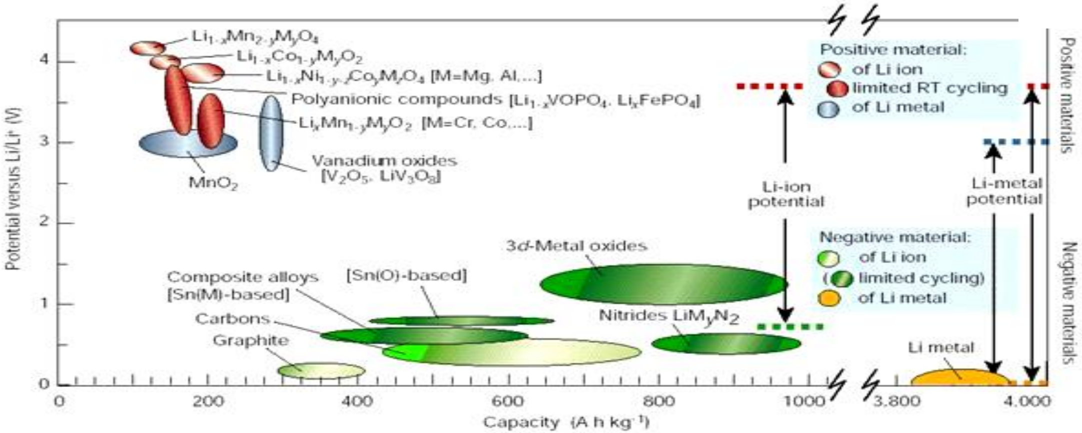
SECONDARY POWER SOURCES

Accumulators



CRITICAL PARAMETERS

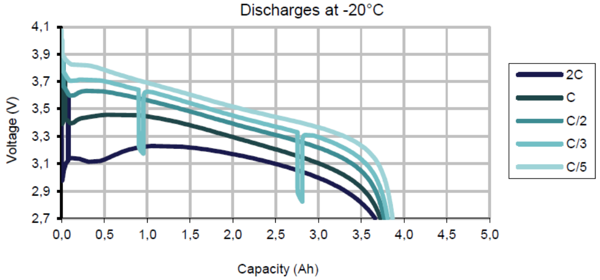
- Charge/discharge rate
- Depth of Discharge
- Extent of over-discharging
- Thermal sensitivity to each of these parameters

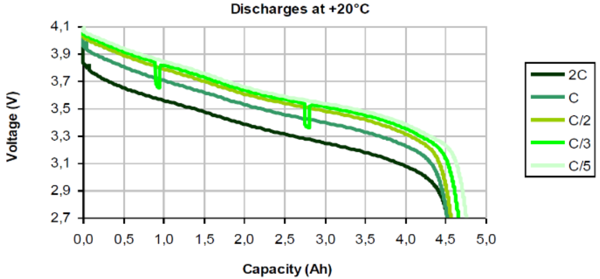


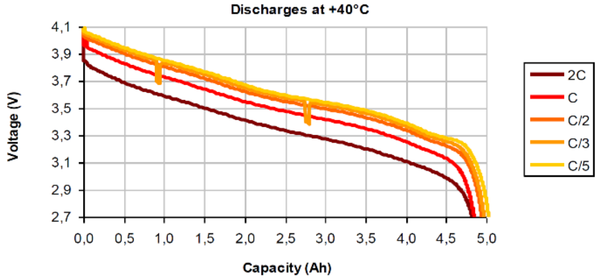
SECONDARY POWER SOURCES

Typical Characteristics

- Capacity : a battery's capacity is the amount of electric charge it can store. Capacity is given in A.h (1 A.h = 3600 Coulomb).
  - 1.5Ah -> 100Ah
- C rate: the C-rate signifies a charge or discharge rate relative to the capacity of a battery in one hour.
- Cell Open Circuit Voltage : difference between cell electrode potentials
- Voltage range
  - 4.1V -> 3.3V (or 3V or 2.7V)
- Series Resistance
  - 1mΩ -> 10m Ω
- Leakage current
  - 0mA -> 5mA







SECONDARY POWER SOURCES

Battery tapering & energy sizing

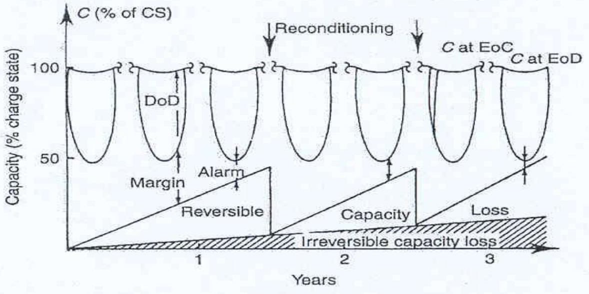
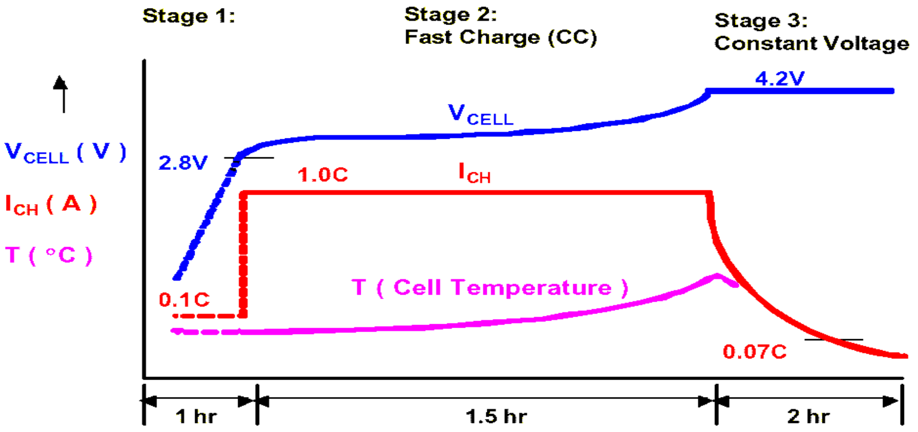
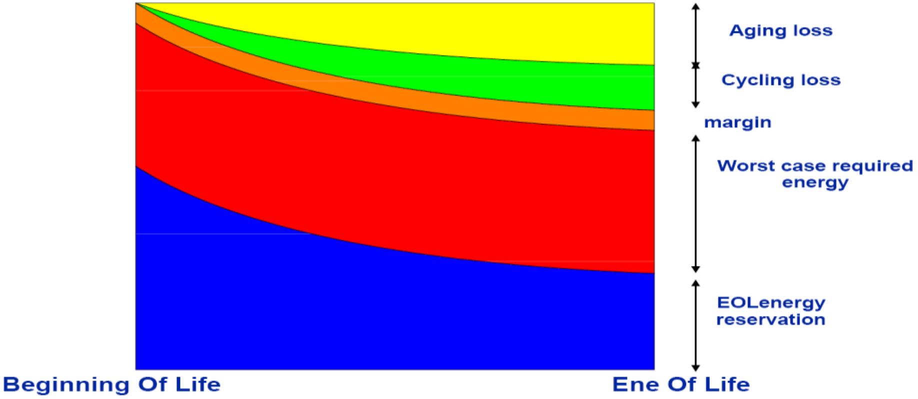


Figure 10.16 Battery reconditioning via complete discharge to improve battery capacity. Both reversible and irreversible capacity loss occurs [17] (Reproduced by permission of European Space Agency and P. Montalenti)





SECONDARY POWER SOURCES

Battery

- **ROLE: SUPPORT THE SOLAR ARRAY DURING**
  - LEOP phases
  - Eclipses
  - Loss of sun pointing
  - Peak power demands
  - …
- **SERIES / PARALLEL ASSEMBLING OF ACCUMULATOR CELLS**
  - In series to reach the desired voltage
    - 22-37 V in LEO
    - Galileo FoC: 42.5 V
    - SPACEBUS 4000/NEO: 100 V
  - In parallel to reach the desired capacity
- **BALANCING**
  - Mandatory in GEO
    - deep discharges (up to 80%)
  - Trade OFF in LEO:
    - Thousands of cycles
    - smaller discharges

