Satellite Engineering

Overview & Conclusion

Ir. Julien Tallineau
Tel: +32 472 72 32 43
Julien.tallineau@gmail.com

Presentation to ULg (19/12/18)
Satellite Design Lecture

0 TABLE OF CONTENT

1. INTRODUCTION
2. SMALL SATELLITE SECTOR
3. BEGLIAN SATELLITES
4. CASE STUDY
5. CONCLUSION

I INTRODUCTION

1 INTRODUCTION

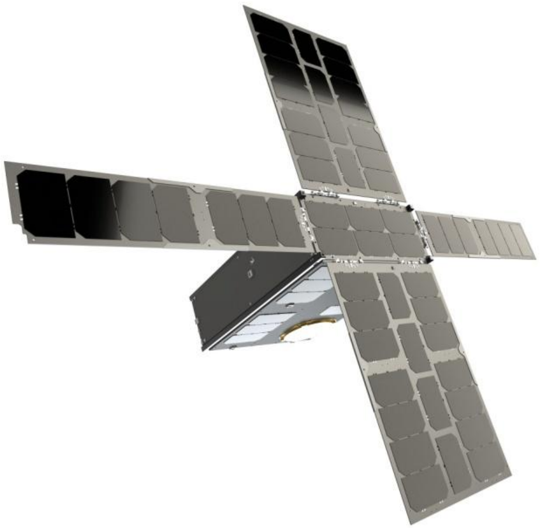
A satellite is a product

sold for a given price to a customer

Products can be classified using their application (Earth Observation, Communication, Navigation), using their mass (1kg, 50kg, 100kg), using their standard (1U, 6U, 12U)

1 INTRODUCTION

- **What is the price of these satellites?**



Earth Observation 6U Cubesat (12kg)

Worldview-4: Earth Observation (2,800 kg)

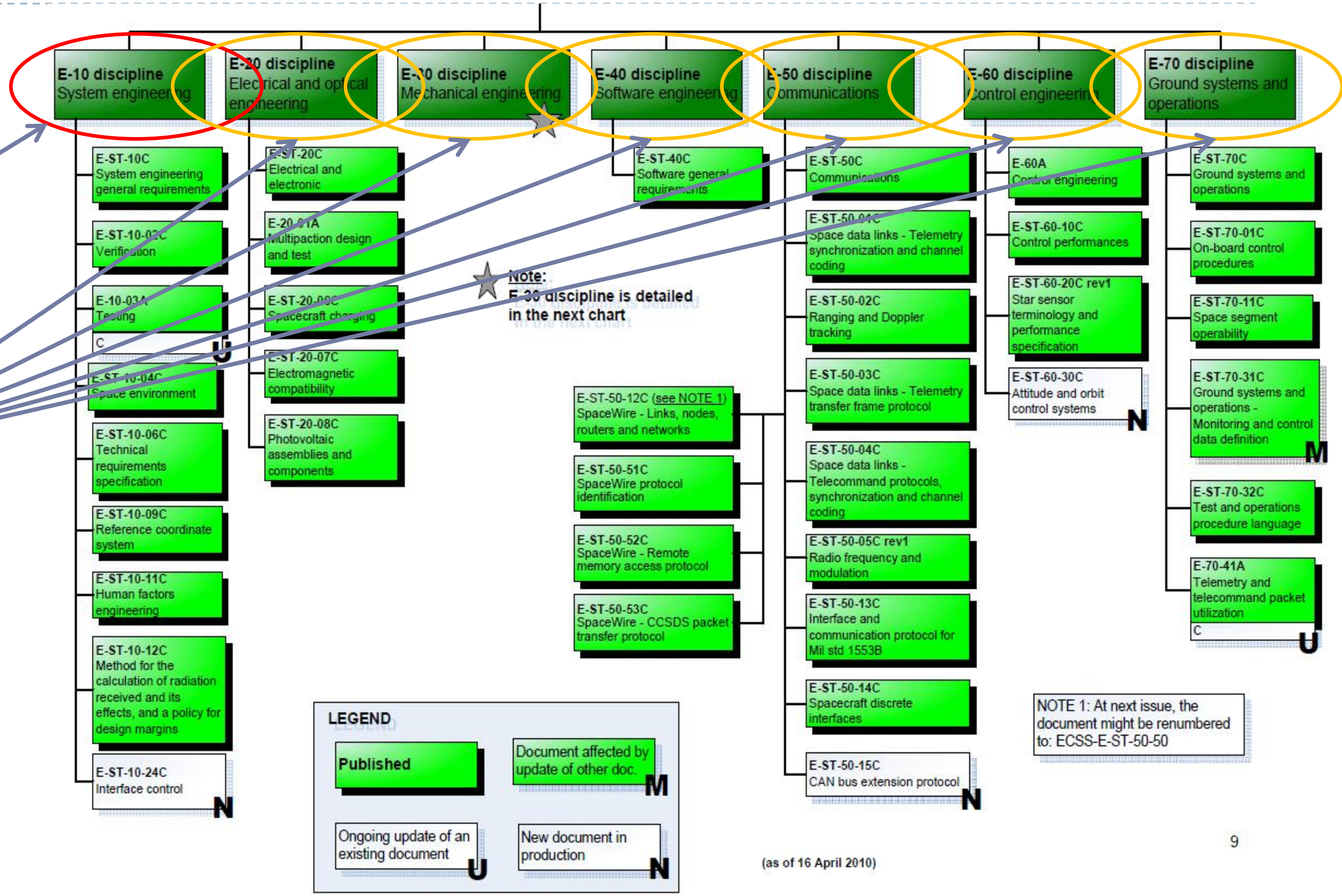
1 INTRODUCTION

- **How much profit do you make?**

1 INTRODUCTION

Mandatory ECSS

Usefull ECSS



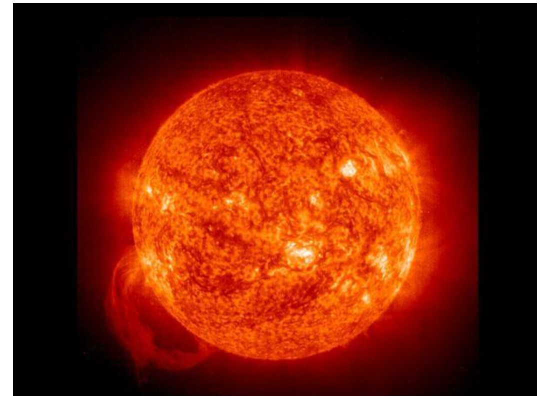
1 INTRODUCTION

- **Satellite Engineering**

1. Customer Need (Scientist)
2. Creation of System Requirements
3. Phase 0 (CDF Study)
4. Phase A (Feasibility Study)
5. Phase B (Preliminary Design)
6. Phase C (Final Design)
7. Phase D (Manufacturing / Testing)
8. Phase E (Launch / Commissionning & Operations)
9. Phase F (De-orbiting)

