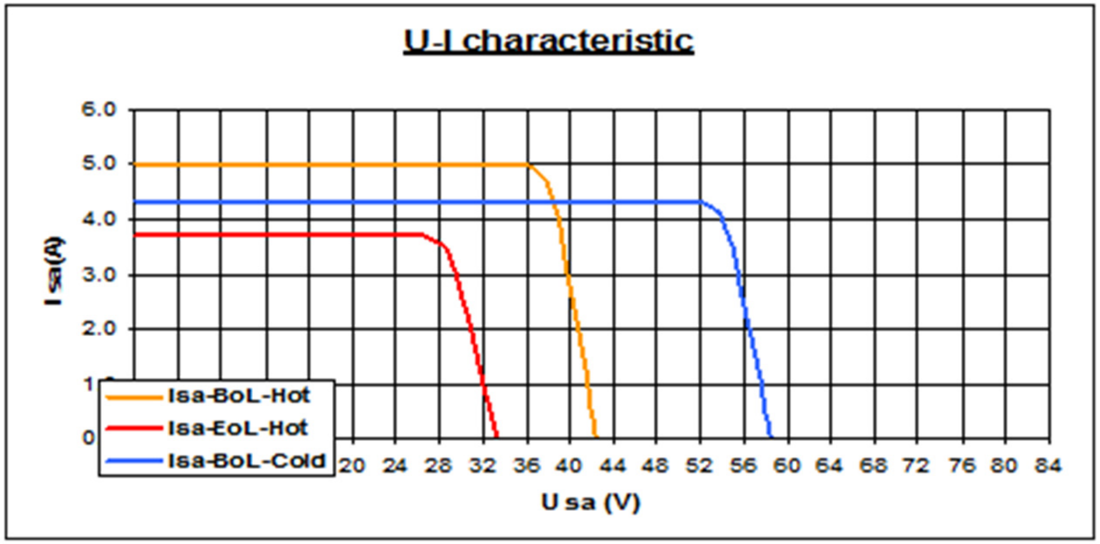
PRIMARY POWER SOURCES / SOLAR CELLS & ARRAYS

/// Optimal working point – at max. available power



Pmax is largely depending of temperature & ageing

PRIMARY POWER SOURCES / SOLAR CELLS & ARRAYS

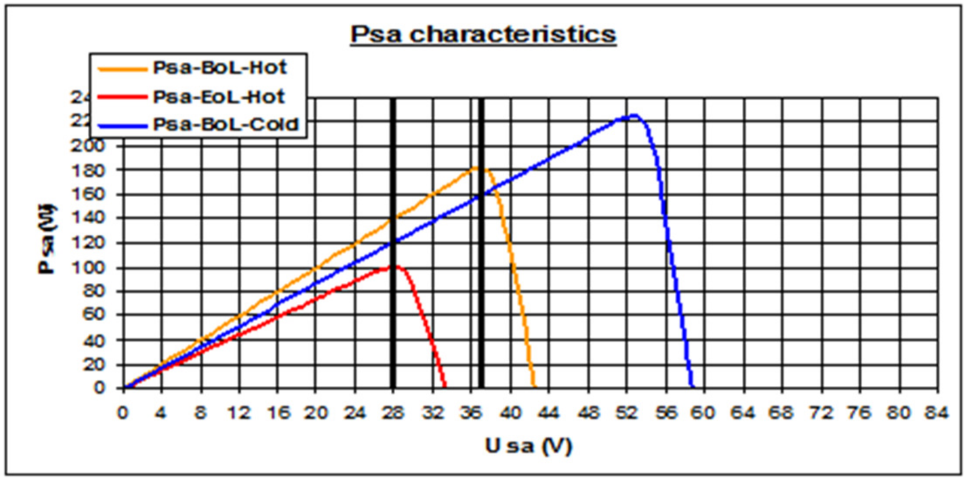
/// Temperature is linked to

/ THE INCOMING FLUX

- Direct solar flux
- Albedo (ratio between reflected flux vs incoming flux)
- IR flux of the earth

/ THE OUTCOMING FLUX

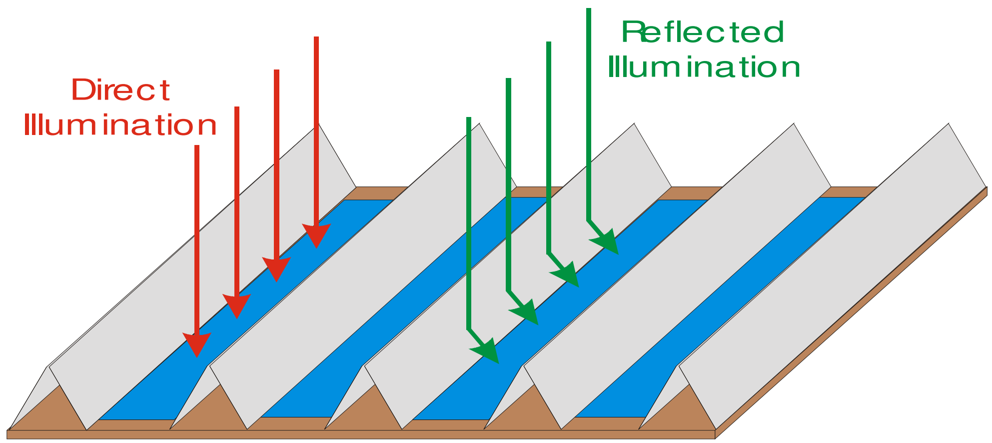
- Flux reflected by the cells
- Power delivered to the satellite
- IR flux of the front and rear part of the SA



PRIMARY POWER SOURCES / SOLAR CELLS & ARRAYS

/// Concentrators / Advantages

- CONCENTRATE SA FLUX ON SA CELLS
- REDUCE SA CELLS SURFACE
- BASED ON REFLECTORS OR ON LENS



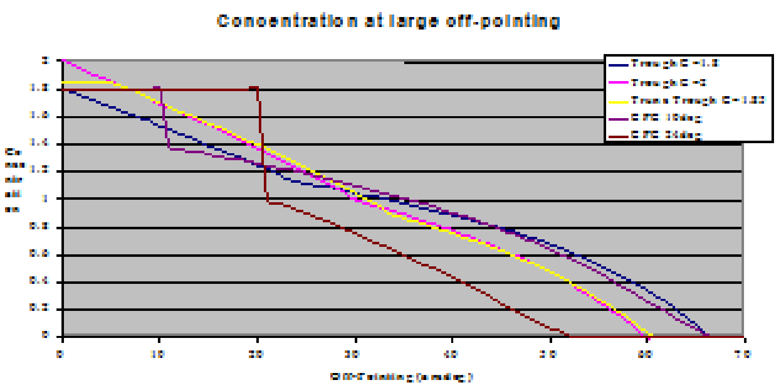
PRIMARY POWER SOURCES / SOLAR CELLS & ARRAYS

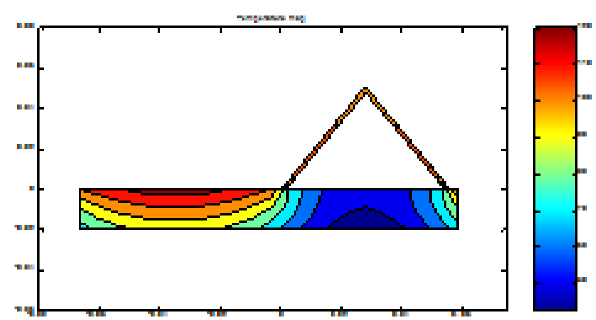
/// Concentrators / Drawbacks

/ NOT COMPATIBLE WITH LARGE OFF-POINTING ANGLE

- Oblique rays can hit the reflectors two times and then they may be reflected back to space
- Off-pointing (hors pointage) property of SA concentrator is not compliant with un-stabilized S/C

/ INDUCES HIGHER THERMAL CONSTRAINTS ON CELLS AND SA PANEL



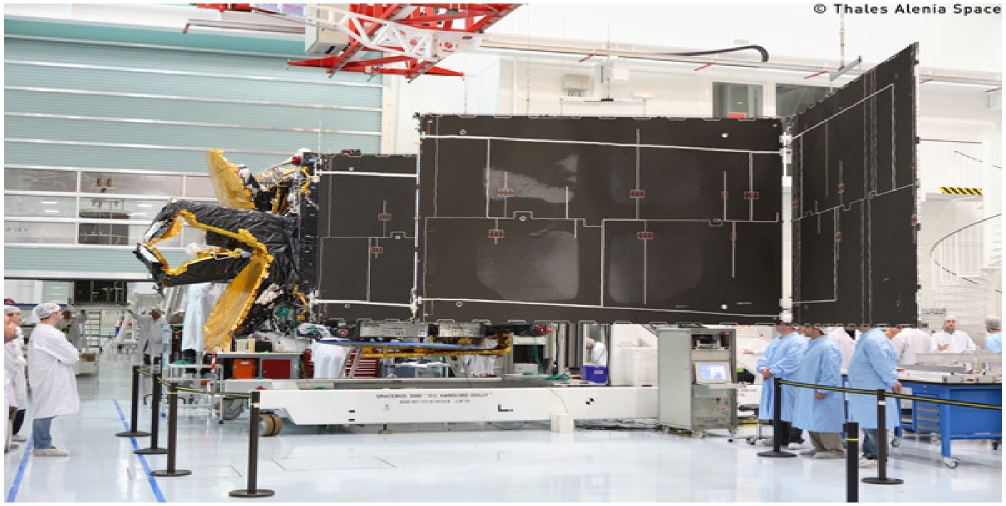


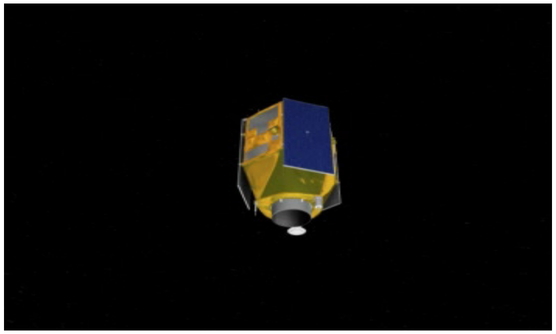
/ OUTGASSING MAY BECOME CRITICAL

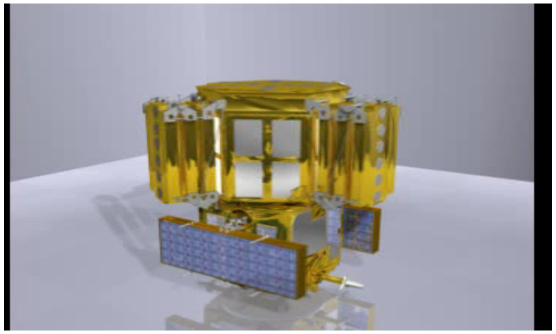
PRIMARY POWER SOURCES / SOLAR CELLS & ARRAYS