Illustration SAFT

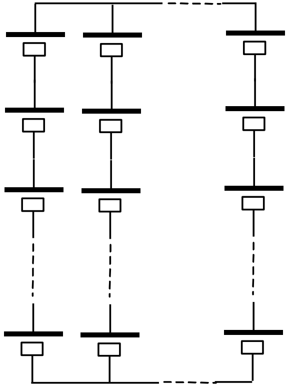


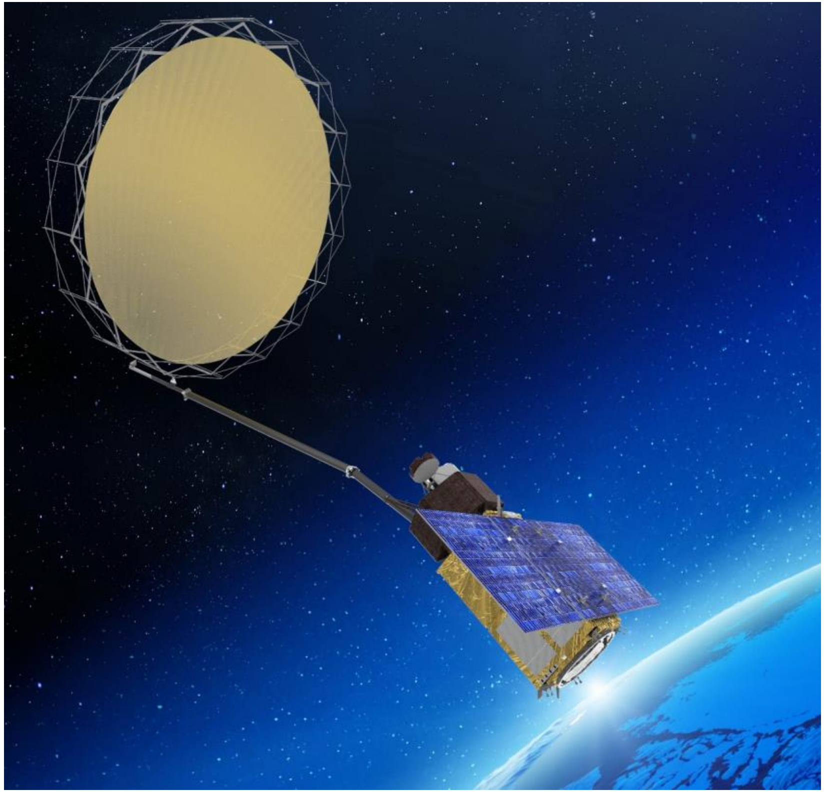
Illustration TAS

SECONDARY POWER SOURCES

BoL	SAFT NiCd VOS 40	SAFT NiH2 93 AN	SAFT LiIon VOS140	SAFT LiIon MP76065	SONY LiIOn 18650HC
Capacity	46 Ah	89 Ah	38.6 Ah	6.1 Ah	1.4 Ah
Mean voltage	1.2 V	1.36 V	3.6 V	3.6 V	3.7 V
Energy	55 Wh	120 Wh	140 Wh	22 Wh	5.2 Wh
Mass	1610 g	2108 g	1107 g	155 g	41.2 g
Energy/kg	34 Wh/kg	57 Wh/kg	126 Wh/kg	141 Wh/kg	126 Wh/kg
Efficiency	70 %	70 %	90 %	90 %	90 %

Data CNES

SECONDARY POWER SOURCES



Copernicus CIMR Battery Configuration (8S5P)2S

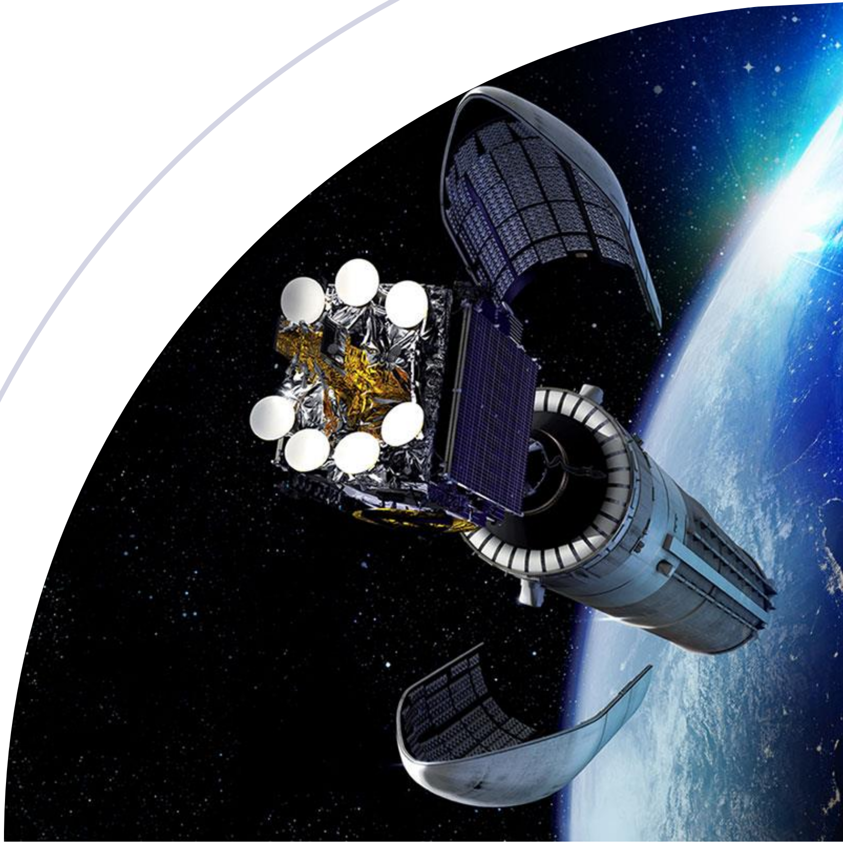
8S5P Module

Nameplate characteristics module	
End of Charge Voltage	33.6 V
Capacity nameplate	61.5 Ah
Energy nameplate	1810 Wh

CIMR Battery

Nameplate characteristics CIMR	
End of Charge Voltage	67.2 V
Capacity nameplate	61.5 Ah
Energy nameplate	3620 Wh

AGENDA



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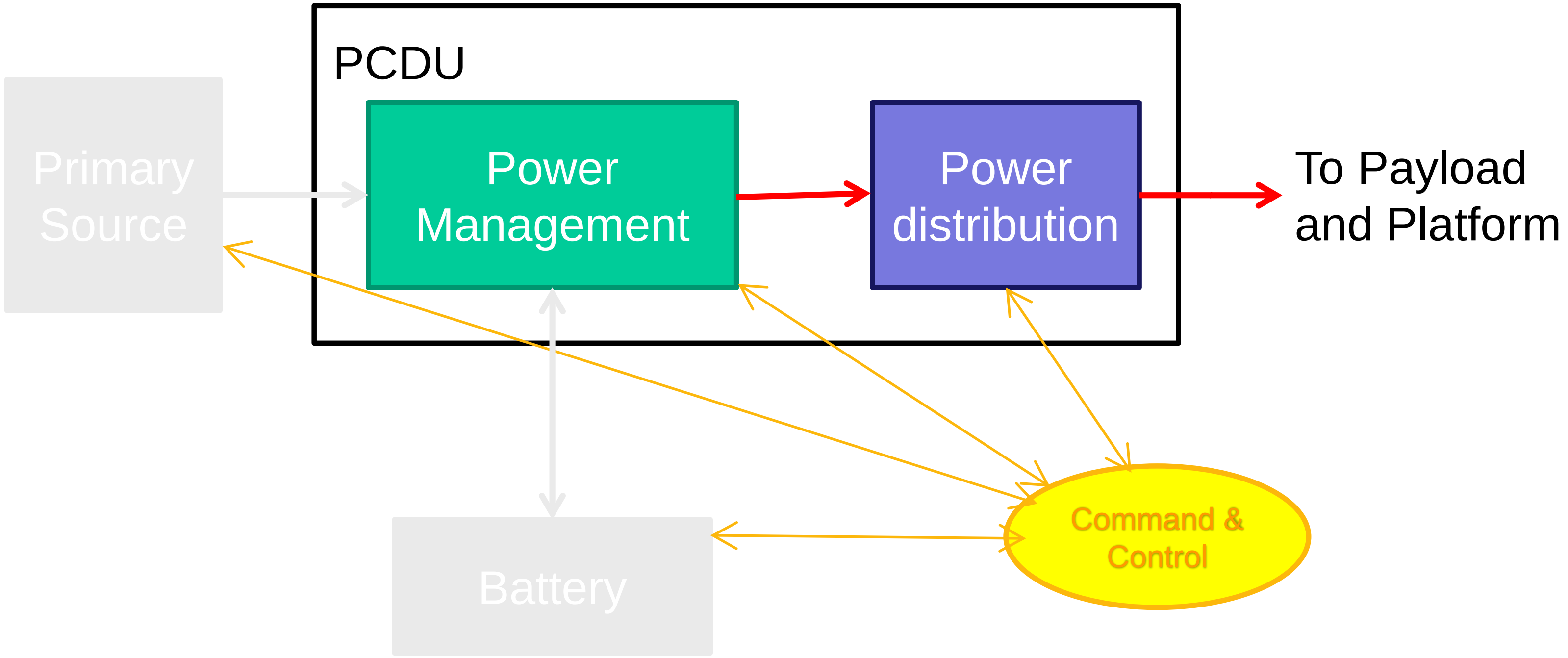
General functional block diagram

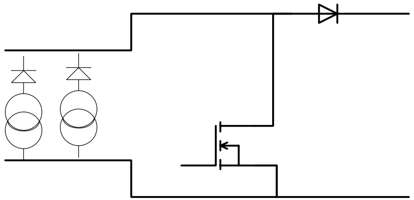


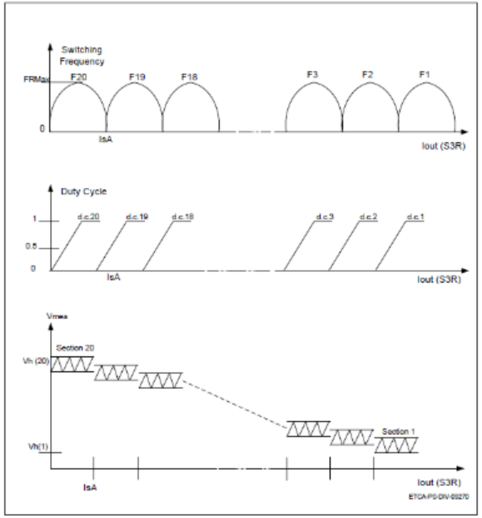
POWER MANAGEMENT, CONTROL & DISTRIBUTION / ARCHITECTURE

Conditioning topology

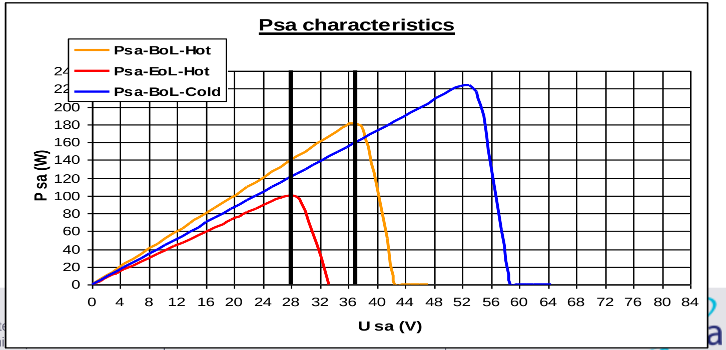
= HOW SOLAR ARRAY POWER IS USED TO BE DELIVERED TO THE DIFFERENT USERS / CHARGE THE BATTERY