1 INTRODUCTION

- **Satellite Engineering**

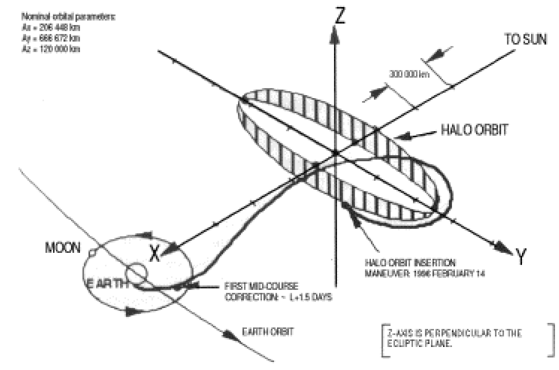


1. Customer Need (Scientist)
2. Creation of System Requirements
3. Phase 0 (CDF Study)
4. Phase A (Feasibility Study)

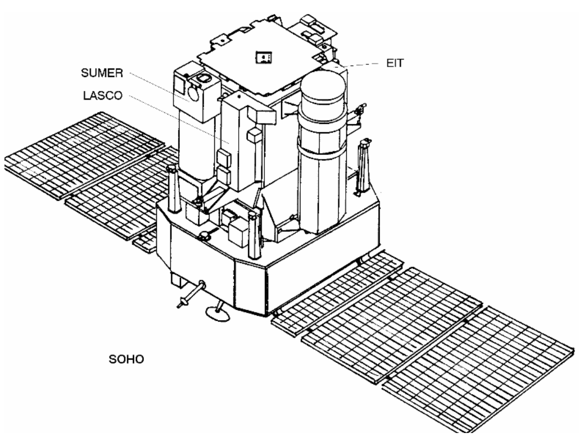
1 INTRODUCTION

- **Satellite Engineering**

5. Phase B (Preliminary Design)
   - Detailed System Analysis
   - Preliminary Subsystem Analysis
   - Trade-offs



6. Phase C (Final Design)
   - Detailed Subsystem Analysis
   - Procurement
   - Qualification Testing



1 INTRODUCTION

- **Satellite Engineering**

7. Phase D
   - Manufacturing
   - Acceptance Testing
   - Requirement Verification
   - Shipment to Launch site

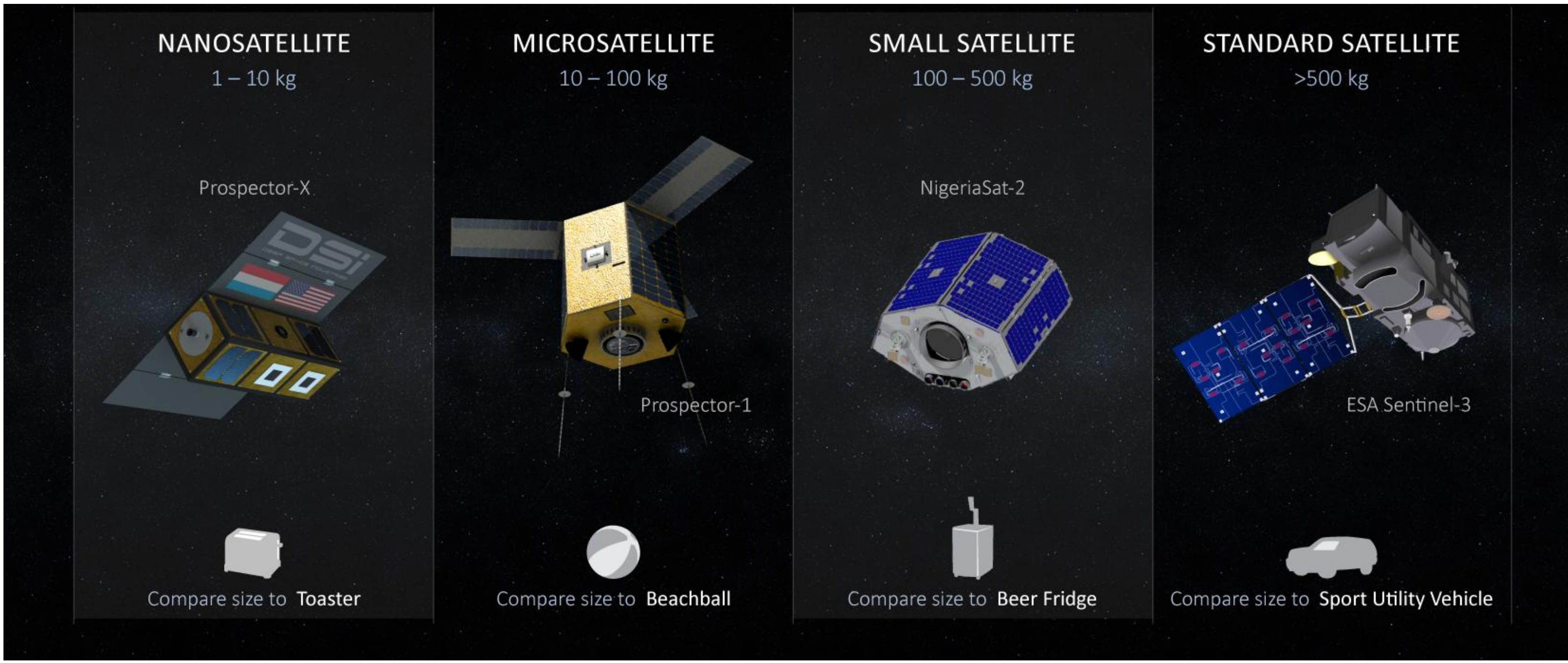
1 INTRODUCTION

- **Satellite Engineering**

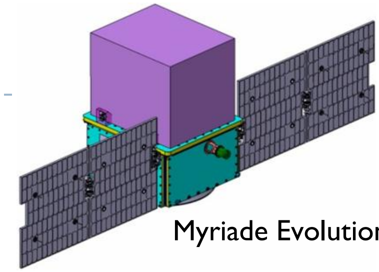
8. Phase E
   - Launch
   - Commissionning
   - Operations

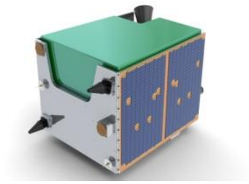
9. Phase F (De-orbiting/End of Life)
   - None in this case