/// Solar Arrays – mechanisms (hold-down & release)

- **SA ARE FOLD (REPLIÉS) DURING LAUNCH**
- **DEPLOYMENT IS**
  - Initiated by pyro actuation (or thermal knifes)
  - Controlled by the use of hinge mechanisms







PRIMARY POWER SOURCES / SOLAR CELLS & ARRAYS

/// Solar Arrays – mechanisms (orientation)

- **MOBILE SA IS CONTROLLED BY SADM (SOLAR ARRAY DEPLOYMENT MOTOR)**
- **CURRENT IS TRANSFERRED TO S/C MAIN PART VIA BAPTA (BEARING AND POWER TRANSFER ASSEMBLY)**
- **TENSIONING WIRES – ACHIEVE MINIMUM FUNDAMENTAL FREQUENCY OF THE ARRAY (AOCS CONSTRAINTS): (ATTITUDE & ORBIT CONTROL SYSTEM)**

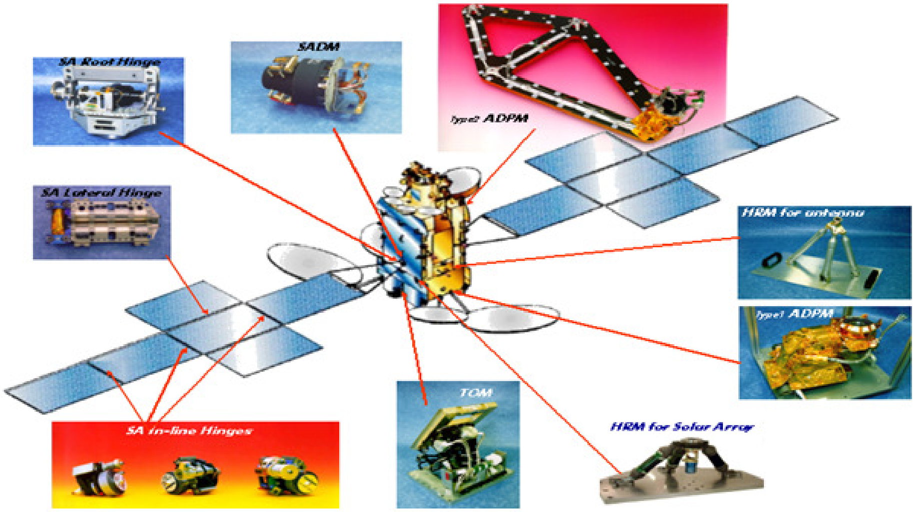


_Illustration Thales Alenia Space

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 / FUEL CELLS

/// Electromechanical devices performing a controlled chemical reaction (oxydation) 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, …

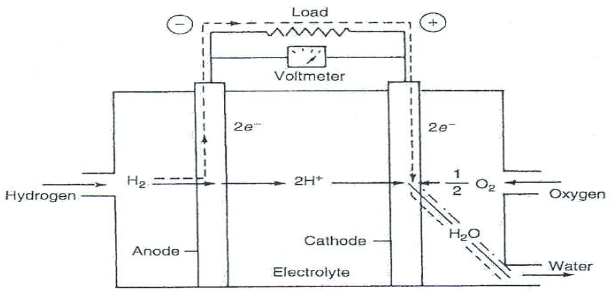


Figure 10.10 Schematic of a hydrogen/oxygen fuel cell. At the anode–electrolyte interface, hydrogen dissociates into hydrogen ions and electrons. The hydrogen ions migrate through the electrolyte to the cathode interface where they combine with the electrons that have traversed the load [2] (From Angrist, 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

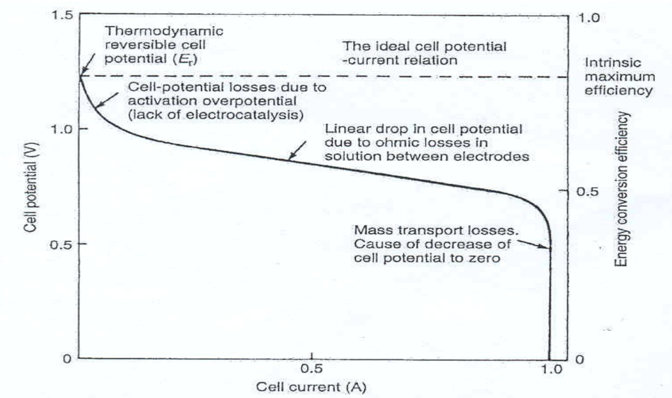


Figure 10.11 Typical cell potential and efficiency–current relation of an electrochemical electricity producer showing regions of major influence of various types of overpotential losses (Source [10])

/// Performance summary of fuel cells for space use

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 (TRAÎNÉE) IS IMPORTANT (VERY LOW ORBITS) -> REDUCTION OF PROPELLANT USED FOR ORBIT CONTROL**

PRIMARY POWER SOURCES / RTG (RADIOISOTOPE THERMOELECTRIC GENERATOR)

///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**

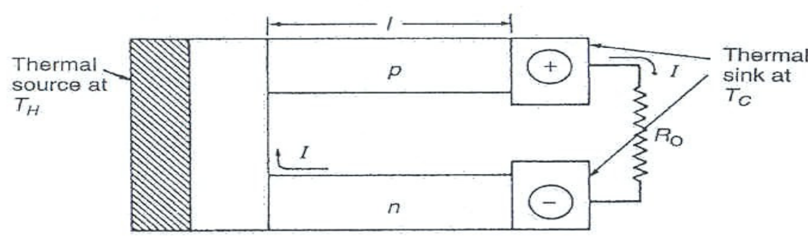


Figure 10.12 Schematic diagram of a semiconductor radioisotope generator (From Angrist, S. W. (1982) *Direct energy conversion*, 4th edn, Copyright Allyn and Bacon, New York)

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)

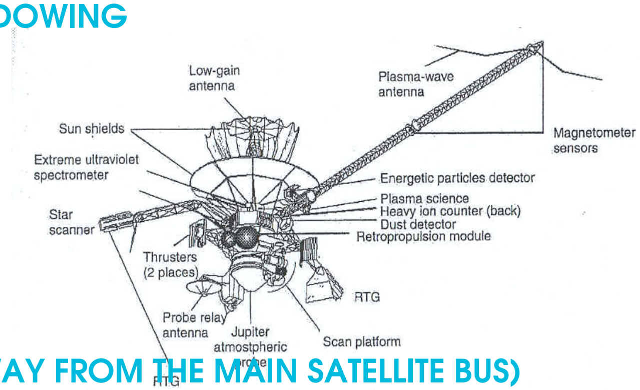


Figure 10.13 The Galileo spacecraft configuration, showing the position of the RTG sources (Courtesy of NASA/JPL/Caltech)

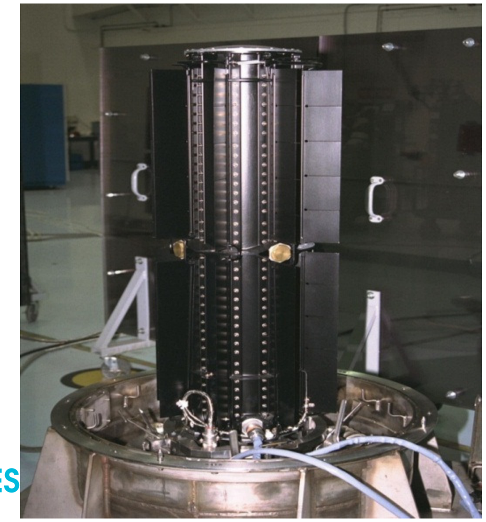
/// Drawbacks

- 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

PRIMARY POWER SOURCES / RTG & OTHERS

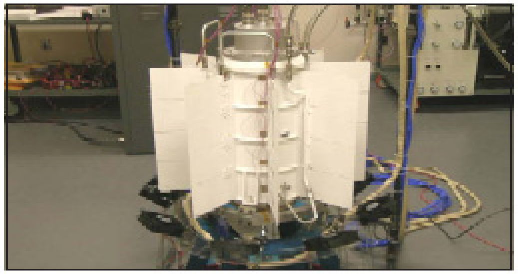
///Example of RTG

/ APOLLO 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 PROCES (AS FOR TERRESTRIAL NUCLEAR POWER PLANTS)
- USED TO DRIVE THERMOELECTRIC CONVERTER AS RTG