- Sequential Switching Series Regulator (S3R)
  - S3R operates at the bus voltage and extracts the available power
    from the solar array for this precise voltage (aka DET Direct Energy Transfer)
    - Simplest solution
    - One section in switching
    - Fsw < 10kHz
    - Reliable or Non reliable





- Maximum Power Point Tracking (MPPT)
  - MPPT can operate in a wide range of voltages to track the maximum available power from the solar array, converts the (VMP, IMP) into (Vbus, Ibus)
    - More complex and dissipative solution
    - Drawbacks: Efficiency & Mass
    - Advantages: Works for different SA characteristics / MPP achieved
    - Main interest: Interplanetary missions / Non-pointing Solar Arrays (vibrations constrains or PF complexity)



POWER MANAGEMENT, CONTROL & DISTRIBUTION / ARCHITECTURE

Bus voltage 1/3

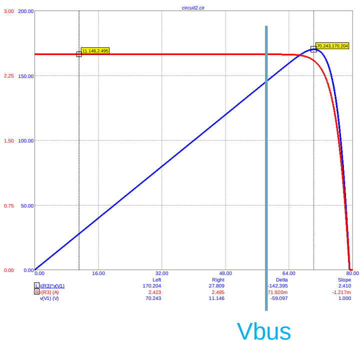
- ARCHITECTURE
  - Regulated
    - Voltage variation is limited to about 0,5% whatever the satellite modes
  - Unregulated
    - Bus voltage is imposed by the battery voltage
- MANY STANDARDS
  - Regulated →28V, 50 V,100 V
  - Unregulated→ [22-33V]; [42-52]; [40-67V] ….
- CHOICE IS BASED ON
  - Bus power
    - Recommended ESA rule: $P < U^2/0.5$ for bus impedance reasons
    - High bus voltage means
      - Less current and harness simplification
      - « High » voltage management at equipment level (SA, battery, PCDU, …)
  - Payload flight heritage
  - User's need (mission)
    - Scientific payloads may require regulated bus to fulfill their precisions
    - Thermal stability of some specific loads may requires regulated bus (thermal management is easier in that architecture)

-> SOME ARCHITECTURE MAY EVEN REQUIRES TWO BUSES

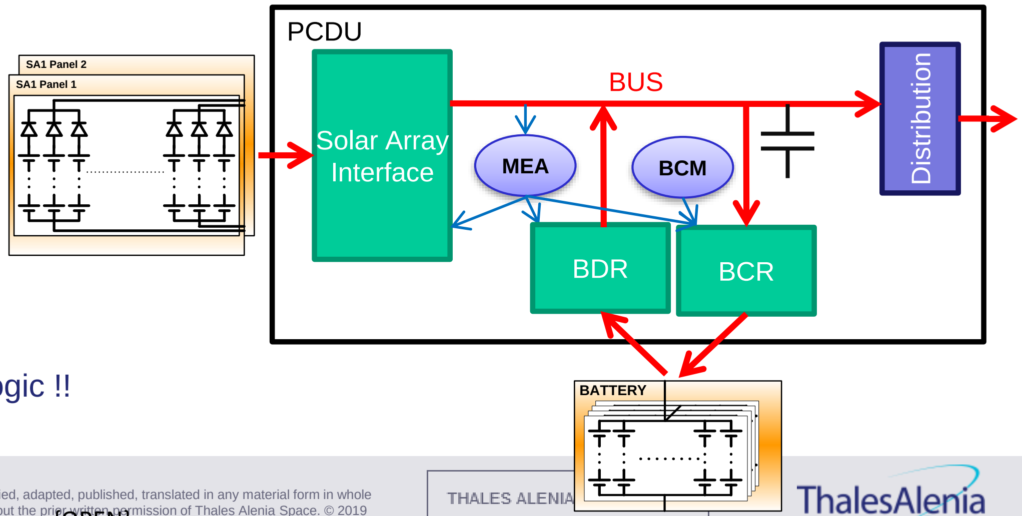
POWER MANAGEMENT, CONTROL & DISTRIBUTION / ARCHITECTURE

Bus voltage 2/3

- REGULATED BUS
  - Voltage variation is limited to about 0,5% whatever the satellite modes
    - Solar Array operative voltage is constant
  - Need of dedicated electronics to manage the battery discharge
    - Substantial power dissipation inside the PCDU during eclipse



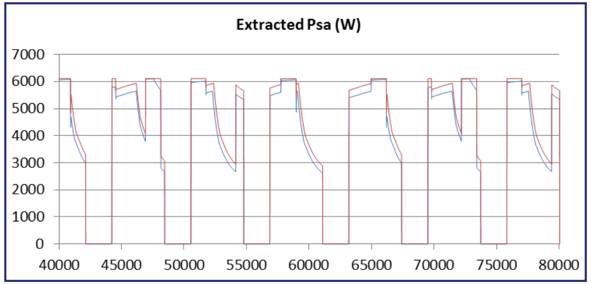
- Bus controlled by the MEA
  - Acts on Solar Array Interface if enough power is available
  - Acts on Battery Discharge Regulator in any other case
- Battery Recharge Controlled by BCM
  - Acts on the Battery Charge Regulator when power recharge is allowed



!! SpaceInspire is a regulated bus with a different control logic !!

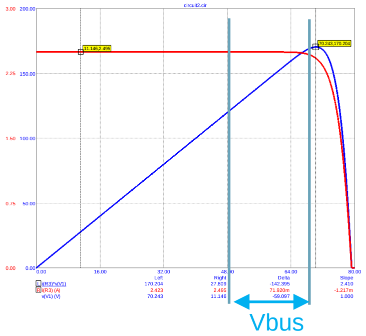
POWER MANAGEMENT, CONTROL & DISTRIBUTION / ARCHITECTURE

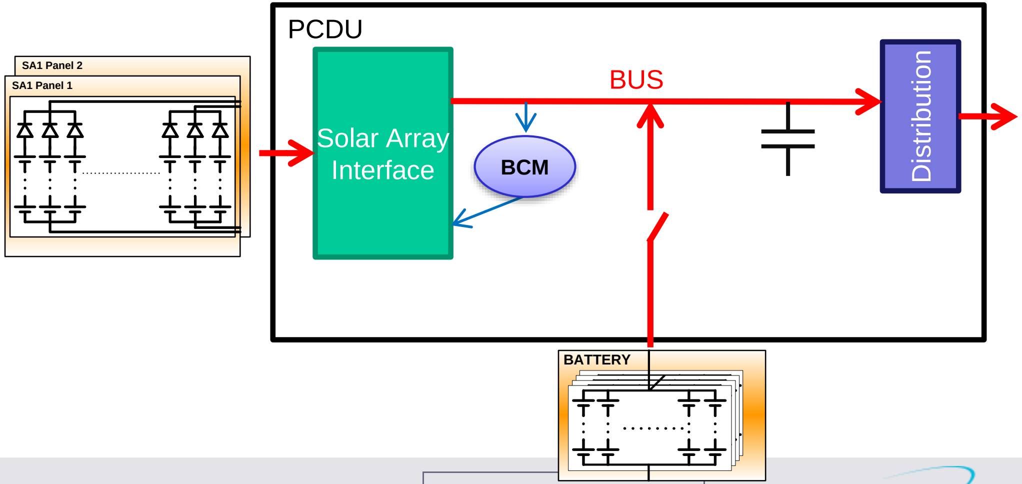
Bus Voltage 3/3



- **UNREGULATED BUS**
  - Bus Voltage follows the battery voltage
    - Solar Array extracted power depends on battery State of Charge

- Battery Recharge Controlled by BCM
  - Acts on the Solar Array interface to guarantee the battery charge





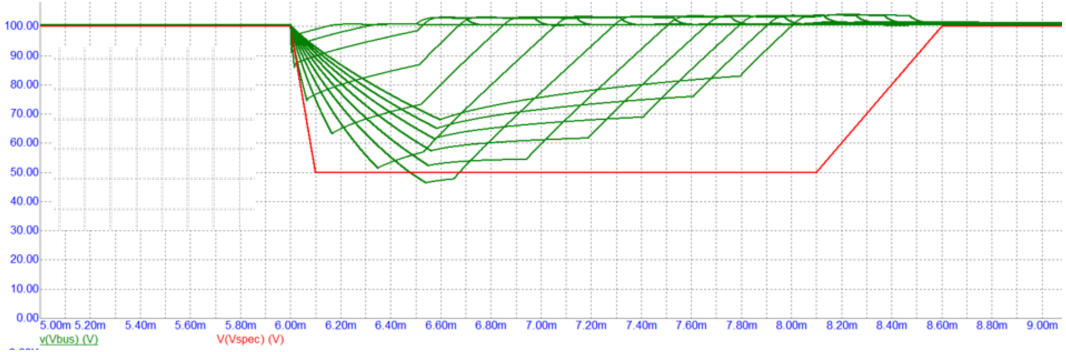
POWER MANAGEMENT, CONTROL & DISTRIBUTION / ARCHITECTURE

Distribution architecture

DISTRIBUTION CONCERNS THE WAY THE POWER IS DISTRIBUTED FROM PRIMARY & SECONDARY SOURCES TO USER'S THROUGH PCDU. TO AVOID FAILURE PROPAGATION IN CASE OF USER'S SHORT FAILURE, THESE LINES SHALL BE PROTECTED BY

- **FUSE**
  - Simplest solution
  - Imposes all the user's to be compatible with bus transients induces by fuse blowing
  - Imposes the need of extraction during AIT phase