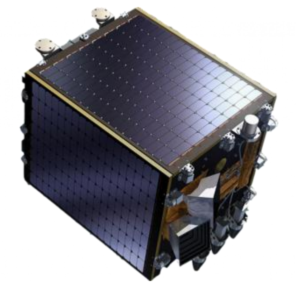
2 SMALL SATELLITE SECTOR



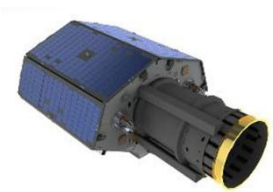
2 MAIN PRODUCTS

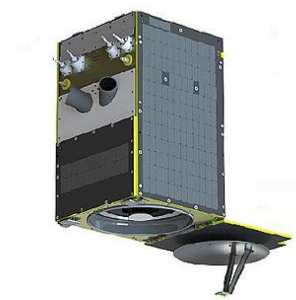

Myriade Evolution


SSTL-150


PROBA

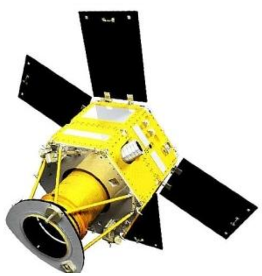

Skysat-1


SSTL-300S

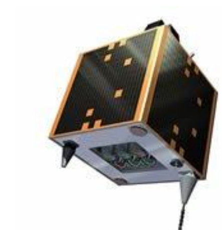

Skysat-3


Satellogic-1


SpaceEye-X

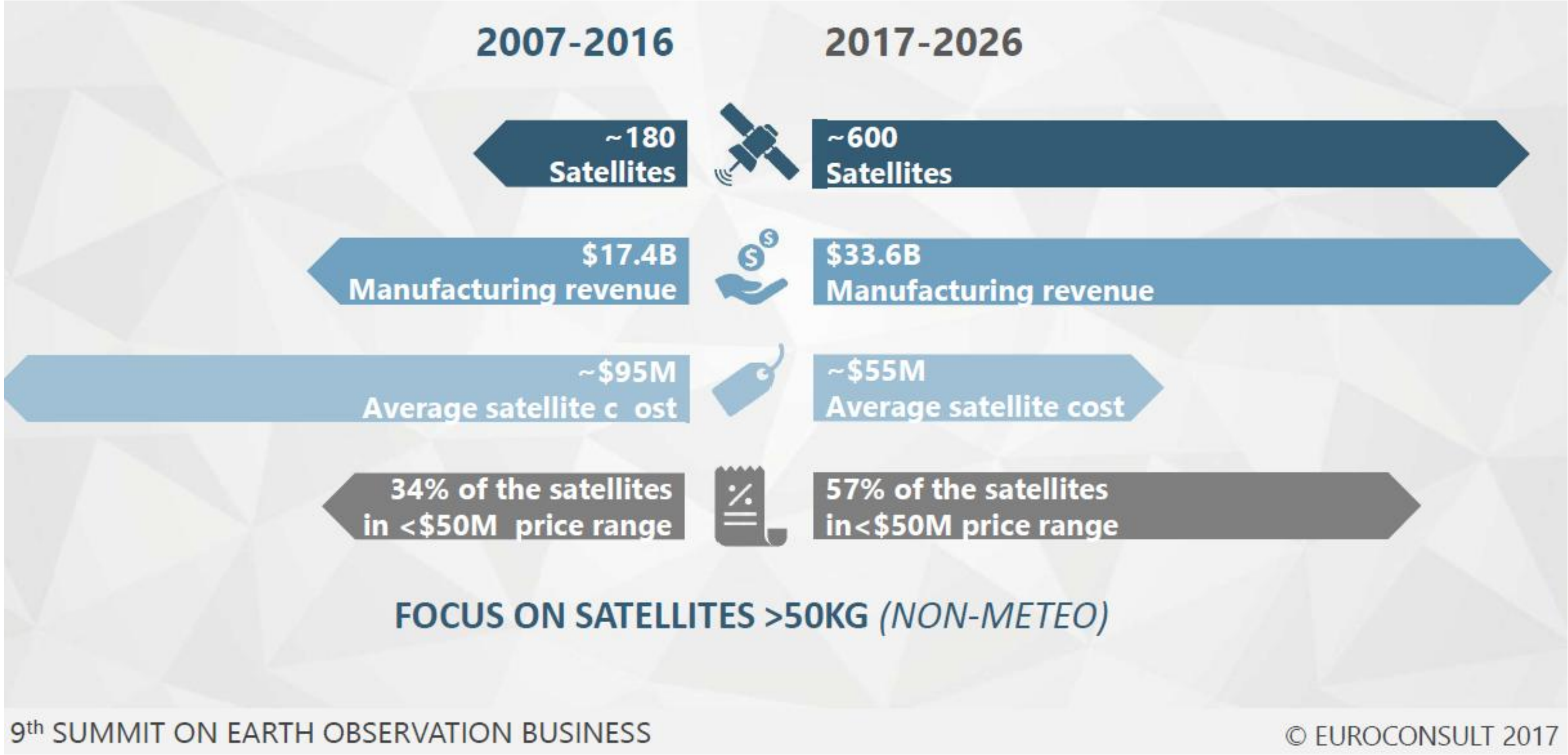

SpaceEye-D


12U Cubesat


SSTL-50 Platform


3U Cubesat

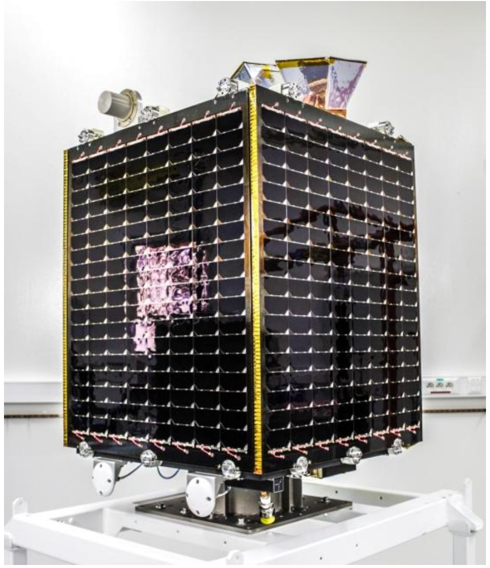
2 MARKET MAP


2007-2016
2017-2026
~180 Satellites
~600 Satellites
$17.4B Manufacturing revenue
$33.6B Manufacturing revenue
~$95M Average satellite c ost
~$55M Average satellite cost
34% of the satellites in <$50M price range
57% of the satellites in<$50M price range
FOCUS ON SATELLITES >50KG (NON-METEO)
9th SUMMIT ON EARTH OBSERVATION BUSINESS
© EUROCONSULT 2017

2 MARKET MAP


POTENTIAL DEPLOYMENT OVER 2017-2021
Earth observation (incl. meteo)
350-400
>100
~75
~40
~165
Of which 88% for 5 commercial constellations
Data collection (IoT/M2M/AIS/ADS-B etc.)
~150
81 of Iridium NEXT satellites equipped with Aireon ADS-B payload - 20 already launched
<10kg
10-50kg
51-250kg
251-500kg
>501kg
9th SUMMIT ON EARTH OBSERVATION BUSINESS
© EUROCONSULT 2017

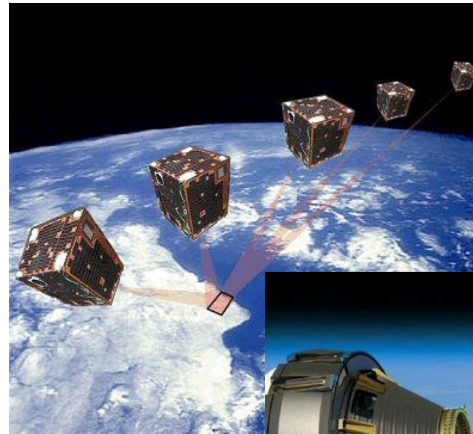
- **3 BELGIAN SATELLITES**



3 ACHIEVEMENTS

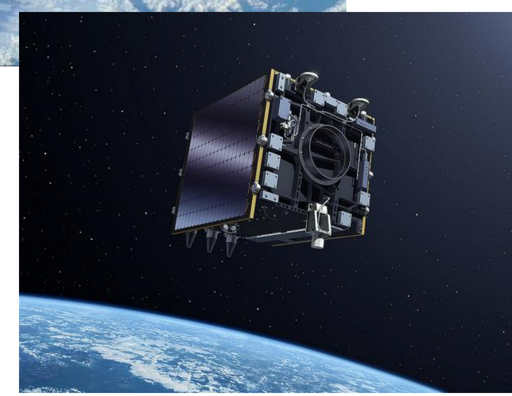
QinetiQ Space nv

- **PROBA-1 (2001)**
  - Earth Imaging
  - Technology Demo



- **PROBA-2 (2009)**
  - Sun Observation
  - Technology Demo

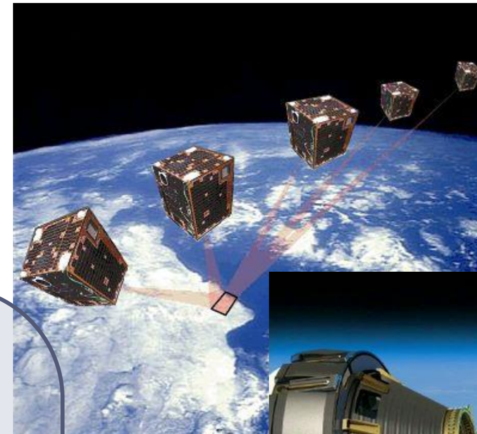
- **PROBA-V (2013)**
  - Global Vegetation Monitoring
  - Operational Mission