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

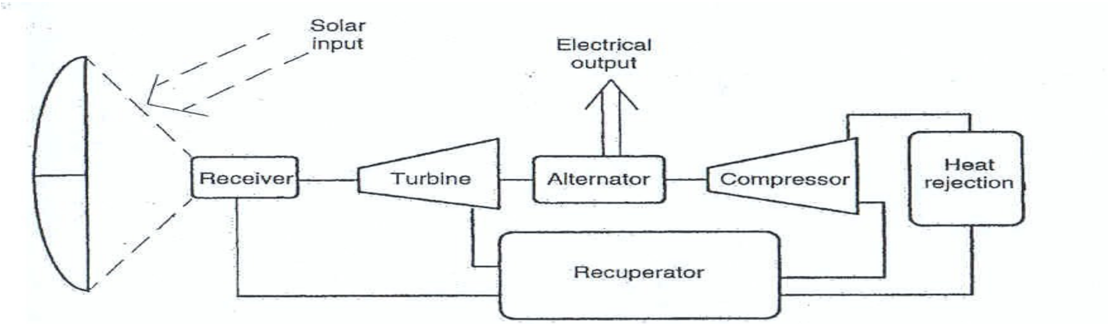
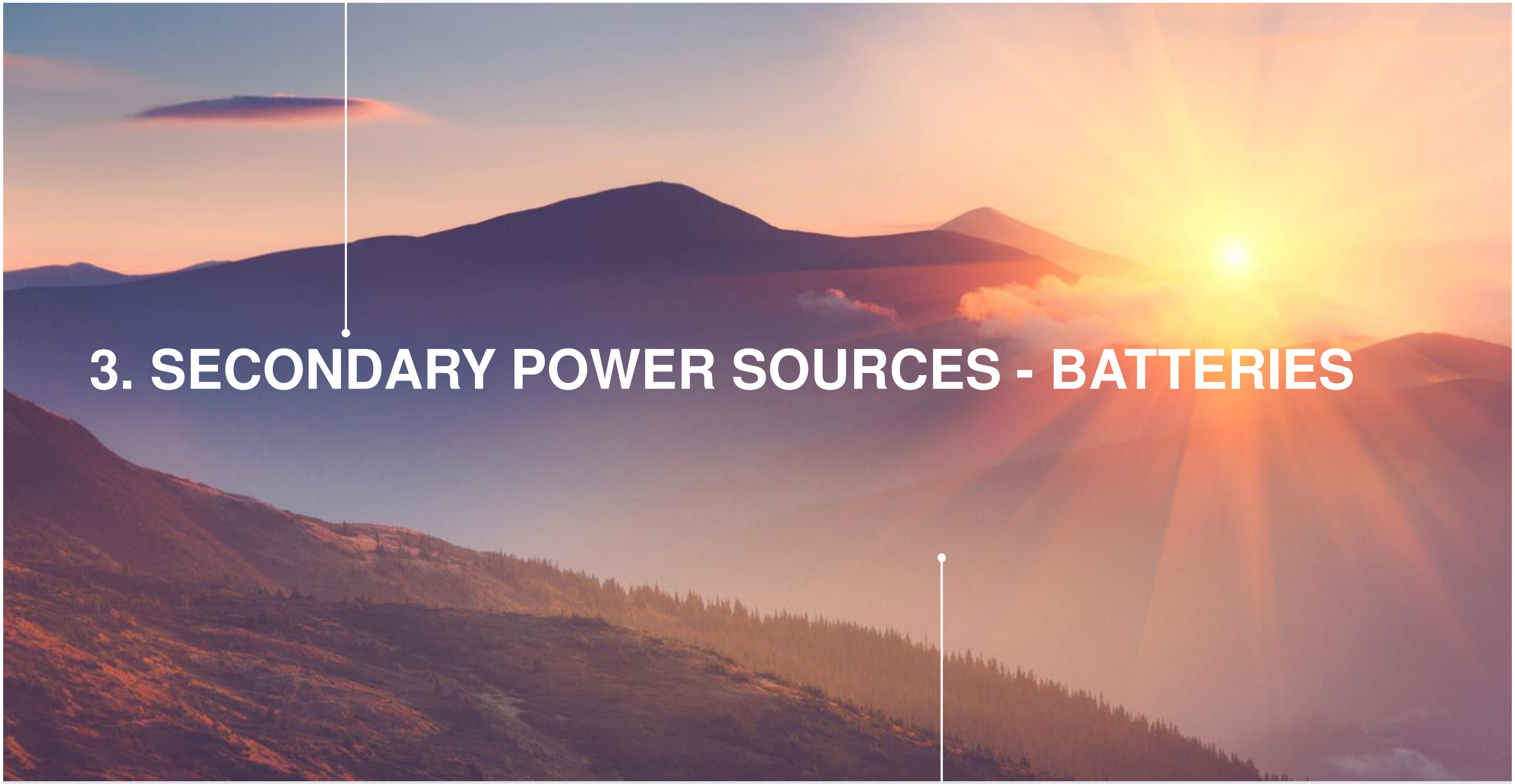


Figure 10.14 Solar dynamic Brayton cycle



3. SECONDARY POWER SOURCES - BATTERIES

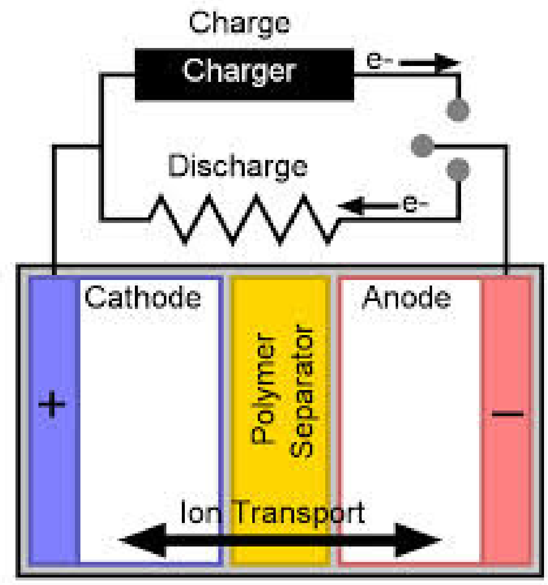
SECONDARY POWER SOURCES

/// 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).

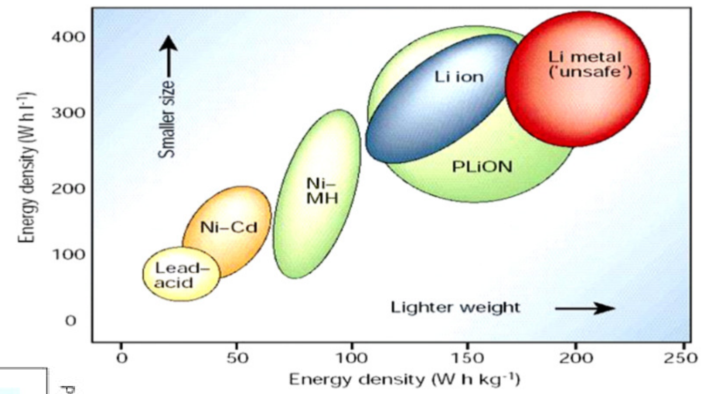


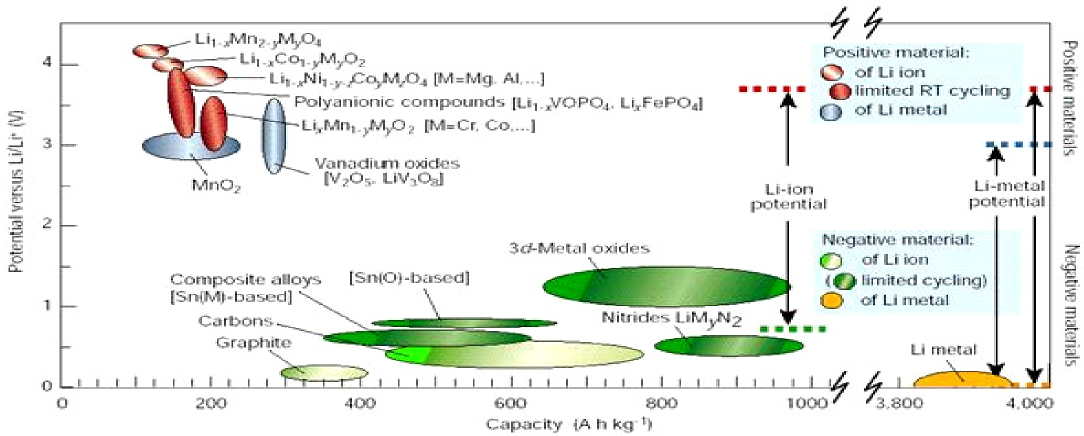
SECONDARY POWER SOURCES

/// Accumulators

/ CRITICAL PARAMETERS

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

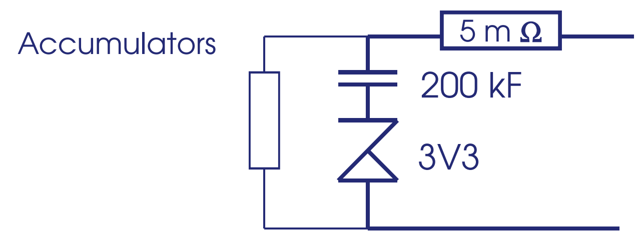


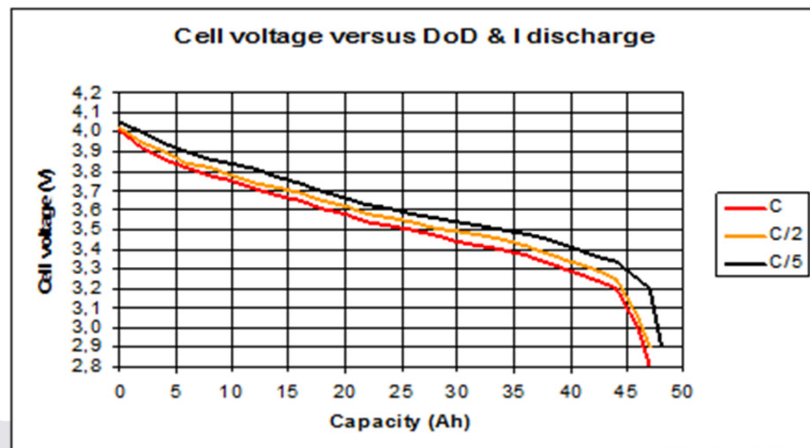


SECONDARY POWER SOURCES

/// Typical Characteristics

- **CAPACITY : A BATTERY'S CAPACITY IS THE AMOUNT OF ELECTRIC CHARGE IT CAN STORE. THE MORE ELECTRODE MATERIAL THERE IS IN THE CELL THE GREATER THE CAPACITY OF THE CELL. A SMALL CELL HAS LESS CAPACITY THAN A LARGER CELL WITH THE SAME CHEMISTRY, AND THEY DEVELOP THE SAME OPEN-CIRCUIT VOLTAGE. 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. FOR AN 1.6 A.H BATTERY, A RATE OF 1C WOULD MEAN A CURRENT OF 1.6 AMP. A C/2 RATE WOULD MEAN A CURRENT OF 0.8 AMP**
- **CELL OPEN CIRCUIT VOLTAGE : DIFFERENCE BETWEEN CELL ELECTRODE POTENTIALS**
- **VOLTAGE RANGE**
  - 4.1V -> 3.3V
- **SERIES RESISTANCE**
  - 1mΩ -> 10m Ω
- **LEAKAGE CURRENT**
  - 0mA -> 5mA