- **ACTIVE SWITCHES**
  - Flexible solution
  - ON/OFF switching capability
  - Control of fault current



POWER MANAGEMENT, CONTROL & DISTRIBUTION / ARCHITECTURE

Distribution architecture / some definitions

- **LCL**
  - Latching Current Limiter
  - Limits current at user's switch ON or short failure during limitation time
  - Trips-OFF if limitation time is exceeded
  - ON/OFF command capability
- **R-LCL**
  - Permanent-ON LCL
  - Essential load (e.g. OBC)
  - LCL + automatic periodic re-arming
- **OP-LCL**
  - 2 switches in series

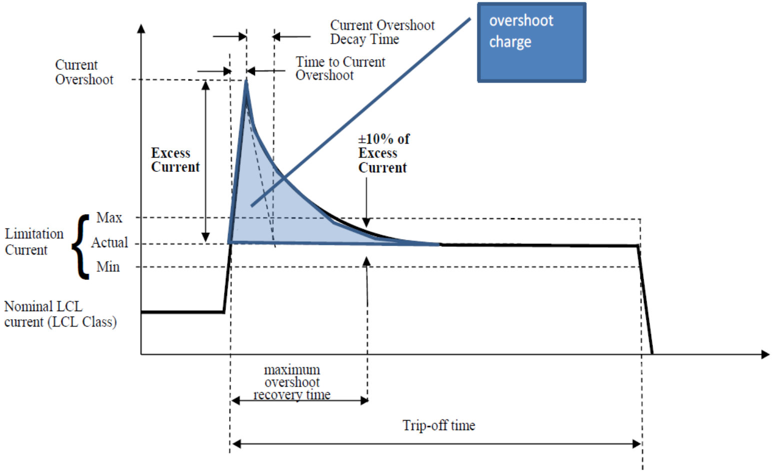


Figure 3-1: LCL overload timing diagram (case 1)

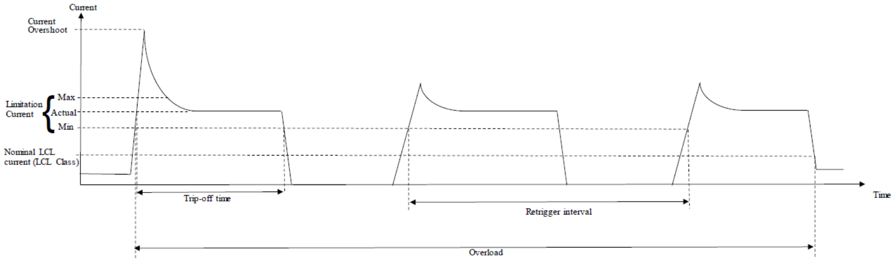


Figure 3-4: RLCL overload timing diagram

See ECSS-E-ST-20-20C for more details

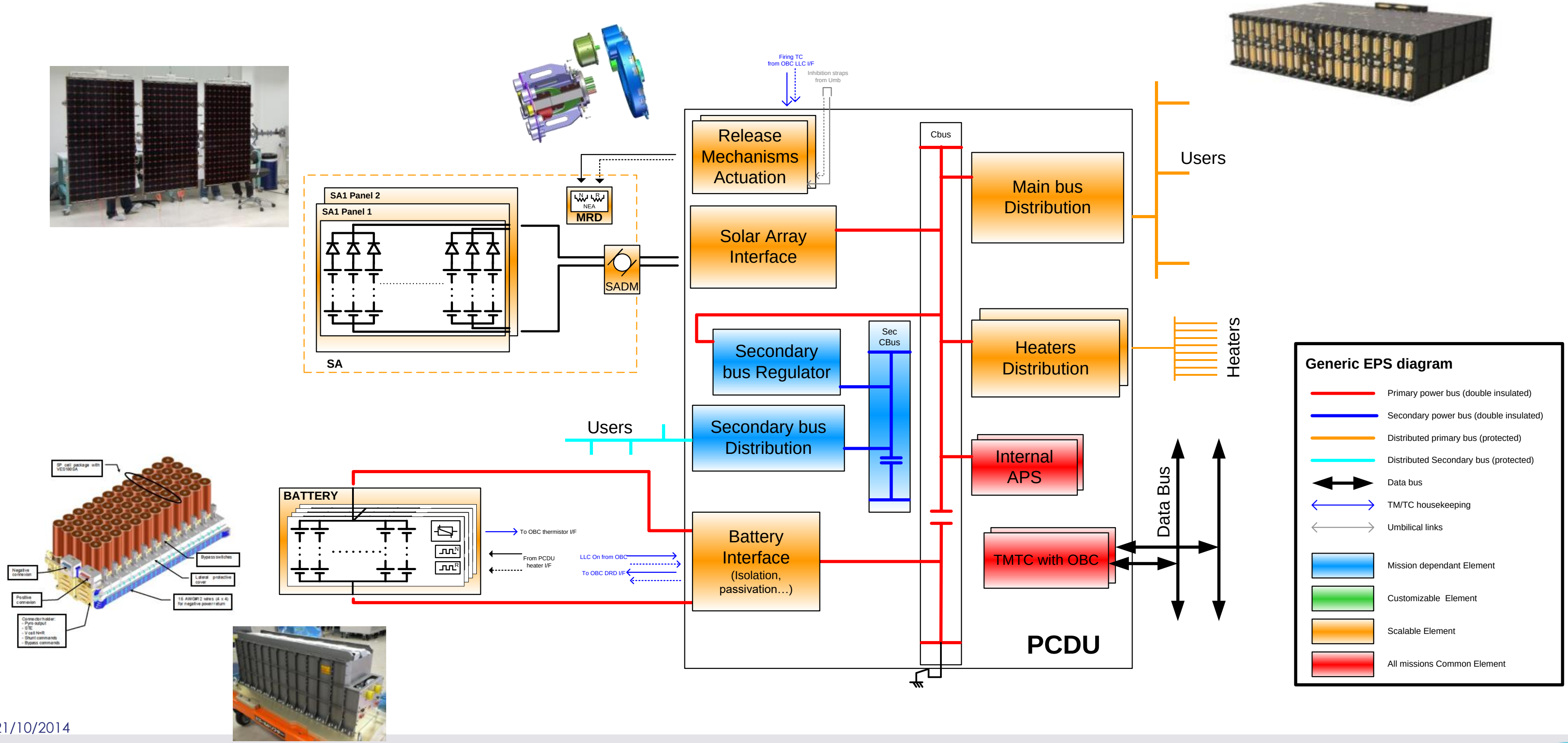
21/10/2014

POWER MANAGEMENT, CONTROL & DISTRIBUTION / ARCHITECTURE

Other constituents of PCDU

- **COMMAND OF MECHANISMS**
  - SADM motor driver
  - Antenna motor driver
  - …
- **COMMAND ON DEPLOYMENT**
  - Actuation of pyro - MRD
  - Actuation of thermal knifes - MRD
- **LI-ION BATTERY CELLS MANAGEMENT**
- **ACQUISITION OF THERMISTORS**
- **HEATERS**

POWER MANAGEMENT, CONTROL & DISTRIBUTION / ARCHITECTURE



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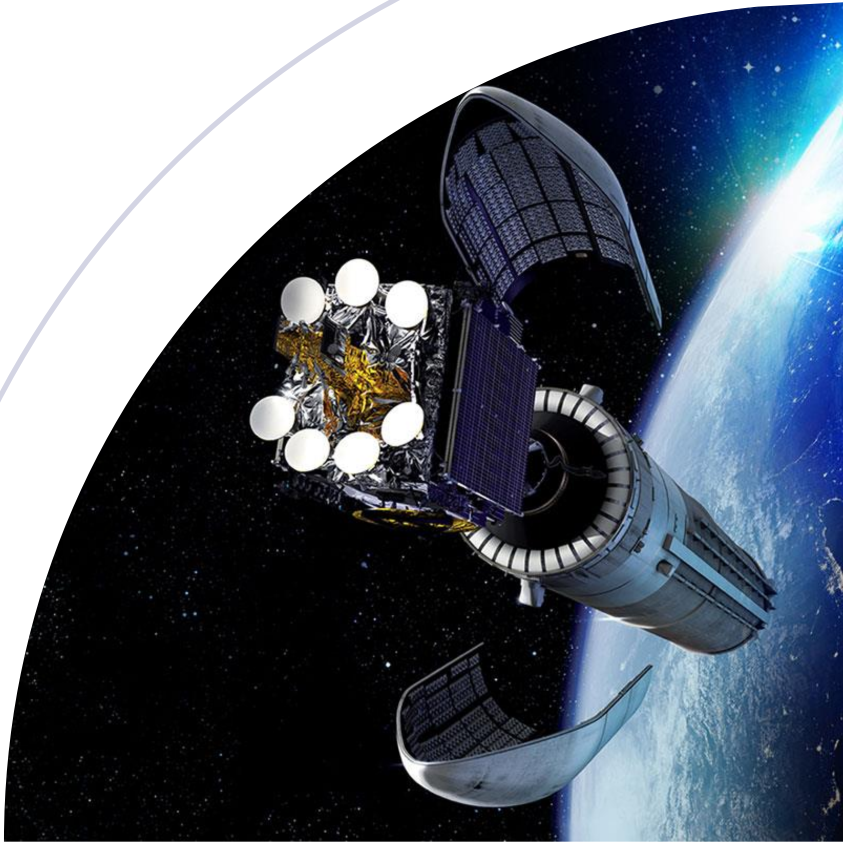
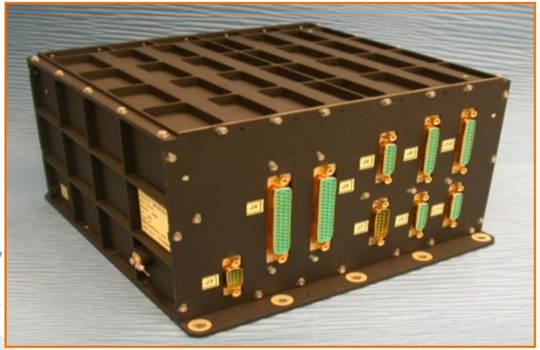