3 ACHIEVEMENTS

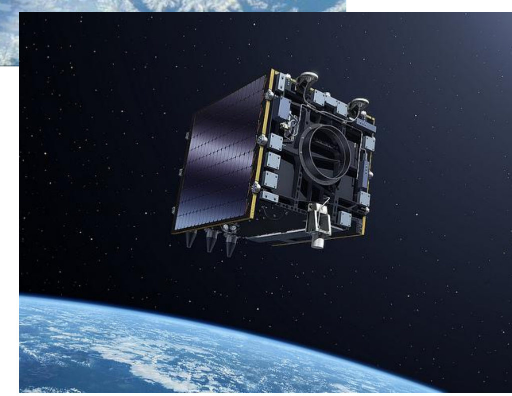
QinetiQ Space nv

- **PROBA-1 (2001)**
  - Earth Imaging
  - Technology Demo

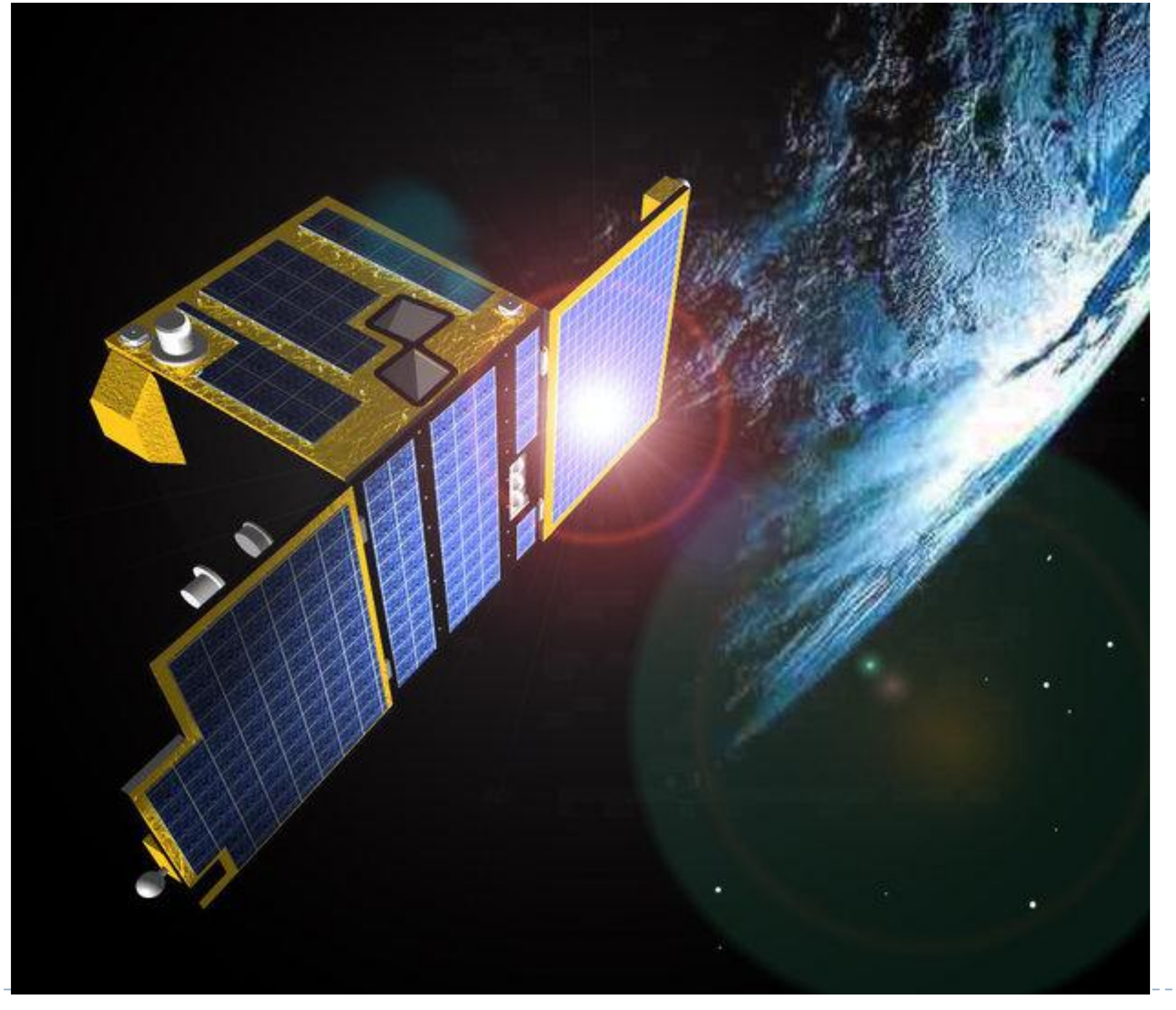


- **PROBA-2 (2009)**
  - Sun Observation
  - Technology Demo

- **PROBA-V (2013)**
  - Global Vegetation Monitoring
  - Operational Mission



3 PROBA - 2



3 PROBA - 2

- **Mission**

1. In orbit Demonstration, PROBA-2 aimed at **technological innovation**.

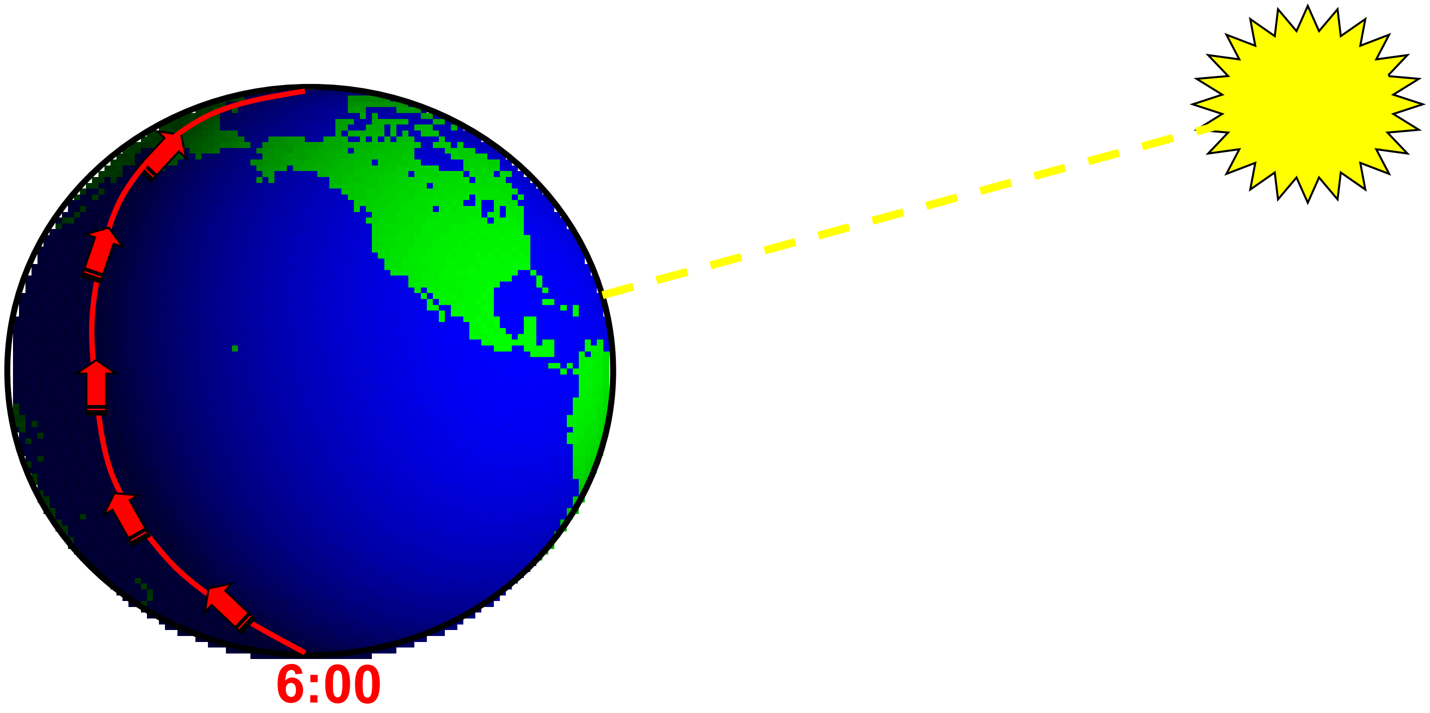
   Altogether, 17 new technological developments and four scientific experiments are being flown on Proba-2.