SECONDARY POWER SOURCES

/// Comparison of performances

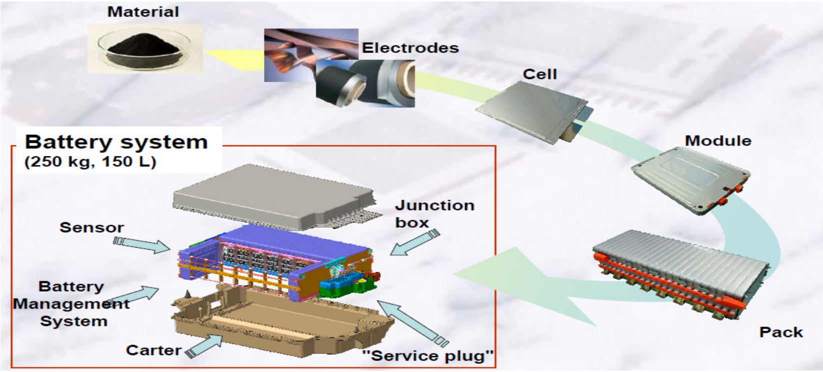
Type	NiCd	NiH2	Li-Ion
Energy/kg (Wh/kg)	30-40	55-65	100-130
Energy/l (Wh/l)	110	80	200-250
Discharge voltage mean (V)	1.25	1.25	3.5
Working temperature (°C)	[-5;+15]	[0;+10]	[+15;+25]
Charge current (A)	⇒C/10 (GEO) ⇒C/2 (LEO)	⇒C/8 (GEO) ⇒0.7C (LEO)	C/10 ⇒ C/3
Discharge current (A)	⇒2C	⇒C	⇒C
Energy efficiency (%)	75	75	90
Max. voltage (V)	1.55	1.6	4.0
Min. voltage (V)	1.0	1.0	2.7
Capacity (Ah)	4 ⇒ 50	30 ⇒ 350	1.5,2.2,26,40
Life duration in Geo	7 years at 50 % of DoD	15 years at 80 % of DoD	15 years at 80 % of DoD
Life duration in Leo	10 years at 15 % of DoD	5 years at 40 % of DoD	7 years at 30 % of DoD

SECONDARY POWER SOURCES

/// Battery

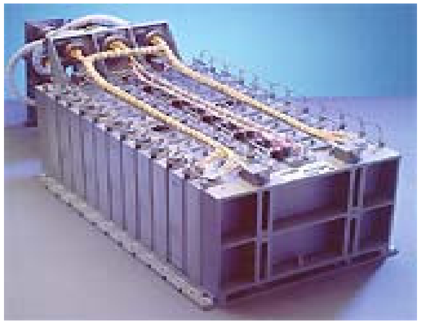
ROLE: SUPPORT THE SOLAR ARRAY DURING

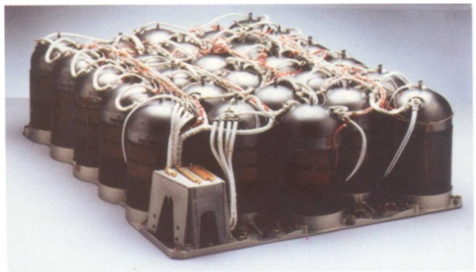
- LEOP phases
- Eclipses
- Loss of sun pointing
- Peak power demands
- …



SERIES / PARALLEL ASSEMBLING OF ACCUMULATOR CELLS

- Put in series to reach the required voltage
  - 22V-37V in LEO
  - EUROSTAR 2000: 42.5 V
  - EUROSTAR 3000 & SPACEBUS 3000: 50 V
  - SPACEBUS 4000: 100 V
- In parallel to reach the desired capacity





NICKEL HYDROGEN BATTERY

SECONDARY POWER SOURCES

/// Battery tapering & energy sizing

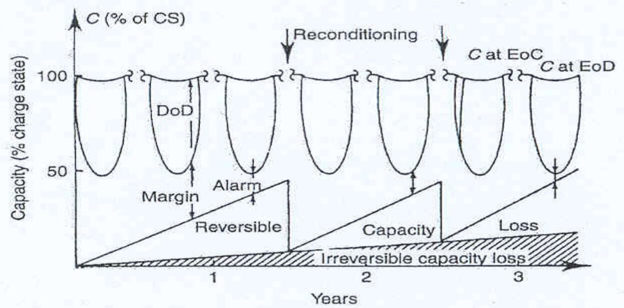
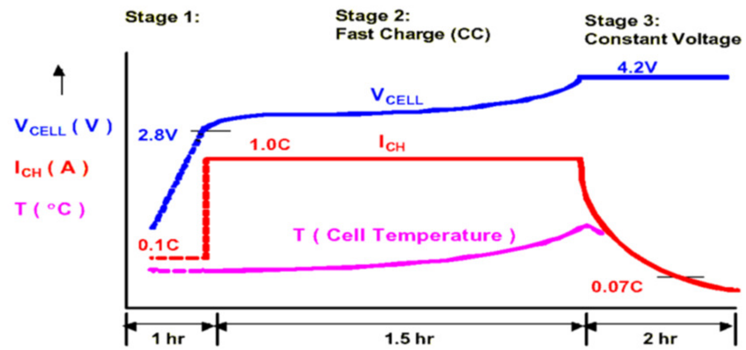
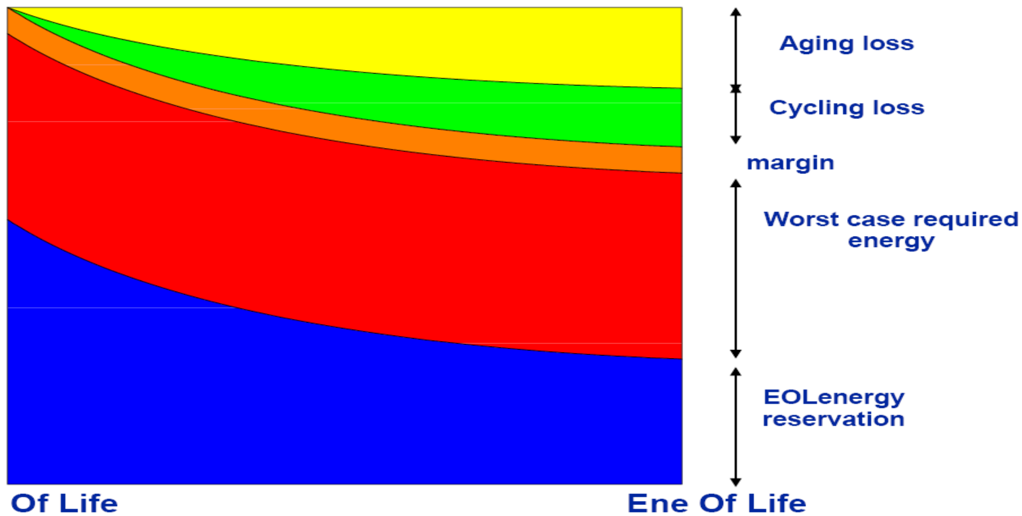


Figure 10.16 Battery reconditioning via complete discharge to improve battery capacity. Both reversible and irreversible capacity loss occurs [17] (Reproduced by permission of European Space Agency and P. Montalenti)





SECONDARY POWER SOURCES

BoL	SAFT NiCd VOS 40	SAFT NiH2 93 AN	SAFT Li-Ion VOS140	SAFT Li-Ion MP76065	SONY Li-Ion 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 %

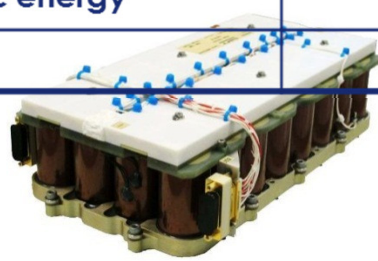
SECONDARY POWER SOURCES

BoL	SPOT-5	EURASIASAT	STENTOR	SKYBRIDGE
Configuration	24s-1p VOS40	27s-1p 93AN	11s-2p VOS140	12s-4p VOS140
Capacity	46 Ah	93 Ah	80 Ah	154 Ah
Mean voltage	29 V	37 V	39.6 V	43 V
Energy	1325 Wh	3415 Wh	3168 Wh	6620 Wh
Mass	47.4 kg	66 kg	34 kg	72 kg
Dimensions	467X261X260mm	863X441X310mm	490X380X290mm	910X520X300mm
Specific energy	28 Wh/kg	52 Wh/kg	93 Wh/kg	92 Wh/kg
Density	42 Wh/l	29 Wh/l	59 Wh/l	47 Wh/l

SECONDARY POWER SOURCES

/// Sampling of batteries

BoL	PLEIADES	µSAT
Configuration	8s-100p 18650HC	8s-10p 18650HC
Capacity	140 Ah	14 Ah
Mean voltage	30 V	30 V
Energy	4200 Wh	420 Wh
Mass	40,4 kg	4 kg
Dimensions	2 x (355 x 295 x 180 mm)	226 x 166 x 95 mm
Specific energy	105 Wh/kg	105 Wh/kg
Density	170 Wh/l	118 Wh/l



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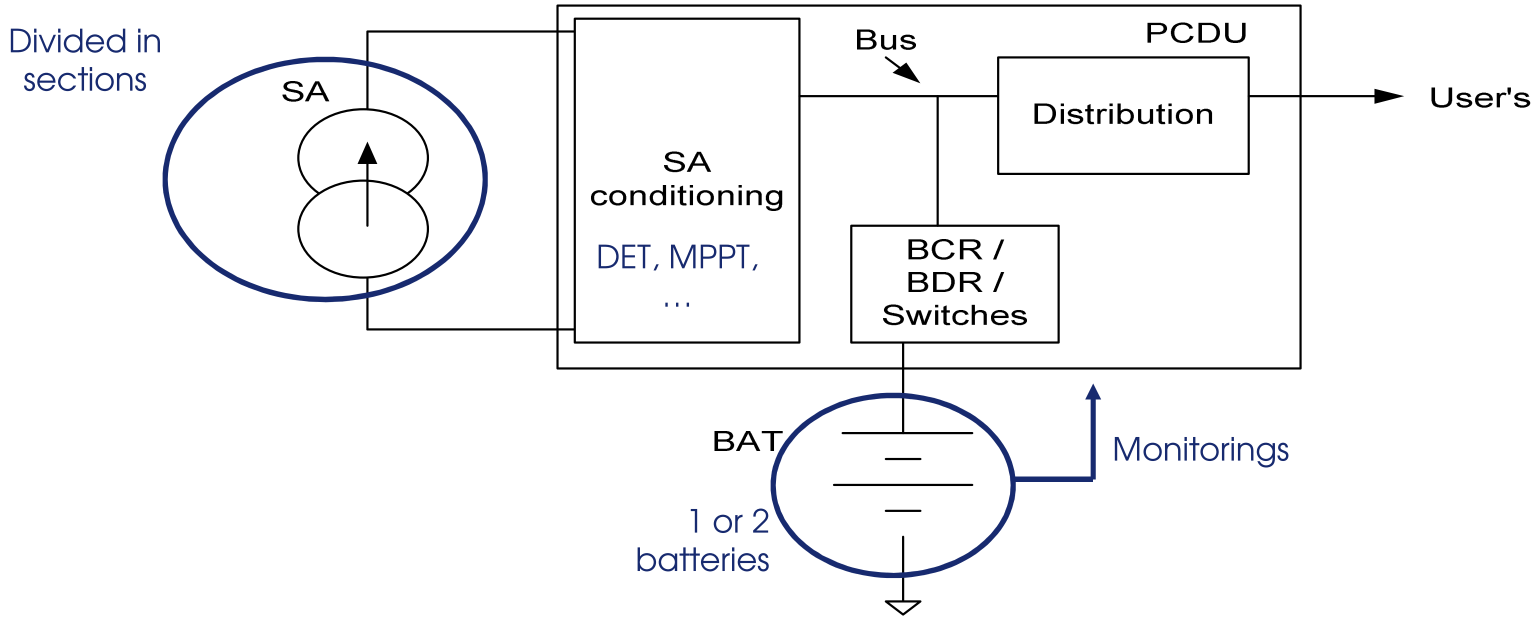
4. POWER MANAGEMENT, CONTROL & DISTRIBUTION