Examples µSAT

- LOW POWER: 260 W / LOW VOLTAGE : UNREGULATED BUS (22-37 V
- SOLAR ARRAY REGULATOR: BOOST CONVERTER
- NOT RELIABLE
- DISTRIBUTION FUNCTIONS
  - LCL, Pyro
  - DC/DC for secondary (+5, +-15,+20 V) + LCL protection
  - Adaptability of the distribution by paralleling of LCL's
- CNES/ASTRIUM/TAS-F MYRIADE PLATFORM BASELINE

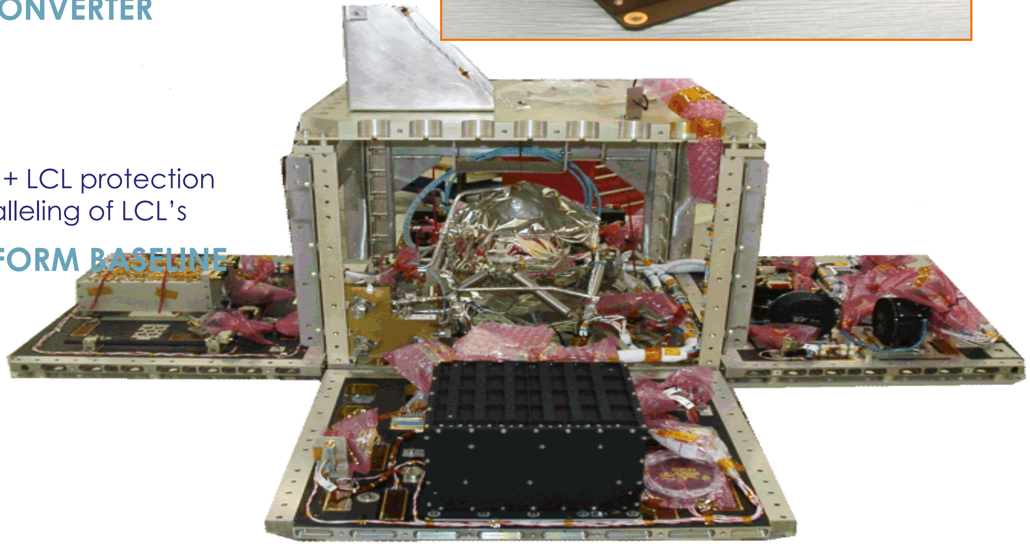
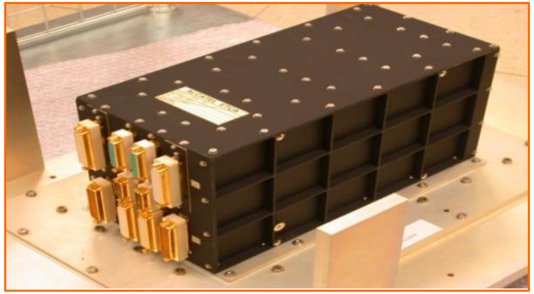


Illustration CNES

POWER MANAGEMENT, CONTROL & DISTRIBUTION / PC(D)U

Examples Scientific, earth observation & constellations

Adobe Acrobat Document



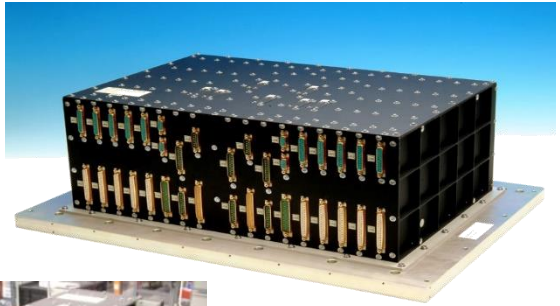
- LARGE FLEXIBILITY NEEDED
- MODULAR STRUCTURE
- LARGE FLEXIBILITY
- REDUNDANCY (TOLERANT TO ONE FAILURE)
- BUS POWER : 500 W TO 4200 W
- BUS VOLTAGE : UP TO 50 V, NON-REGULATED OR REGULATED
- SOLAR ARRAY REGULATION : MPPT OR DET (S3R OR S2R)
- LITHIUM CELLS MANAGEMENT : CELLS VOLTAGE BALANCING AND BY-PASS ELECTRONICS
- DISTRIBUTION : LCLS, FCLS, RELAYS+FUSES, HEATER SWITCHES, PYRO ELECTRONICS
- TMTC : MIL-1553B BUS OR OTHER

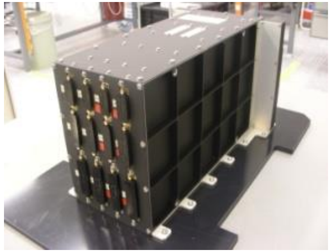
Challenges of new constellations

- USE OF COTS (COMPONENT OFF-THE-SHELF) TAKEN FROM AUTOMOTIVE PRODUCT LINES AND TESTED IN RADIATION “A POSTERIORI” – INCLUDING PLASTIC PACKAGE
- USE OF AUTOMATIVE PRODUCTION LINES
- REVIEW OF COMPLETE VALIDATION / TEST CONCEPT (BURN-IN AT PART LEVEL, SCREENING AT BOARD LEVEL, LIMITED TESTS AT S/C LEVEL,,,)

POWER MANAGEMENT, CONTROL & DISTRIBUTION / PC(D)U

Examples Scientific, Earth observation & constellation





Herschel Planck PCDU

- 1.9 kW
- Regulated bus / S3R: 28 V
- Mass: 24 kg
- 3 FMs manufactured, 2 in flight since May 2009 (missions ended mid-2013)
- Customer: Thales Alenia Space Italy (ESA)

PCDU for constellation

- 1.2 kW to 1.8 kW
- Unregulated bus / MPPT or S3R: 28 V
- Mass: 7 kg to 13kg
- G*2 : 25 FMs delivered, 24 in flight
- O3B : 8 FMs delivered, 8 in flight
- Iridium : 84 FMs and delivered, 75 in flight

ARSAT 50V PCU

- 4,2 kW
- Regulated bus / S3R: 50 V
- Mass: 19 kg
- 1 EM + 2 FMs in flight (Oct-14 & Sept-15)

SENTINEL-3 PCDU

- 2,1 kW
- Unregulated bus / S3R: 28 V
- Mass: 16.2 kg
- 1 EM + 4 FMs delivered, FM1&2 in flight since February 2016, April 2018

SENTINEL-1 PCDU

- 5,8 kW
- Unregulated bus/S3R 60V +

Regulated bus: 28 V

- Mass: 23.2 kg
- 4 FMs delivered, FM1&2 in flight since April 2014 and 2016

FRENCH OBSERVATION SATELLITE PCDU

- 2,7 kW
- Unregulated bus/MPPT: 28 V
- Mass: 19.2 kg
- 1 EM + 6 FMs delivered

POWER MANAGEMENT, CONTROL & DISTRIBUTION / PC(D)U

Examples GEO high power

SPACEBUS 4000 PCU

- FULL REGULATED BUS 6 TO 27 KW / 100 V
- SOLAR ARRAY REGULATION: S3R
- NO DISTRIBUTION FUNCTION (PCU ONLY)
- FLIGHT HERITAGE : 84 PCU'S, 58 IN FLIGHT, 480 YEARS

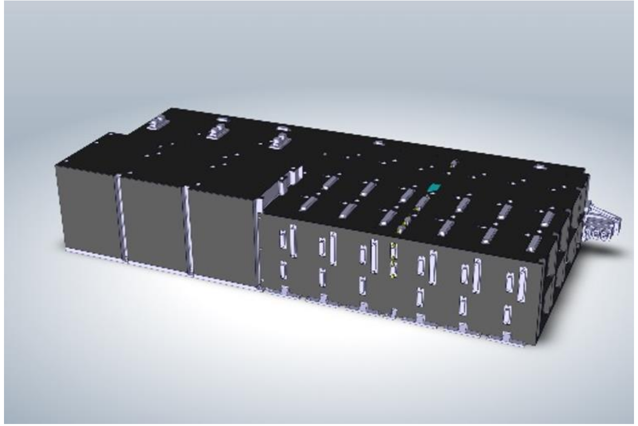
POWER MANAGEMENT, CONTROL & DISTRIBUTION / PC(D)U