Orbital Parameter	
LTAN	06:00 (AM)
a (km)	7100
e (deg)	0
i (deg)	98
w	0

3 PROBA - 2

- **Mission**

1. RAAN selected for 6:00 AM Local Time Ascending Node

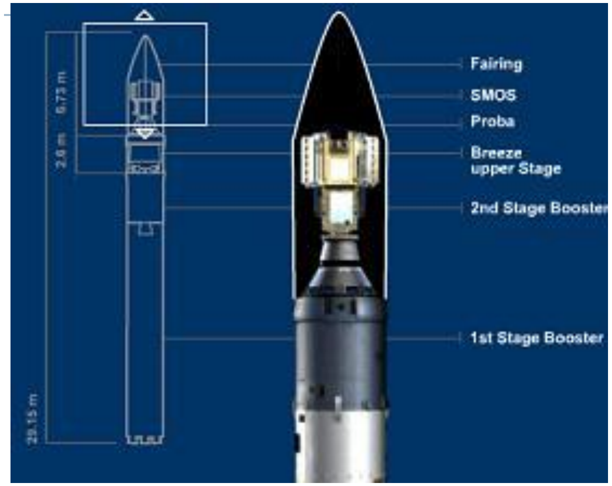


3 PROBA - 2

- **Mission**

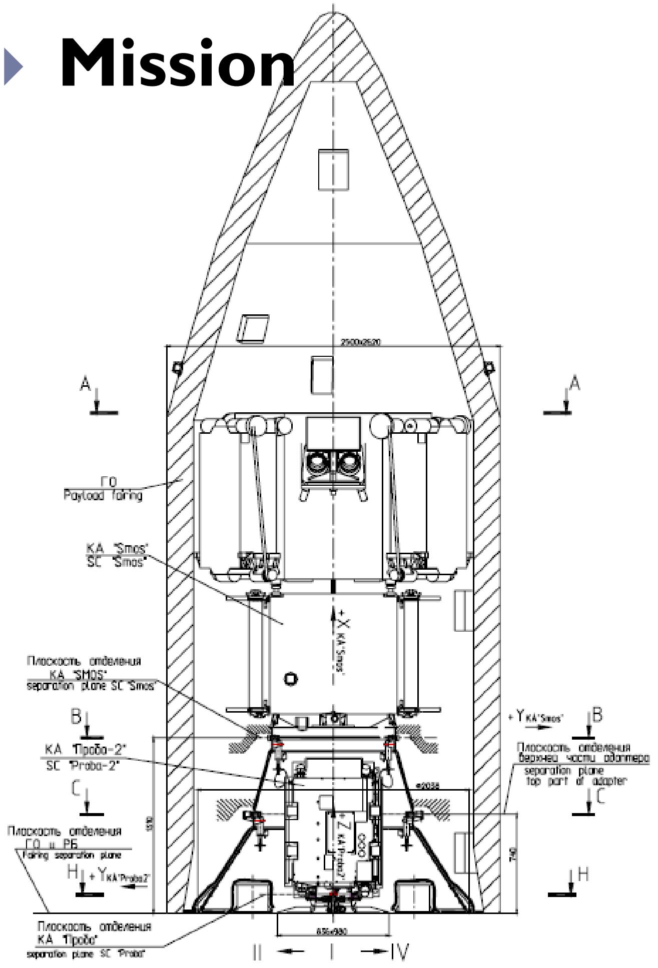
1. Launcher is **Rockot**
   - <u>Worst Case</u> separation rate of 8° per sec.
   - Inclination accuracy of 0.05°
   - Altitude accuracy of 12km
   - RAAN accuracy of 3.75° (≈15min LT)

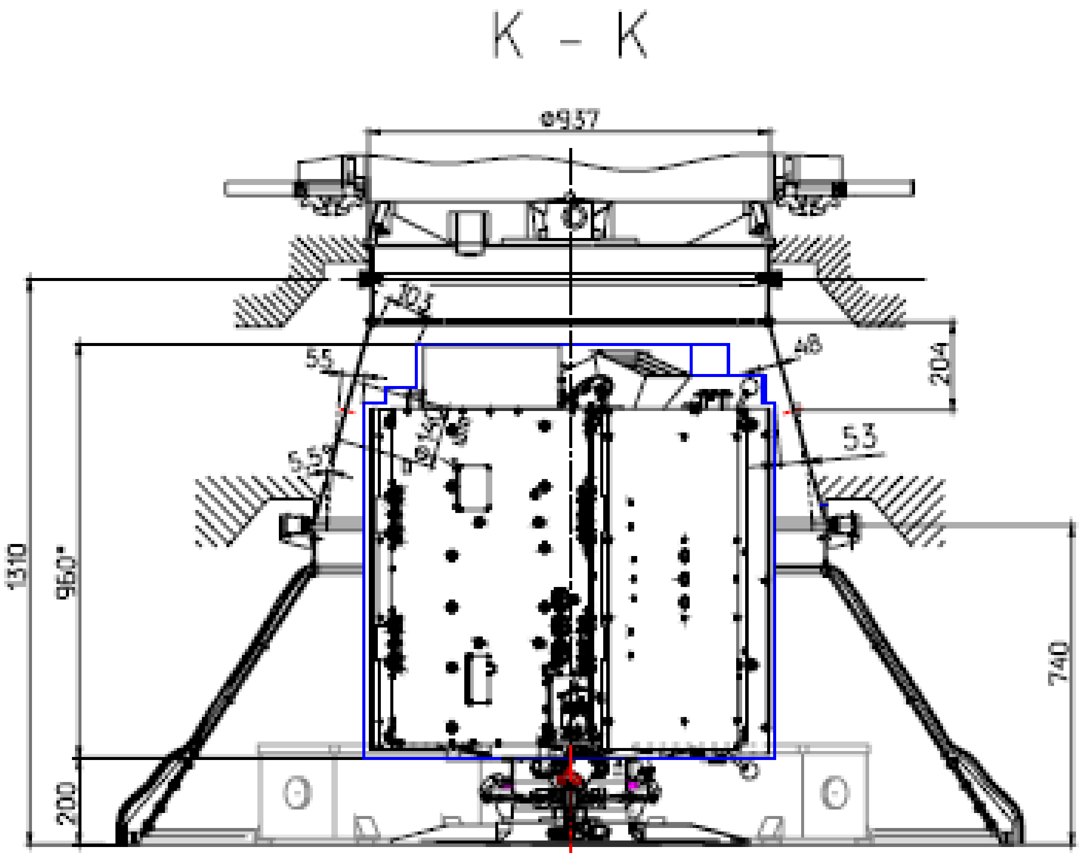
2. Injected via the **Breeze upper stage**