- ARCHITECTURE
- PCU/PCDU

POWER MANAGEMENT, CONTROL & DISTRIBUTION / ARCHITECTURE

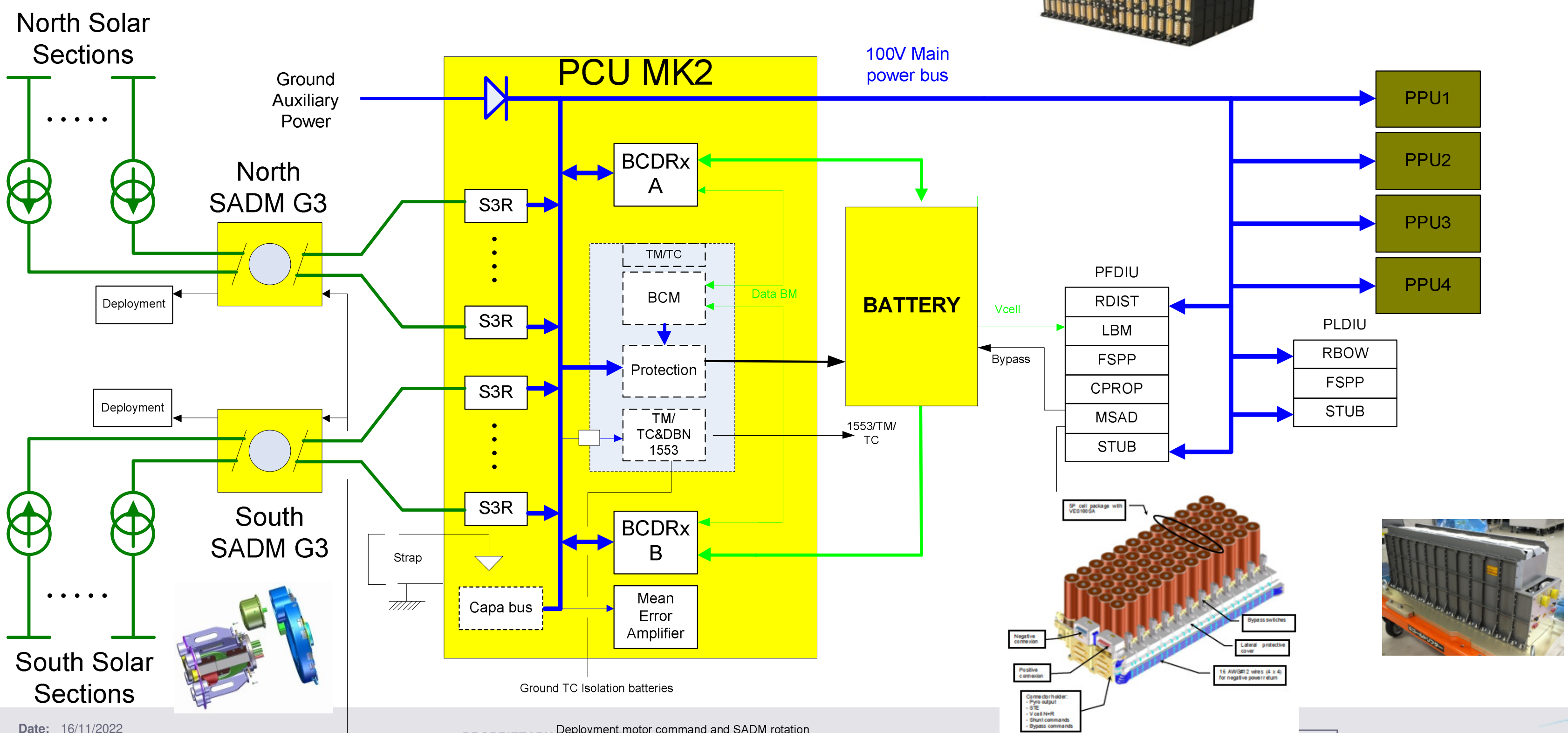
/// Block diagram



-> Operation with primary & secondary power sources whose characteristics are changing with time and conditions of operations

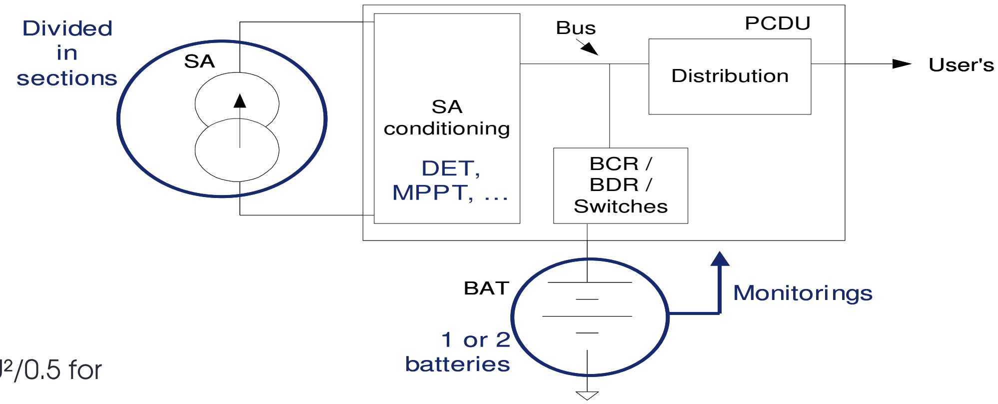
POWER MANAGEMENT, CONTROL & DISTRIBUTION / ARCHITECTURE

/// SpaceBus NEO EPS (9 to 23 KW 100V bus reg)



POWER MANAGEMENT, CONTROL & DISTRIBUTION / ARCHITECTURE

/// Bus voltage selection



/ MANY STANDARDS

- DC: 28V, 50V, 65V, 100V, …
- Even AC busses are used for high power spacecrafts (e.g. ISS)

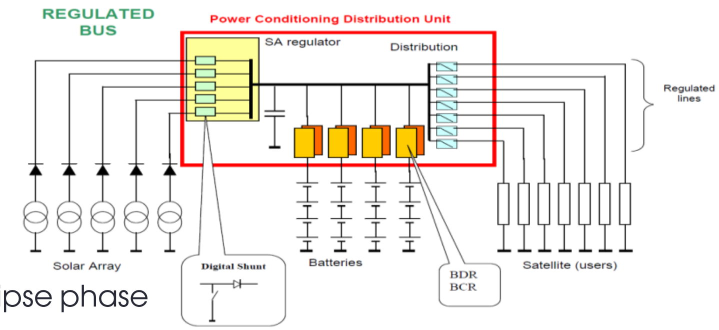
/ CHOICE IS BASED ON

- Bus power
  - Recommended ESA rule: $P < U^2/0.5$ for bus impedance reasons
  - High bus voltage means
    - Less current
    - Simplification of harness
    - « High » voltage management at equipment level (SA, battery, PCDU, …)
- Payload flight heritage

-> SOME ARCHITECTURES MAY EVEN REQUIRES TWO BUSES !!

POWER MANAGEMENT, CONTROL & DISTRIBUTION / ARCHITECTURE

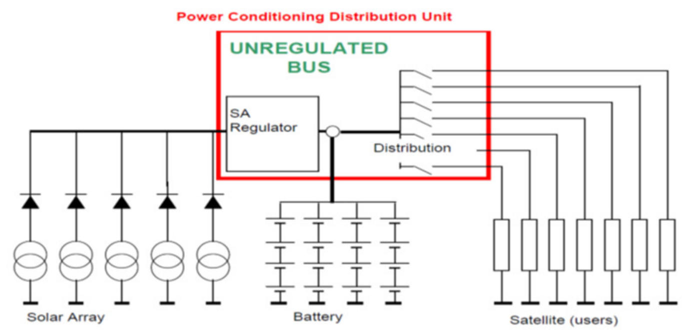
/// Conditioning architecture



/ REGULATED BUS
- Voltage variation is limited to about +/- 1 V whatever the satellite modes
- Need of dedicated electronics to manage the battery charge & discharge
  - Substantial power dissipation inside the PCDU during eclipse phase

/ UNREGULATED BUS
- Bus voltage is imposed by the battery voltage
  - Impact on all DC/DC converters efficiency

/ SEMI-REGULATED BUS
- Regulated bus in sunlight only

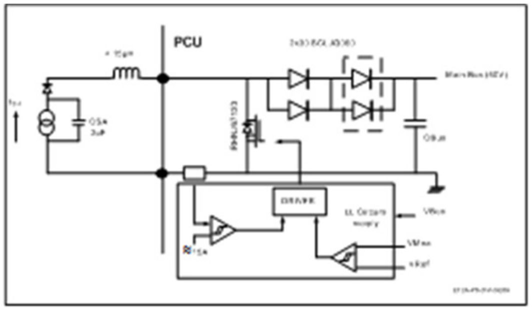


/ CHOICE IS BASED ON
- User's need (mission)
  - Scientific payloads may require regulated bus to fulfill their precisions
  - Thermal stability of some specific loads may requires regulated bus (thermal management is easier in that architecture)
- User's flight heritage