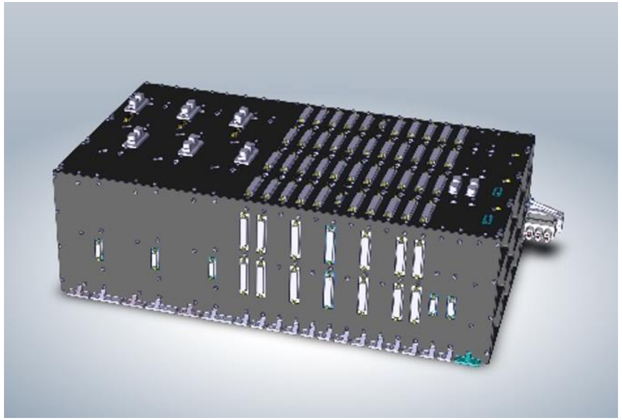
Examples GEO high power

SPACE INSPIRE

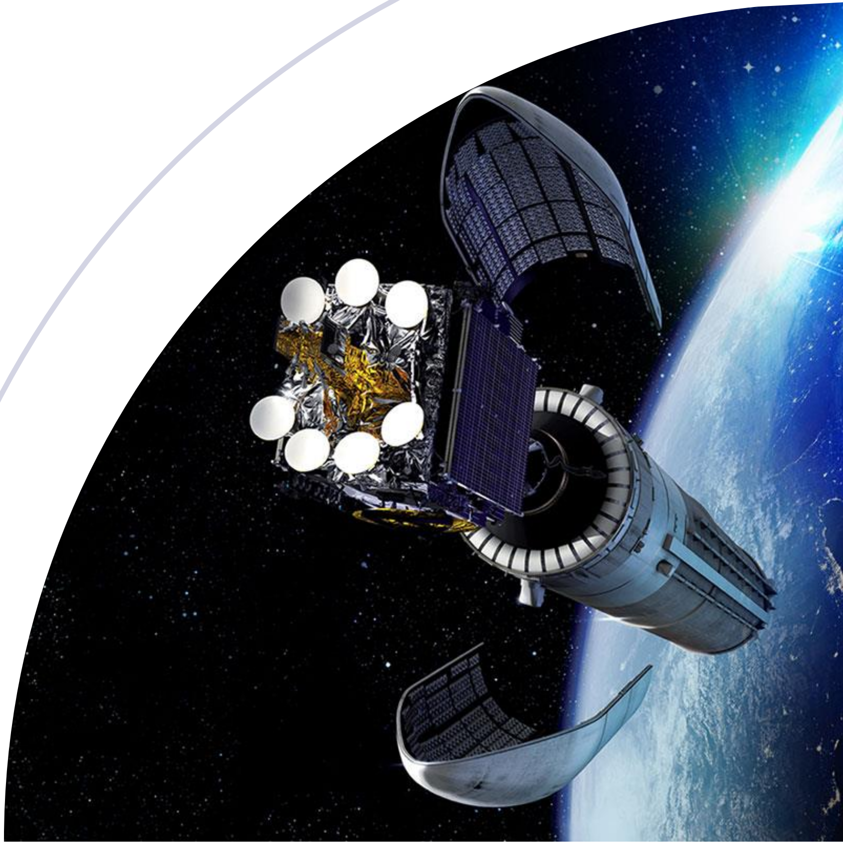
- **HPU**
  - Full regulated bus 8 to 32 kW / 100 V
  - Solar array regulation: S3R
  - Distribution by fuses

- **ACE (PCDU PART)**
  - Secondary Power Bus 28V
  - Heaters
  - Pyros / MRD
  - Distribution by fuses





AGENDA



EPS SIZING

1. Orbit selection (altitude & inclination trade-off's)
2. Bus voltage trade-off
3. Bus regulation trade-off
4. Battery sizing
5. Power conditioning topology trade-off
6. Solar array's surface

POWER BUDGET

///Study case #1

STUDY OF A MICRO SATELLITE TO TARGET SHIP BASED AND GROUND BASED RADARS

- Lifetime: 12 years
- Orbit: Leo

PAYLOAD REQUIREMENTS

- Acquisition in sun & eclipse phases
- Bus power of 650 W
  - Max power to be considered
  - Sum of all user's needs (AOCS, payloads, emitters, receivers, thermal control…) including distribution losses (LCL, fuse, harness)
  - Worst case consumption in all satellite phases (acquisition, data transmission, night & day modes, seasons variation on thermal control, …)
  - Excluding power conditioning needs

POWER BUDGET

///Orbit selection

/ ALTITUDE TRADE-OFF

- Lower than 1000 km (to avoid Van Allen belts impacts on radiation level)
- Above 500 km to ensure that the cluster altitude can be maintained during lifetime (atmospheric drag effect)
- Instrument precision is better at low altitude but instrument coverage increases with altitude

-> Circular orbit of 600 km altitude has been selected among several candidates (out of the scope of this study case, based essentially on payload needs)

/ INCLINATION TRADE -OFF

- Polar orbit for best possible coverage worldwide
- Sun-synchronous orbit as other candidate

$$T^2 = \frac{4\pi^2 r^3}{Gm_e}$$

- G is the gravitational constant (=6.67 x 10^{-11} $m^3.s^{-2}.kg^{-1}$),
- m_e is the mass of the Earth (=5.98 x 10^{24} kg),
- r is the distance from the satellite to the centre of the Earth (in metres),

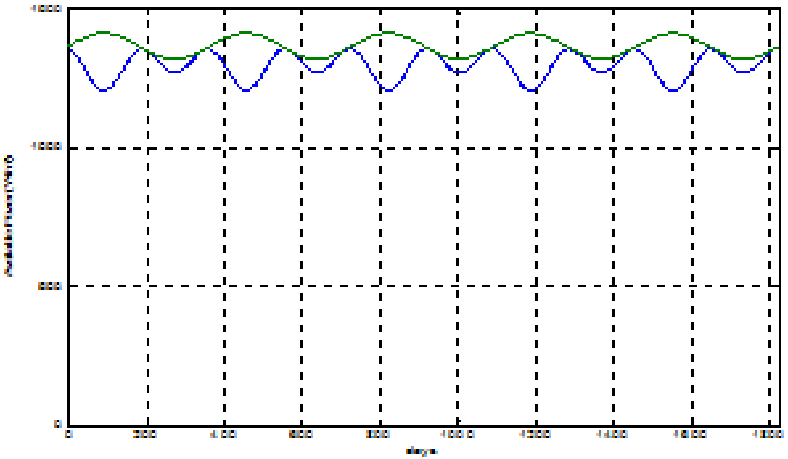
$r = r_E + h$, where $r_E = 6378$ km

Orbit characteristics		
Average height	600 km	600 km
Period	97 min	97 min
Eccentricity	0.001 (circular orbit)	0.001 (circular orbit)
Inclination	90 ° (polar orbit)	98 ° (sun-synchronous)
Eclipse duration	21.3 min	30 min

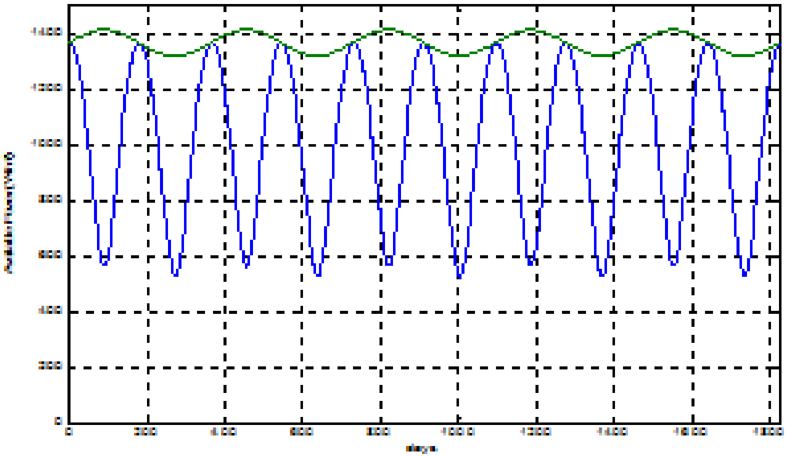
POWER BUDGET

/// Orbit selection / Inclination trade-off

Orbit selection / Inclination trade-off



SA flux (sun-synchronous orbit)
- > Min SA flux = 1220 W



SA flux (polar orbit)
- > Min SA flux = 520 W

POWER BUDGET

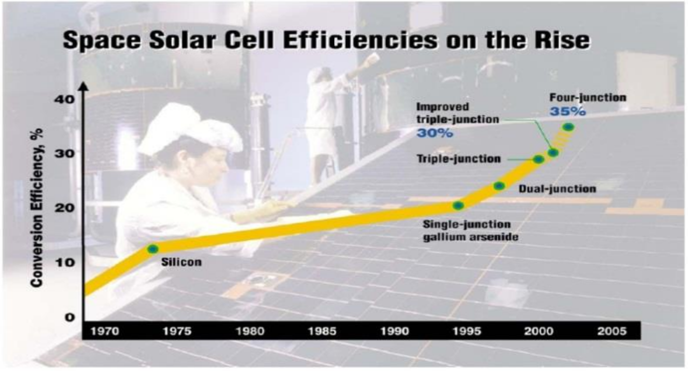
///Orbit selection / Inclination trade-off

- **EPS SIZING SHALL CONSIDER WORST CASE CONDITIONS OF ILLUMINATION AND EOL PHOTOVOLTAIC EFFICIENCY OF SA CELLS. THIS LEADS TO THE FOLLOWING DATA (WORST CASE FIGURES).**

	Sun-synchronous	Polar
Minimum SA flux (W/m²)	1220	520
BOL SA cell efficiency	28 %	
EOL/BOL ratio	76.5 %	
Total available SA power (W / m²)	260	110

Cell manufacturer data

- **NOTE THAT PHOTOVOLTAIC EFFICIENCY EOL/BOL RATIO TAKES INTO ACCOUNT THE FOLLOWING ELEMENTS (SA PANEL MANUFACTURER DATA)**
  - 5-years mission lifetime
  - radiation effects
  - UV and meteoritic impact
  - effect of ATOX density (aggressive and corrosive environment tied to the LEO) on cover glass protection
  - Effect of temperature (including earth albedo)



POWER BUDGET

/// EPS sizing: Bus voltage trade-off

- **28 V**
  - Compatible with bus power (< 1 kW) → remember: Recommended ESA rule: $P < U^2/0.5$ → $U=\sqrt{P*0.5}=\sqrt{1kW*0.5}=22V$
  - High hardware heritage
- **50 V**
  - Reduced current levels
  - Reduced harness & power dissipations