3 PROBA - 2

- **Mission**

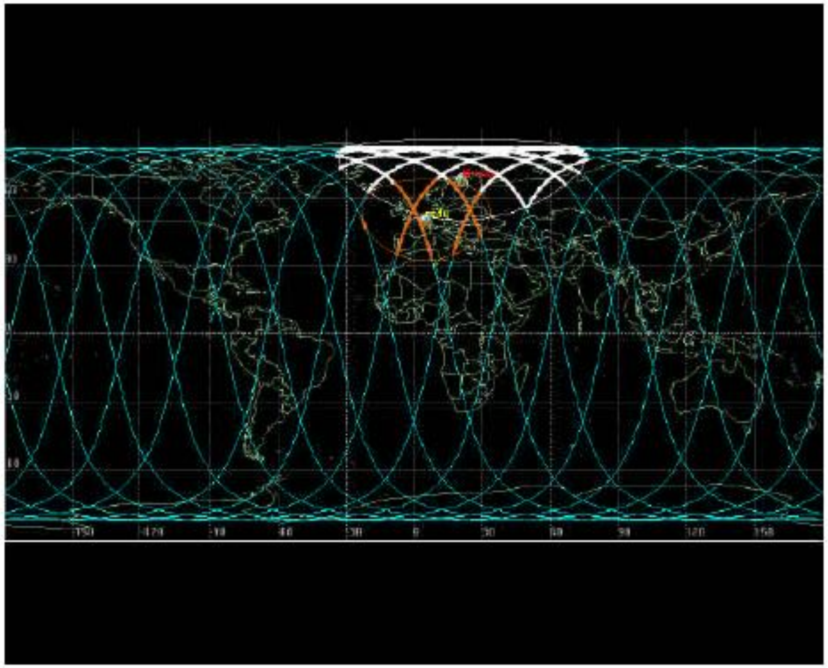




3 PROBA - 2

- **Mission**

1. Ground segment visibility
   - REDU
   - KIROUNA

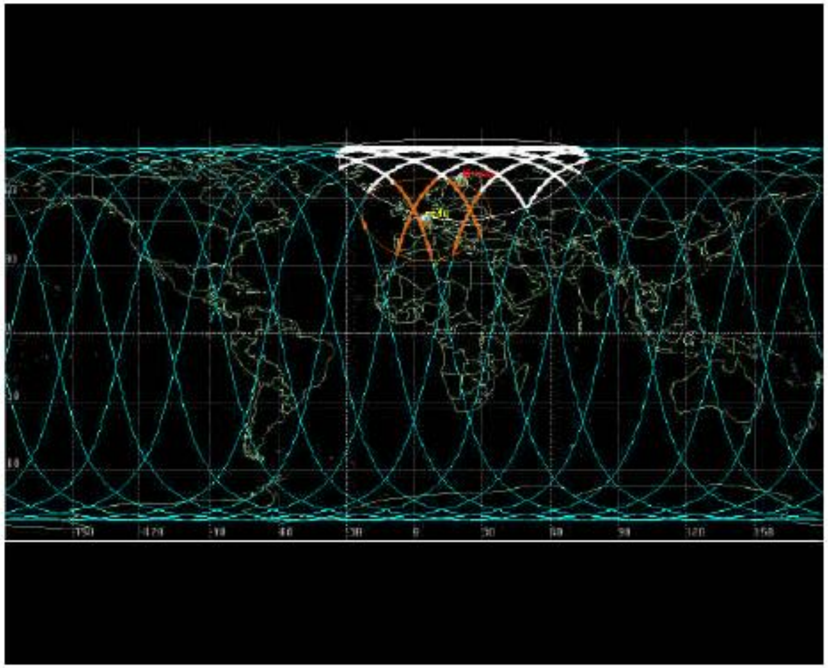


	Mean # contact	Mean duration of contact	Mean contact per day	Gap
Redu	5	8	35	11.5 hours
Kiruna	**+/ - / = ?**	**+/ - / = ?**	**+/ - / = ?**	**+/ - / = ?**

3 PROBA - 2

- **Mission**

1. Ground segment visibility
   - REDU
   - KIROUNA



	Mean # contact	Mean duration of contact	Mean contact per day	Gap
Redu	5	8	35	11.5 hours
Kiruna	**9.5**	**8**	**77**	**10 hours**

3 PROBA - 2

- **Mission**

1. Scenario
   - LEOP
   - Commissioning (three months)
   - Nominal Operations

2. Spacecraft Modes
   - Separation
   - Safe
   - Imaging
   - Stand-by

3 PROBA - 2

- **Satellite Design - Configuration**

1. Single H Structure
2. Sun Shield
   - Standard STR
   - Bepi-Colombo STR
3. High Unit Density
4. Deployable Solar Panel (x2)

3 PROBA - 2

- **Satellite Design - Mechanics**

1. Spacecraft Mass ≈ 120 kg

2. CoG Choice
   - Folded Configuration (LV requirement)
   - Deployed Configuration (GNC requirement)

	X	Y	Z
COG (mm)	**< 5**	**< 5**	< 400
MOI			
X	<15	-1<XY< 1	-1<XZ< 1
Y	-1<YX< 1	<15	-1<YZ< 1
Z	-1<ZX< 1	-1<ZY< 1	<8

	X	Y	Z
COG (mm)	**< 20**	< 5	< 400
MOI			
X	<15	-1<XY< 1	-1<XZ< 1
Y	-1<YX< 1	<15	-1<YZ< 1
Z	-1<ZX< 1	-1<ZY< 1	**<10**

3 PROBA - 2

Satellite Design - Power

1. Power budget is positive, independently of the mode
   - Observation (w/o TX)
   - Observation with TX
   - Safe mode with TX (worst Beta-angle).

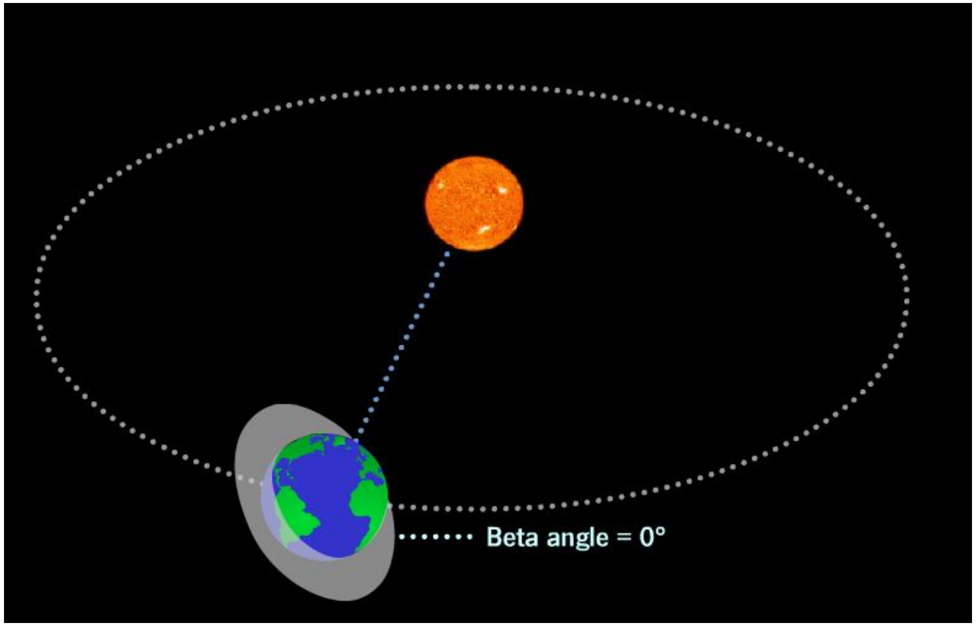
Definition

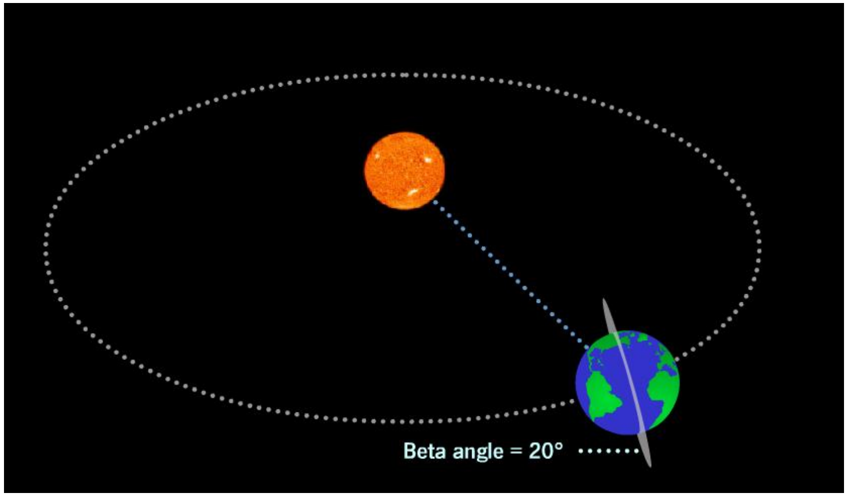
Beta Angle = angle between the Sun-Earth vector and the orbital plane.