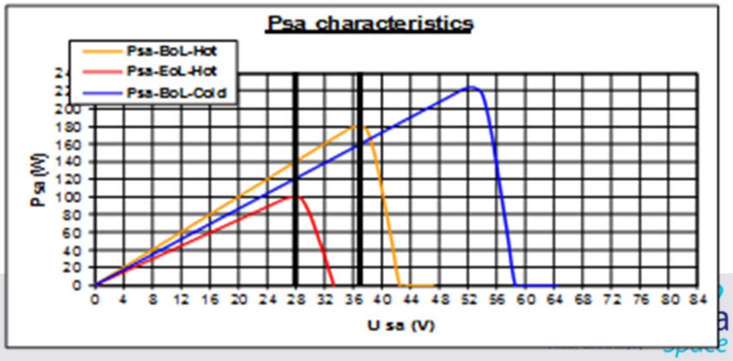
POWER MANAGEMENT, CONTROL & DISTRIBUTION / ARCHITECTURE

/// Conditioning topology

THE CONDITIONING CONCERNS THE WAY THE POWER COMING FROM THE SOLAR ARRAY IS USED TO BE DELIVERED TO THE DIFFERENT USERS OF THE SPACECRAFT, AS WELL AS TO THE BATTERY IN ORDER TO GUARANTEE THE BATTERY RECHARGE

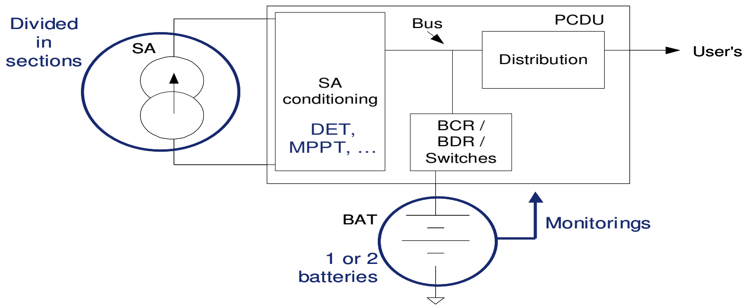


- **DIRECT ENERGY TRANSFER (DET)**
  - DET operates at the bus voltage and extracts the available power from the solar array for this precise voltage
    - Simplest solution
- **PWM CONTROL OF SA**
  - DC/DC converter implemented between SA and bus
- **MAXIMUM POWER POINT TRACKING (MPPT)**
  - MPPT can operate in a wide range of voltages to track the maximum available power from the solar array, converts the (VMP, IMP) into (Vbus, Ibus) and is particularly interesting in case of sensitive flux variations
    - More complex and dissipative solution
- **CHOICE IS BASED ON**
  - Mission & orbit
    - SA flux variation (agility of the S/C, interplanetary missions, …)
    - SA temperature variability
  - …



POWER MANAGEMENT, CONTROL & DISTRIBUTION / ARCHITECTURE

/// Battery management



/ ARCHITECTURE

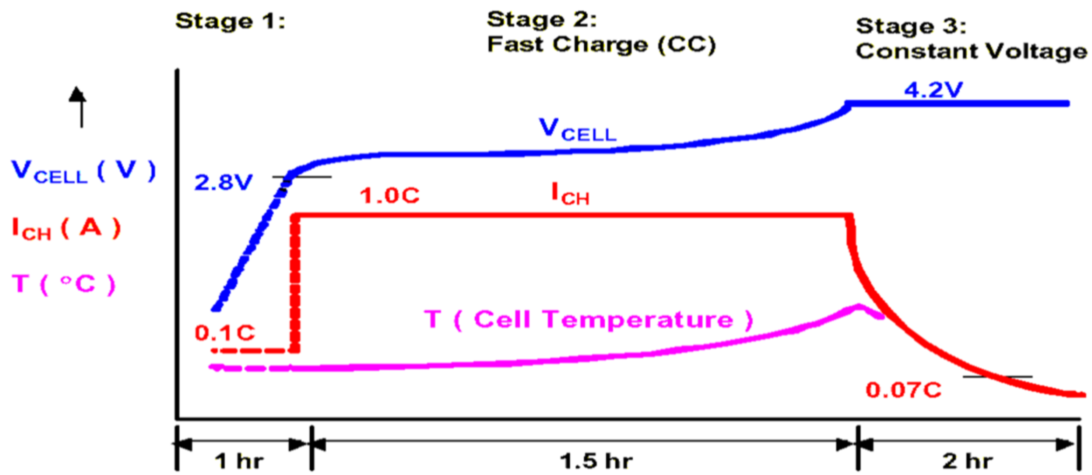
- Centralized
  - Performed by OBC
  - PCDU functions limited to monitoring
- De-centralized at PCDU level
  - Autonomous
    - PCDU ensures battery charge & protections in a reliable way
    - Voltage tapering
    - Protection against over-charge, over-discharge, over-temperature, …
  - Partially autonomous
    - PCDU ensures battery charge & protections
    - OBC is responsible of PCU re-configuration in case of internal failure
- Any intermediate solutions between these two extremes
- Choice based on
  - Price
  - Satellite reliability need
  - …

POWER MANAGEMENT, CONTROL & DISTRIBUTION / ARCHITECTURE

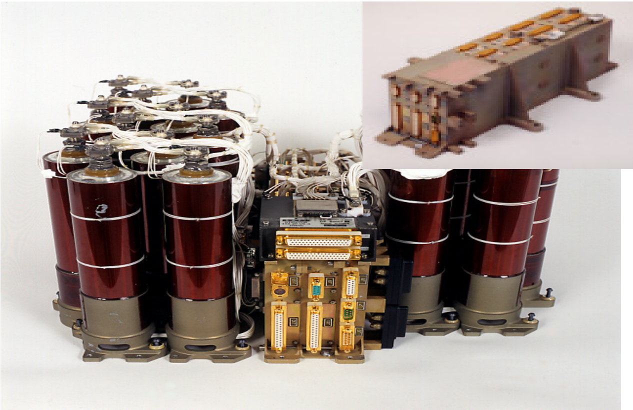
/// Battery management

/ FUNCTIONS

- Voltage / current control during charge
- Current limitation during discharge
- Monitoring
- Protections



POWER MANAGEMENT, CONTROL & DISTRIBUTION / ARCHITECTURE



///Battery failure modes

- **OPEN CIRCUIT**
  - Loss of battery
- **SHORT FAILURE**
  - Degradation of the voltage

/// Battery protections

- **BY-PASS**
  - Actuation of electro-mechanical device allowing to short circuit a failed cell and avoid failure propagation at battery level (arm / fire circuitry)
- **BALANCING (LI-ION)**
  - Voltage balancing at cells level via the actuation of a shunt in parallel with the cell to slightly discharge it to improve battery end-of-charge voltage and increase cell life-time
  - Counterbalancing of cells mismatching
- **MAY BE INTEGRATED AT BATTERY LEVEL OR AT AVIONIC LEVEL (PCDU OR NOT)**
- **SOME LI-ION BATTERIES DO NOT NEED CELLS BALANCING THANKS TO BATTERY INNER PROPERTY**

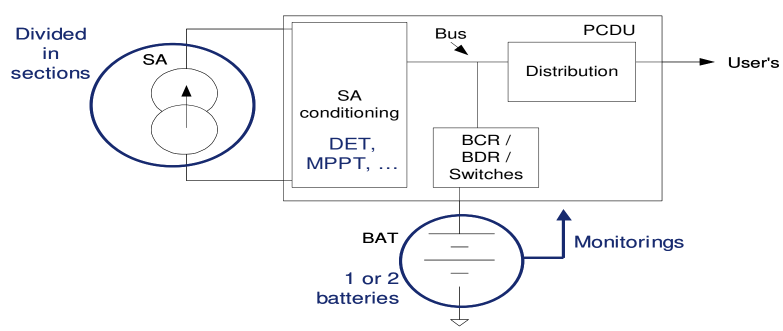
POWER MANAGEMENT, CONTROL & DISTRIBUTION / ARCHITECTURE

/// 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

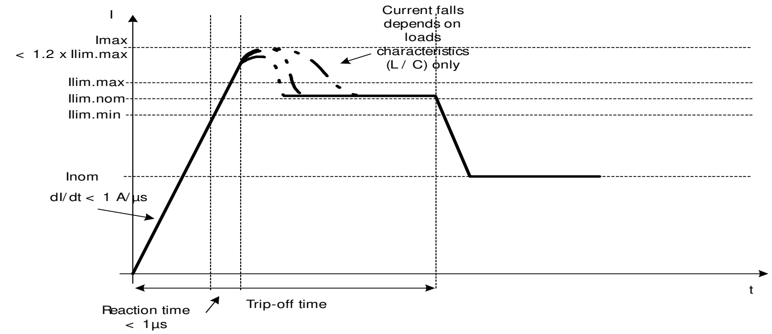


POWER MANAGEMENT, CONTROL & DISTRIBUTION / ARCHITECTURE

///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



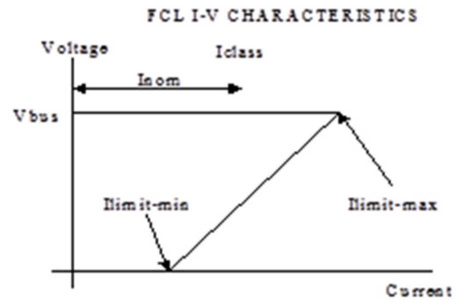
FCL

- Fold-back current limiter
- Essential load (e.g. OBC)
- Limits current at user's switch ON and during short failure (with decreasing level)

PO-LCL

- Permanent-ON LCL
- Essential load (e.g. OBC)
- LCL + automatic periodic re-arming