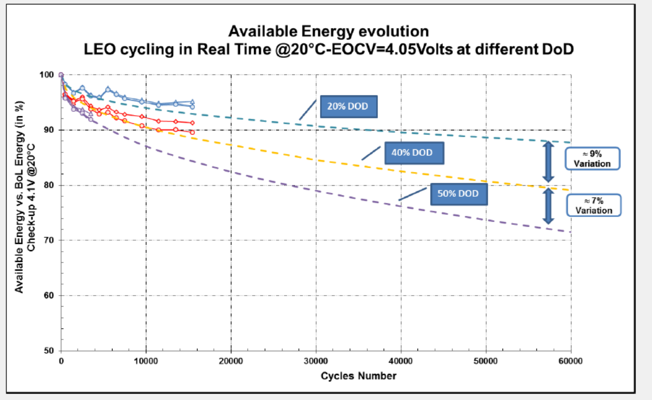
/// EPS sizing: Bus regulation trade-off

- **REGULATED POWER BUS – MAIN HYPOTHESIS**
  - BDR (Battery => bus) conversion efficiency=94%
- **UNREGULATED POWER BUS – MAIN HYPOTHESIS**
  - Internal losses (Battery => bus) internal connections=1%
  - BAT to PCDU harness losses : 3%

NOTE: PCDU LOW LEVEL CONSUMPTION: 30 W FOR BOTH CONFIGURATIONS

/// EPS sizing: Battery sizing

- **MAX DOD OF 40 % CONSIDERED FOLLOWING**
  - Orbit characteristics (period and eclipse)
  - Mission duration 10 years => 55 000 cycles
- **BATTERY DISSIPATION (AT BATTERY LEVEL)**
  - 25 W (discharge)
  - 15 W (charge)



POWER BUDGET

/// EPS sizing / bus regulation trade-off & Battery sizing

	Regulated bus	Unregulated bus
User's power in eclipse (W)	650	
PCDU losses during eclipse (W)	70	35
Satellite power requirement in eclipse (W)	720	685
Harness & Battery losses (W)	50	45
Total battery power need in eclipse (W)	**770**	**730**
Eclipse duration (min)	30	
Battery useful cycled energy requirement EOL(W h)	385	365
Battery energy mission degradation (40% DoD / 56000 cycles)	30 %	
Battery useful cycled energy requirement BOL(W h)	550	520
Battery energy requirement BOL (Wh)	1375	1300

30W +(1 - 94% or 1%)

Pout/n – Pout + LL

25W +3%

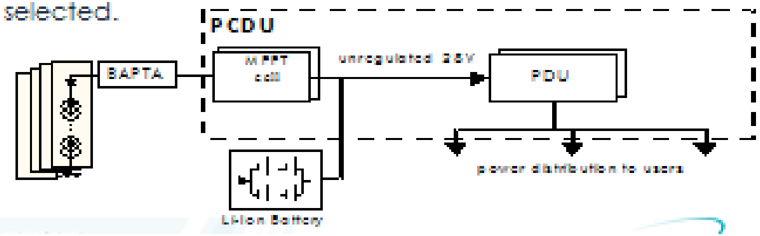
Pecl*0,5 h

20% fading,
5% calendar loss
5% Rbat degradation

Energy/DOD

DoD=40%

Slight advantage for URB coupled with lower PCDU mass / complexity.
If no specific requirement on payload (including EMC), URB is selected.

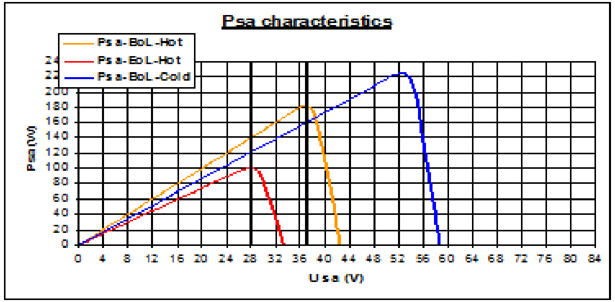


POWER BUDGET

///EPS sizing: Conditioning topology trade-off (Unregulated bus topology)

MPPT

- Power converter efficiency: 95 %
- Control efficiency: ability to track the maximum power whatever the battery state is (charged, discharged, with or without failure, ...): 99 % accuracy



DET

- S3R conversion efficiency: 98 %

/// EPS sizing: Battery data (based on previous selection)

- Battery recharge duration = 90 % of sunlight duration

<u>NOTE:</u> CONSIDERING 28 V URB WITH 40 % DOD, BATTERY VOLTAGE IS COMPRISED BETWEEN 28V & 37V IN NOMINAL OPERATING CASES

POWER BUDGET

/// EPS sizing: Conditioning topology trade-off

	MPPT	DET
Battery power requirement in eclipse (W)	730	
Eclipse duration (min) / Battery charge duration (min)	30 / 60	
Battery charge power need (W)	365	
Harness, BAT & PCDU losses (W)	30	
Battery recharge power need (W)	**395**	
User's power need in sunlight (W)	650	
Battery recharge power need (W)	395	
PCDU low level (W)	30	
Total bus power needs (W)	**1075**	
SA conditioning losses (W)	70	20
TOTAL SA power needs (W)	1145	1095
SA efficiency (W/m²)	261	248
Minimum SA surface requirement (m²)	4,4	4,4

(97 -30)*0.9

((Pecl)*30 min)/T_{charge}

1% PCDU; 3% harness;
15W BAT

5 % MPPT & 1% tracking
vs. 2 % DET

Non-optimization : 5%

Result is similar, thus DET is chosen as it requires less electronics components

No impact on SA size (considering nominal / constant SA illumination)

	Sun-synchronous	Polar
Minimum SA flux (W/m²)	1220	620
BOL SA cell efficiency	28 %	
EOL/BOL ratio	76.5 %	
Total available SA power (W / m²)	261	111

POWER BUDGET

///Study case #2

STUDY OF A COPERNICUS TYPE SATELLITE

- Lifetime: 12.5 years
- Orbit: Leo

PAYLOAD REQUIREMENTS

- Acquisition in sun & eclipse phases
- Bus power of 2kW
    - Max power to be considered
    - Sum of all user's needs (AOCS, payloads, emitters, receivers, thermal control…) including distribution losses (LCL, fuse, harness)
    - Worst case consumption in all satellite phases (acquisition, data transmission, night & day modes, seasons variation on thermal control, …)
    - Excluding power conditioning needs

POWER BUDGET

///Orbit selection

ALTITUDE TRADE-OFF

- Lower than 1000 km (to avoid Van Allen belts impacts on radiation level)
- Above 500 km to ensure that the cluster altitude can be maintained during lifetime (atmospheric drag effect)
- Instrument precision is better at low altitude but instrument coverage increases with altitude

-> Circular orbit of 800 km altitude has been selected among several candidates (out of the scope of this study case, based essentially on payload needs)

INCLINATION FIXED

- Needed for the mission

$$T^2 = \frac{4\pi^2 r^3}{G m_e}$$

- G is the gravitational constant (=6.67 x 10^{-11} $m^3.s^{-2}.kg^{-1}$),
- m_e is the mass of the Earth (=5.98 x 10^{24} kg),
- r is the distance from the satellite to the centre of the Earth (in metres),

$r = r_E + h$, where $r_E = 6378$ km

LEO orbit choice		
	polar orbit	sun synchronous
Inclination	90°	98°
Eclipse duration [min]	?	?
Min SA flux [W/m²]	520	1220
BOL cell efficiency [%]	0.28	
EOL/BOL ratio [%]	0.765	
EOL WC available SA Power [W/m²]	?	?

POWER BUDGET

/// Orbit selection

LEO orbit choice		
	polar orbit	sun synchronous
Inclination	90°	98°
Eclipse duration [min]	22.2	31.3
Min SA flux [W/m²]	520	1220
BOL cell efficiency [%]	0.28	
EOL/BOL ratio [%]	0.765	
EOL WC available SA Power [W/m²]	111	261

/// Bus Voltage Choice

Bus voltage choice	
Min Bus voltage	?

POWER BUDGET

/// EPS sizing: Bus voltage trade-off

Bus voltage choice	
Min Bus voltage	31.6

- **28 V**
  - Compatible with bus power (< 1 kW)
  - High hardware heritage
- **50 V**
  - Reduced current levels → remember: Recommended ESA rule: $P < U^2/0.5$ → $U>\sqrt{P*0.5}=\sqrt{2kW*0.5}=32V$
  - Reduced harness & power dissipations

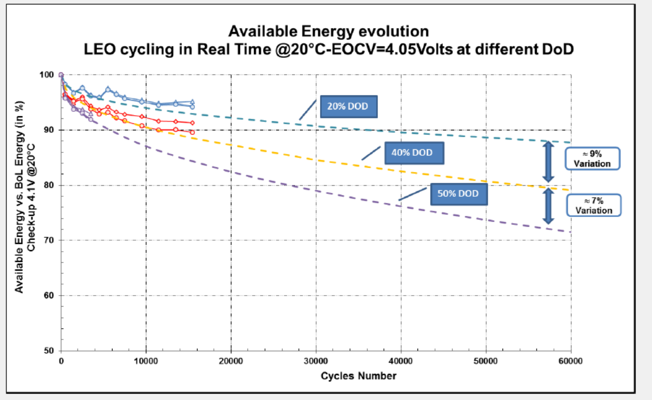
/// EPS sizing: Bus regulation trade-off

- **REGULATED POWER BUS – MAIN HYPOTHESIS**
  - BDR (Battery => bus) conversion efficiency=94%
- **UNREGULATED POWER BUS – MAIN HYPOTHESIS**
  - Internal losses (Battery => bus) internal connections=1%
  - BAT to PCDU harness losses : 3%

NOTE: PCDU LOW LEVEL CONSUMPTION: 30 W FOR BOTH CONFIGURATIONS

/// EPS sizing: Battery sizing

- **MAX DOD OF 45 % CONSIDERED FOLLOWING**
  - Orbit characteristics (period and eclipse)
  - Mission duration 10 years => 55 000 cycles
- **BATTERY DISSIPATION (AT BATTERY LEVEL)**
  - 25 W (discharge)
  - 15 W (charge)



POWER BUDGET

/// EPS sizing / bus regulation trade-off & Battery sizing

Battery design + Bus regulation choice		
	Regulated Bus	Unregulated Bus
User's power in eclispe [W]	?	
PCDU losses during eclipse [W]	?	?
Satellite power requirement in eclipse [W]	?	?
Harness + Battery losses [W]	?	?
Total battery power need in eclipse [W]	?	?
Eclipse duration min [min]	?	
Battery useful cycled energy requirement EOL [Wh]	?	?
Battery energy mission degradation	0.3	
Battery useful cycled energy requirement BOL [Wh]	?	?
Battery energy requirement BOL [Wh]	?	?