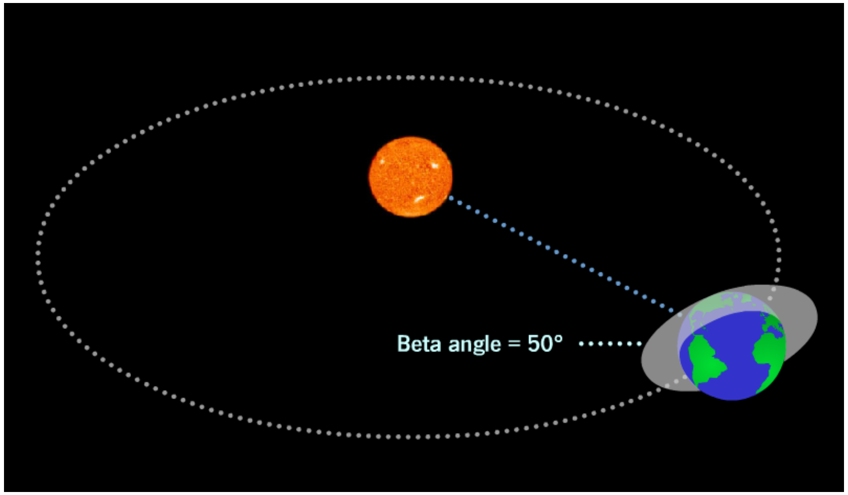
- *β = 90° implies max sunlight time*
- *β = 0° implies min sunlight time*

3 PROBA - 2

- **Satellite Design - Power**



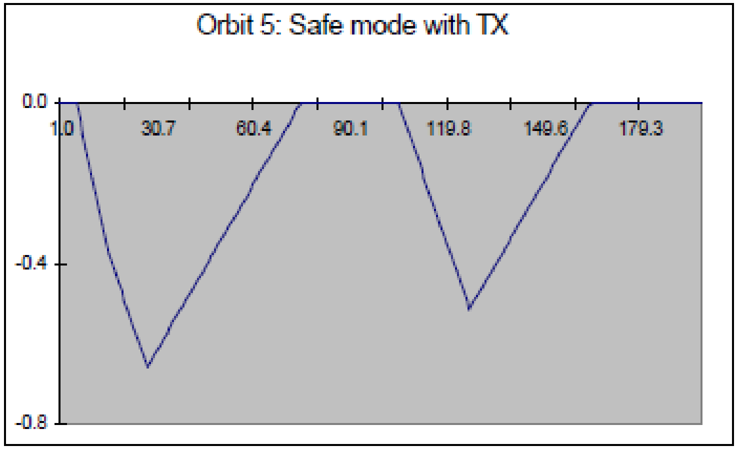


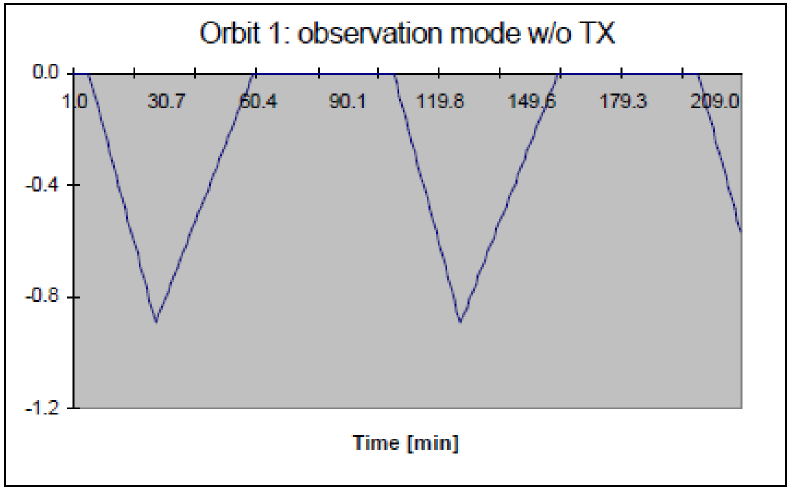


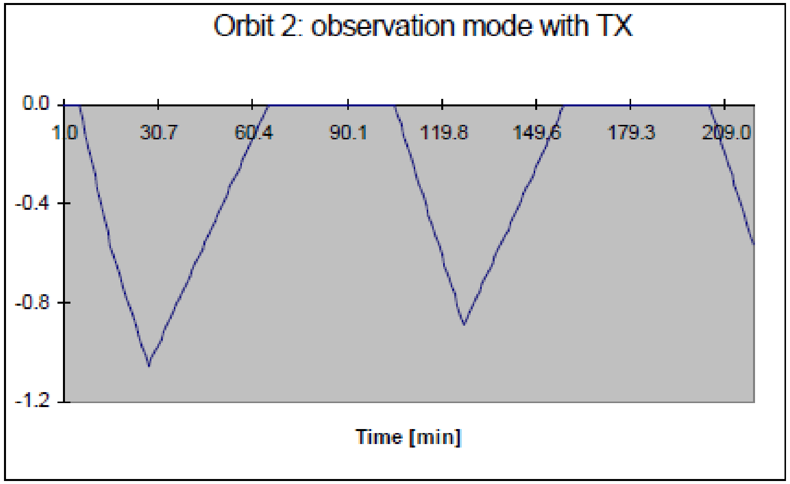
3 PROBA - 2

- **Satellite Design – Power**

1. Battery DoD (Ah) in function of time
2. Non Regulated bus 28V







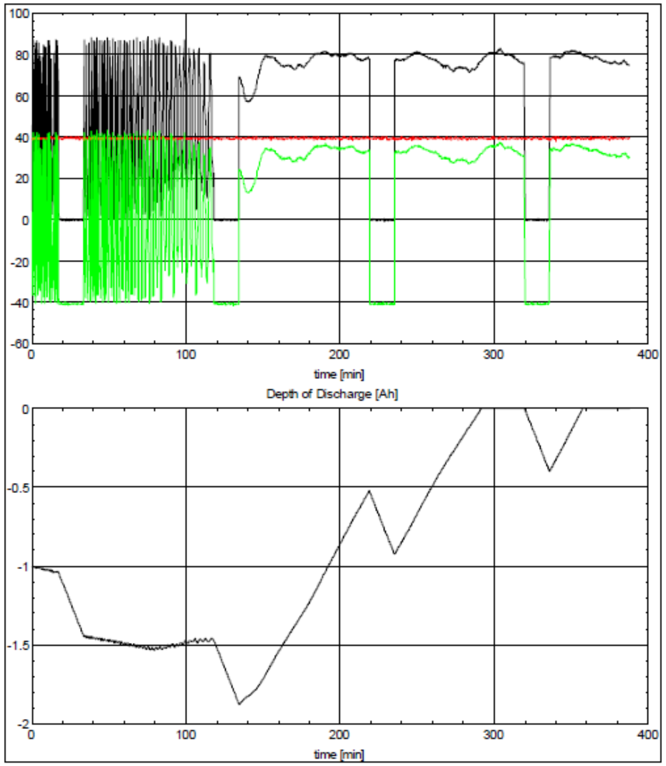
3 PROBA - 2

- **Satellite Design - Power**

1. Power budget while de-tumbling !

2. Trade-off between:
   - Performance (GNC)
   - Time (LEOP schedule)
   - Battery Discharge (Higher DoD)

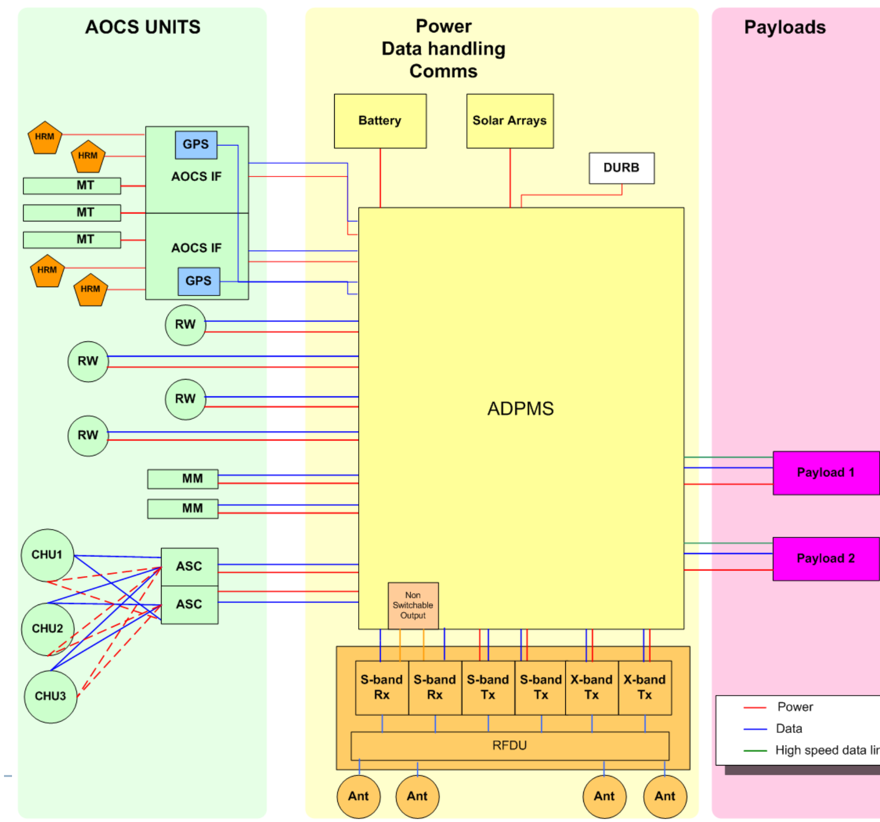
Top: Incoming power (black), power consumption (red, and power to battery (green) [W]; Bottom: battery DoD [Ah]