…



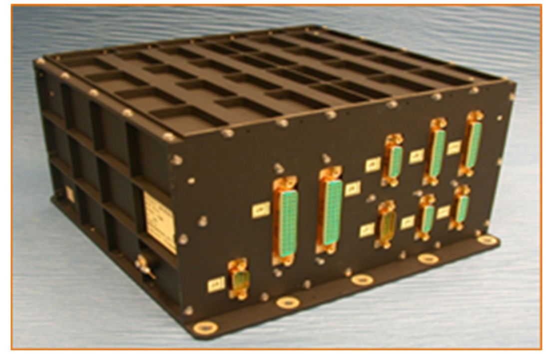
POWER MANAGEMENT, CONTROL & DISTRIBUTION / ARCHITECTURE

///Other constituents of PCDU

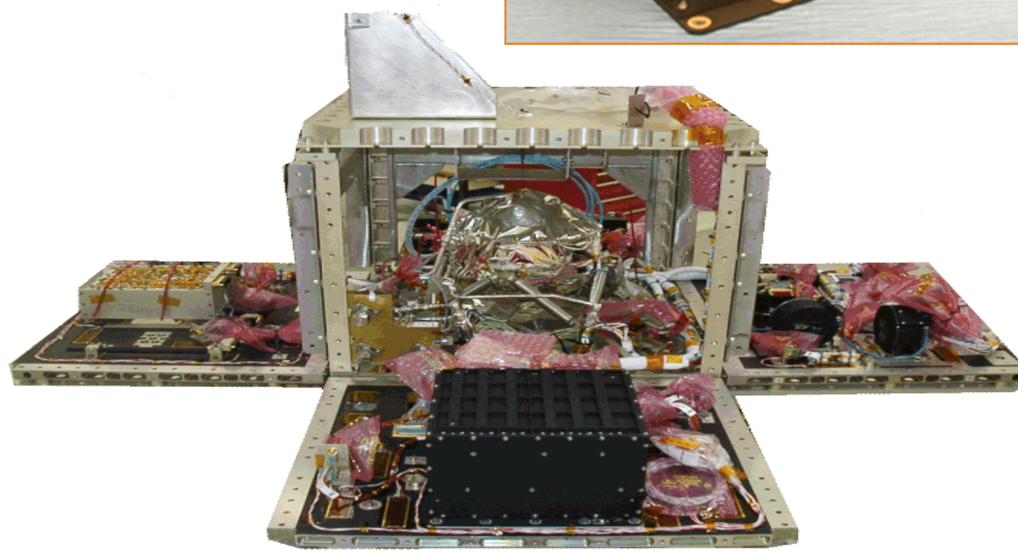
- **COMMAND OF MECHANISMS**
  - SADM motor driver
  - Antenna motor driver
  - …
- **COMMAND ON DEPLOYMENT**
  - Actuation of pyro
  - Actuation of thermal knifes
- **LI-ION BATTERY CELLS MANAGEMENT**
- **ACQUISITION OF THERMISTORS**
- **…**

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

///Examples µSAT

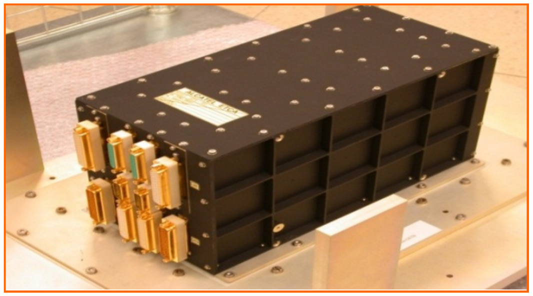


- LOW POWER: 260 W / LOW VOLTAGE : UNREGULATED BUS (22-37 V)
- SOLAR ARRAY REGULATOR: BOOST CONVERTER
- NOT RELIABLE
- DISTRIBUTION FUNCTIONS
  - LCL, Pyro
  - DC/DC for secondary (+5, +-15,+20 V) + LCL p
  - Adaptability of the distribution by paralleling
- CNES/ASTRIUM/TAS-F MYRIADE PLATFORM



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

///Examples Scientific, earth observation & constellations



- 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

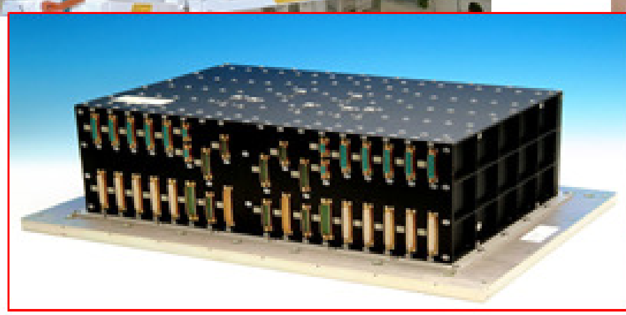
/// Challenges of new constellations

- USE OF COTS (COMPONENT OFF-THE-SHELF) TAKEN FROM AUTOMOTIVE PRODUCT LINES AND TESTED IN RADIATION "A POSTERIORI" – INCLUDING PLASTIC PACKAGE
- USE OF AUTOMOTIVE 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

Illustration CNES



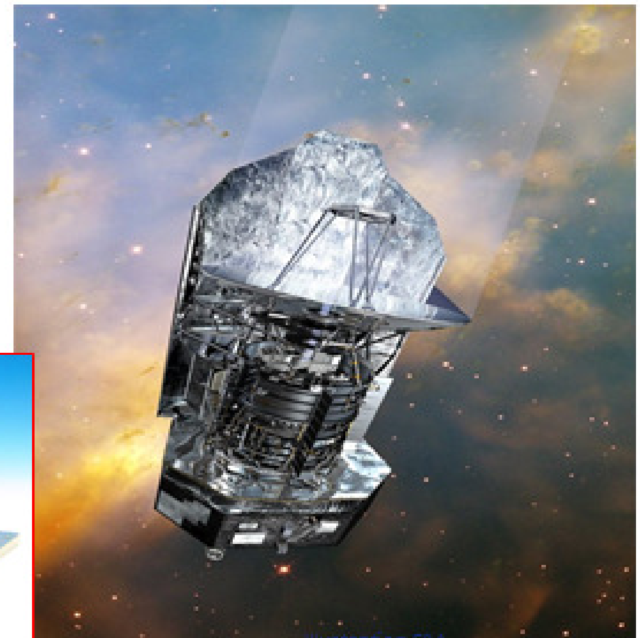


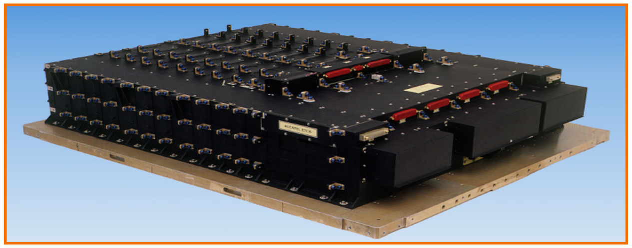
Illustration ESA

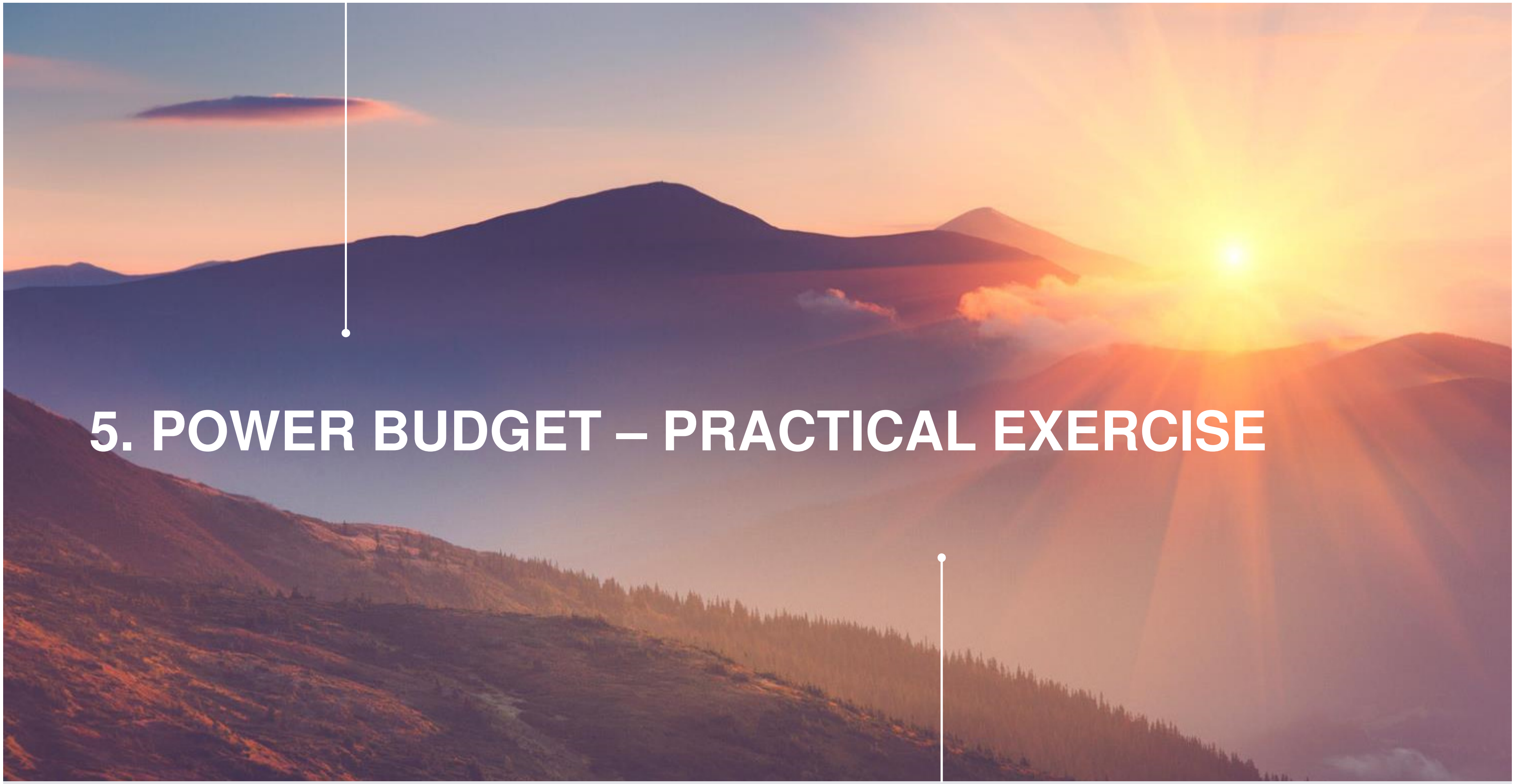
POWER MANAGEMENT, CONTROL & DISTRIBUTION / PC(D)U

/// 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 : 60 PCU'S, 40 IN FLIGHT, 240 YEARS