POWER BUDGET

/// EPS sizing / bus regulation trade-off & Battery sizing

Battery design + Bus regulation choice		
	Regulated Bus	Unregulated Bus
User's power in eclispe [W]	2000	
PCDU losses during eclipse [W]	150	50
Satellite power requirement in eclipse [W]	2150	2050
Harness + Battery losses [W]	90	87
Total battery power need in eclipse [W]	2240	2137
Eclipse duration min [min]	31.3	
Battery useful cycled energy requirement EOL [Wh]	1168	1115
Battery energy mission degradation	0.3	
Battery useful cycled energy requirement BOL [Wh]	1669	1592
Battery energy requirement BOL [Wh]	3709	3538

POWER BUDGET

/// EPS sizing / bus regulation trade-off & Battery sizing

VL48E	VL10E	VES 100	VES 140	VES 180	MPS*	VL8P	
48	10	28	39	50	5.8	7.5	Guaranteed capacity Ah
3.6	3.6	3.6	3.6	3.6	3.6	3.6	Mean voltage at C/1.5
4.1	4.1	4.1	4.1	4.1	4.1	4.1	End of charge voltage V at
170	36	100	140	180	20	100	Energy Wh
150	139	118	126	165	133	118	Specific energy Wh/kg
250	129	185	250	250	65	104	Height mm
54	33.8	54	54	53	18x65**	47	Diameter mm
1.13	0.25	0.81	1.13	1.11	0.15	0.38	Weight kg
Qualified	Qualified	Qualified	Qualified	Qualified	Qualified	Qualified	Status
GEO, LEO	GEO, LEO	LEO	GEO, MEO	GEO, MEO	LEO	Launcher	Main application

<u>NOTE:</u> CONSIDERING 50 V URB WITH 45 % DOD, BATTERY VOLTAGE IS COMPRISED BETWEEN 50V & 67.5V IN NOMINAL OPERATING CASES

POWER BUDGET

/// EPS sizing / bus regulation trade-off & Battery sizing

Battery design	
Bus Max Voltage	?
Minimum #batteries in series	?
Minimum #batteries in series	?
Minimum #batteries in parallel	?
Minimum #batteries in parallel	?
Battery energy BOL [Wh]	?
#batteries	?
Battery Weight [kg]	?

NOTE: CONSIDERING 50 V URB WITH 45 % DOD, BATTERY VOLTAGE IS COMPRISED BETWEEN 50V & 67.5V IN NOMINAL OPERATING CASES

POWER BUDGET

/// EPS sizing / bus regulation trade-off & Battery sizing

Battery design	
Bus Max Voltage	67.4
Minimum #batteries in series	16.4
Minimum #batteries in series	17
Minimum #batteries in parallel	5.8
Minimum #batteries in parallel	6
Battery energy BOL [Wh]	3672
#batteries	102
Battery Weight [kg]	26

NOTE: CONSIDERING 50 V URB WITH 45 % DOD, BATTERY VOLTAGE IS COMPRISED BETWEEN 50V & 67.5V IN NOMINAL OPERATING CASES

POWER BUDGET

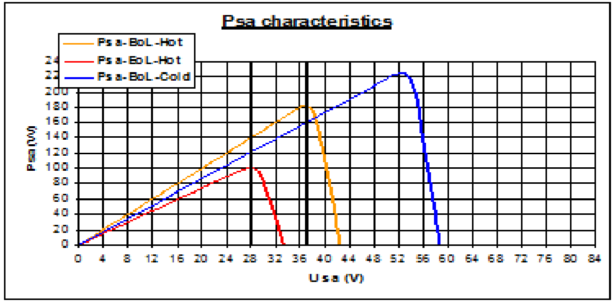
///EPS sizing: Conditioning topology trade-off (Unregulated bus topology)

MPPT

- Power converter efficiency: 95 %
- Control efficiency: ability to track the maximum power whatever the battery state is (charged, discharged, with or without failure, …): 99 % accuracy

DET

- S3R conversion efficiency: 98 %



/// EPS sizing: Battery data (based on previous selection)

- Battery recharge duration = 75 % of sunlight duration

NOTE: CONSIDERING 50 V URB WITH 45 % DOD, BATTERY VOLTAGE IS COMPRISED BETWEEN 50V & 67.5V IN NOMINAL OPERATING CASES

POWER BUDGET

/// EPS sizing: Conditioning topology trade-off

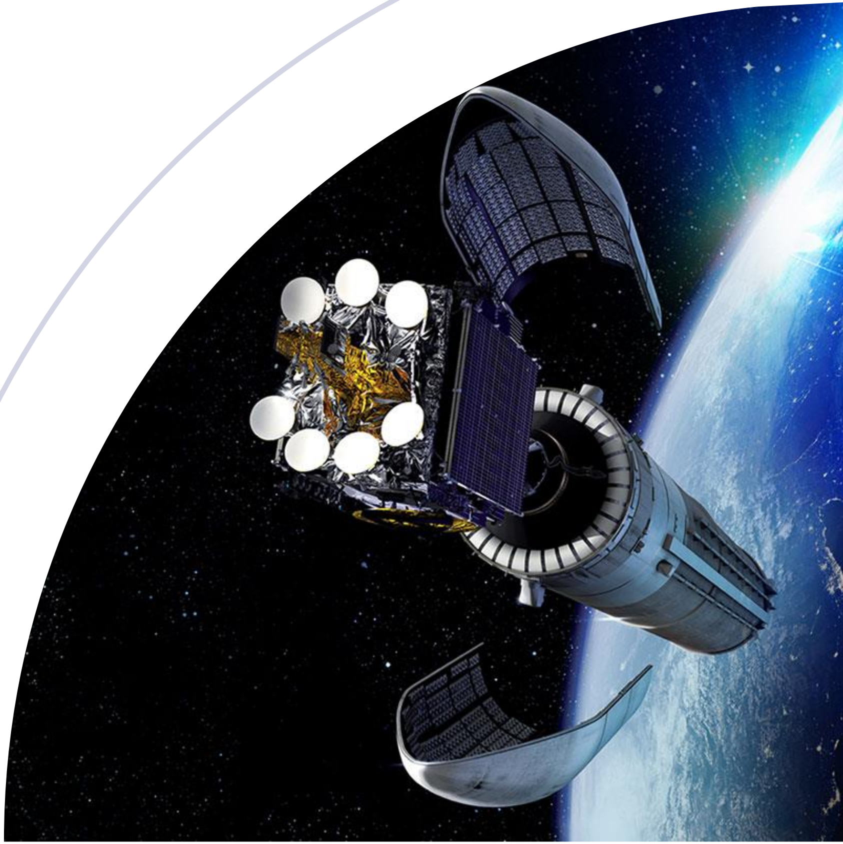
SA interface design		
	MPPT	DET
Battery Power requirement Eclipse [W]	?	
Eclipse duration [min]	?	
Battery charge duration [min]	?	
Battery charge power need [W]	?	
Harness, BAT & PCDU lossed [W]	?	
Battery recharge power need [W]	?	
User's Power need in sunlight [W]	2000	
Battery recharge power need [W]	?	
PCDU low level [W]	30	
Total Bus Power need [W]	?	
SA conditionning losses [W]	?	?
Total SA power need [W]	?	?
SA efficiency [W/m²]	?	?
Minimum SA surface requirement [m²]	?	?

POWER BUDGET

/// EPS sizing: Conditioning topology trade-off

SA interface design		
	MPPT	DET
Battery Power requirement Eclipse [W]	2137	
Eclipse duration [min]	31.3	
Battery charge duration [min]	52	
Battery charge power need [W]	1282	
Harness, BAT & PCDU lossed [W]	66	
Battery recharge power need [W]	1348	
User's Power need in sunlight [W]	2000	
Battery recharge power need [W]	1348	
PCDU low level [W]	30	
Total Bus Power need [W]	3378	
SA conditionning losses [W]	203	68
Total SA power need [W]	3581	3446
SA efficiency [W/m²]	261	248
Minimum SA surface requirement [m²]	13.7	13.9

AGENDA

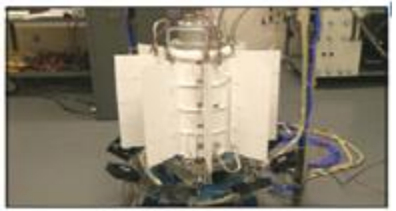


CONCLUSIONS

/// The design of any Power Subsystem is strongly linked with System analyses (Attitude & Orbit, Mission, Operations)

/// The electrical architecture of spacecrafts is not standard

- / **UNREGULATED OR REGULATED POWER BUS**
- / **VOLTAGE (28 V, 50 V, 100 V, ...)**
- / **CONDITIONING (S3R, MPPT, ...)**
- / **PROTECTIONS (RELIABLE OR NOT)**
- / **DISTRIBUTION (FUSE, LCL, ...)**
- / ...



MMRTG Engineering Unit

AND SHALL BE ADAPTED NEARLY ON CASE BY CASE ....

CONTACTS

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THANK YOU
FOR YOUR ATTENTION