5. POWER BUDGET – PRACTICAL EXERCISE

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

- **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

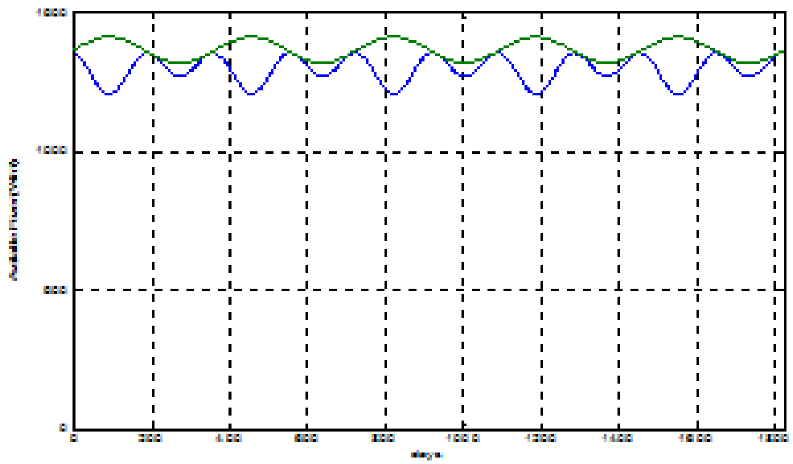
- Polar orbit for best possible coverage worldwide
- Sun-synchronous orbit as other candidate

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

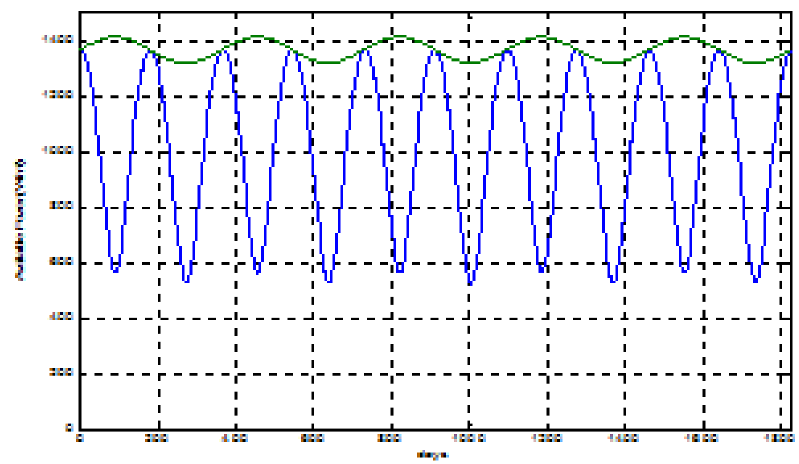
POWER BUDGET

/// 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

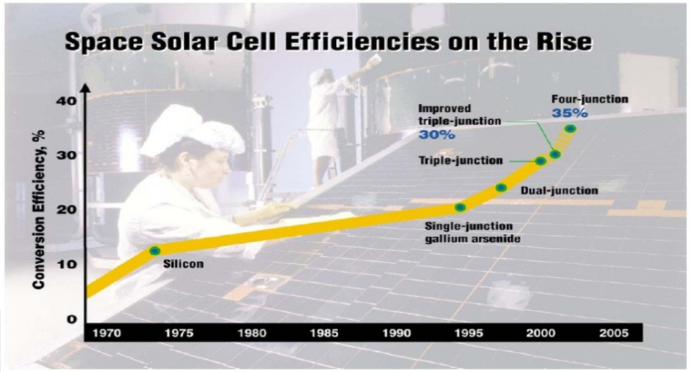
///Orbit selection / Inclination trade-off

- **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: $P < U^2/0.5$ → $U=\mathrm{sqrt}(P*0.5)=\mathrm{sqrt}(1kW*0.5)=22V$
  - High hardware heritage
- **50 V**
  - Reduced current levels
  - Reduced harness & power dissipations

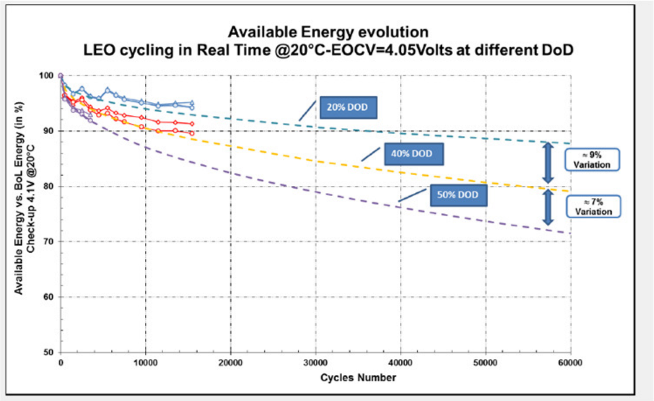
/// EPS sizing: Bus regulation trade-off

- **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%

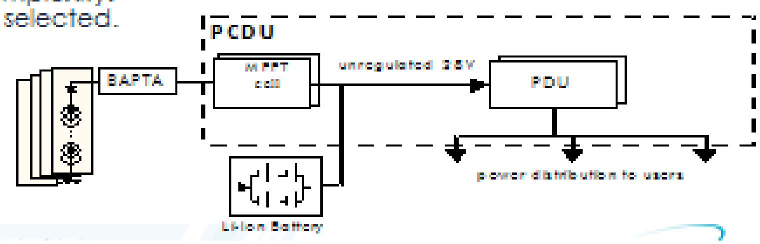
Pecl*0,5 h

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

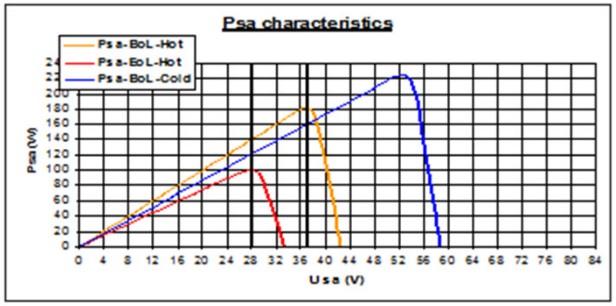
///EPS sizing: Conditioning topology trade-off (Unregulated bus topology)

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

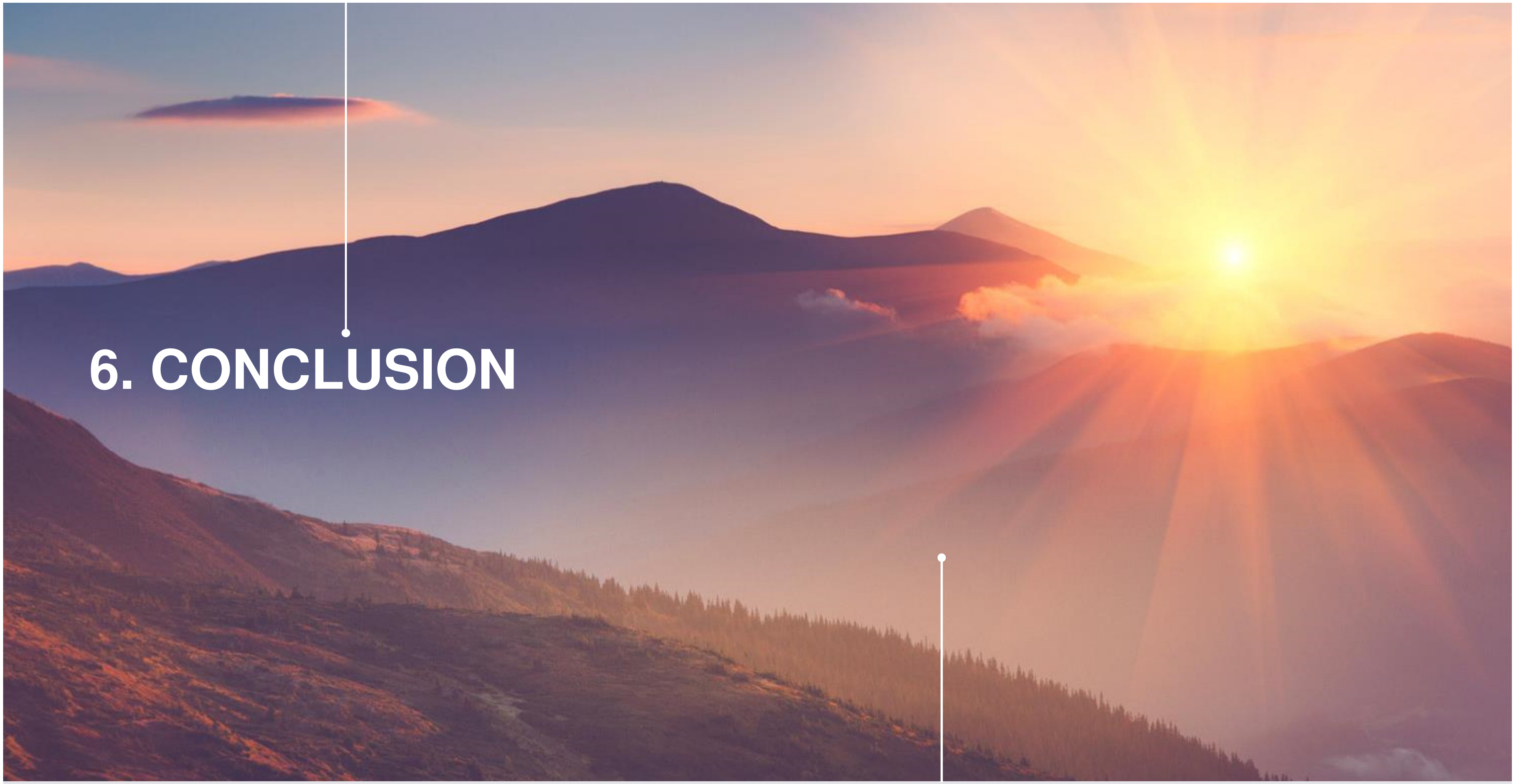
Non-optimization : 5%

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	520
BOL SA cell efficiency	28 %	
EOL/BOL ratio	76.5 %	
Total available SA power (W / m²)	261	111

6. CONCLUSION

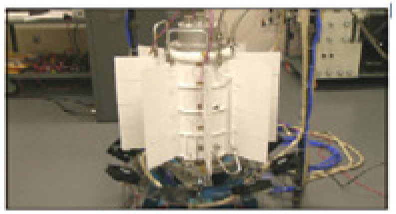


CONCLUSIONS

/// 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 ….

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THANK YOU
FOR YOUR ATTENTION