3 PROBA - 2

- **Satellite Design - Avionics**

1. System is **fully redundant**
2. Data & Power centralized (**ADPMS**)
3. Interface Unit
   - AOCS Module
   - Deployment Module
   - Propulsion Module
   - Thermal Control Module

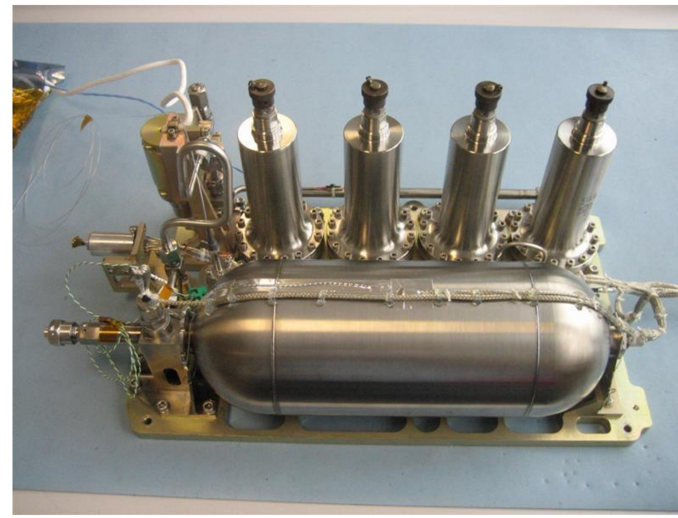


Source: eoportal

3 PROBA - 2

- **Satellite Design – AOCS**

1. Low power resistojet (Xenon)
   - 15W for heater (x2)
   - 50s Isp (min)
   - 20mN Thrust
   - Total $\Delta V = 2m/s$

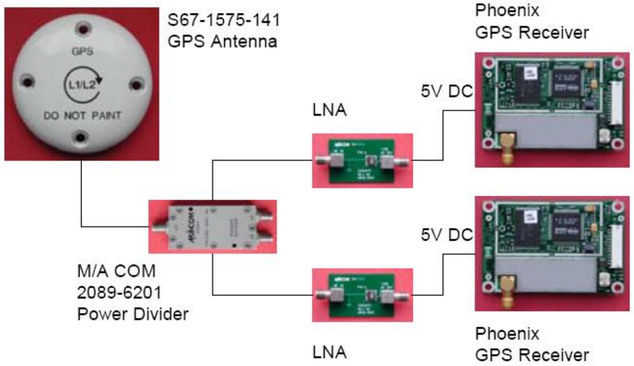


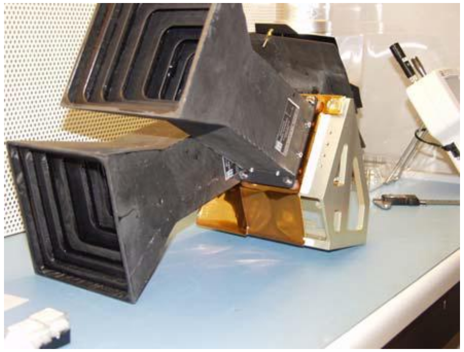
$$\Delta u = V_{eq} \ln\left(\frac{mf}{me}\right) = V_{eq} \ln MR = Isp\ g_o \ln MR$$

3 PROBA - 2

- **Satellite Design – AOCS**

1. Sensors
   - 2 Star-tracker
   - 2 GPS RX
   - 2 Magnetor-Meter

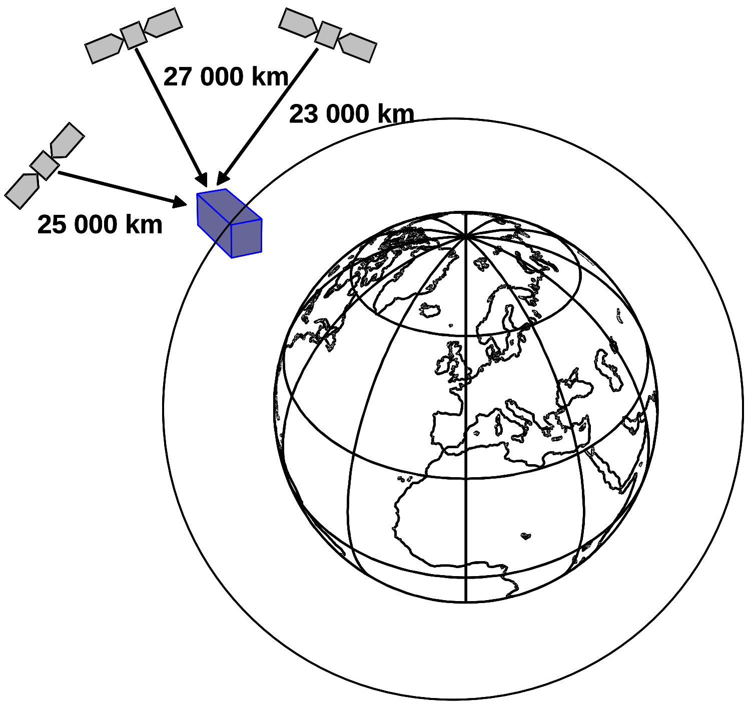


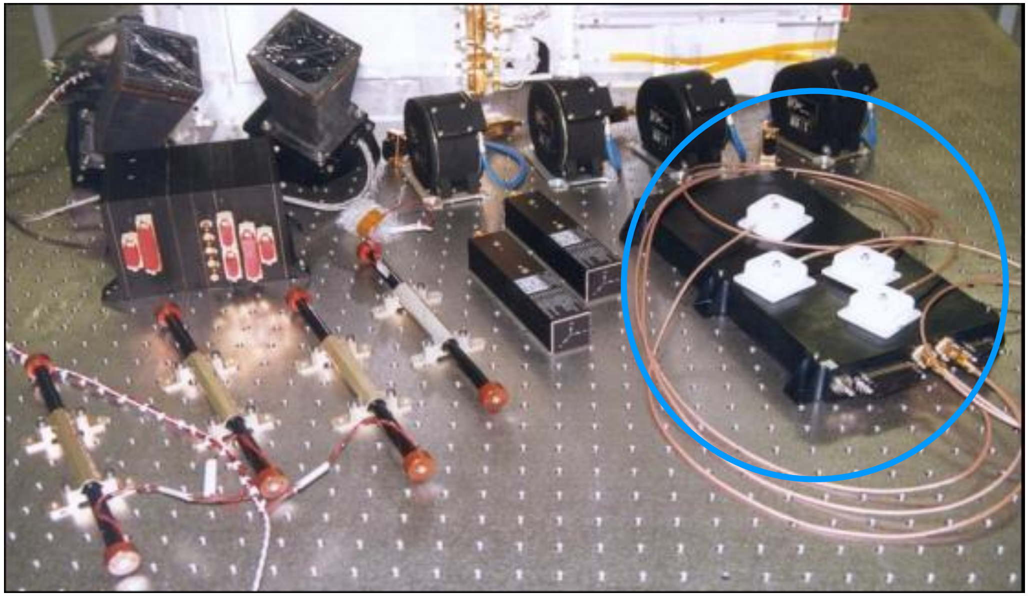


2. Actuators
   - 4 Reaction Wheels
   - 3 dual-coil magneto-Torquer

3 PROBA - 2

GPS RX



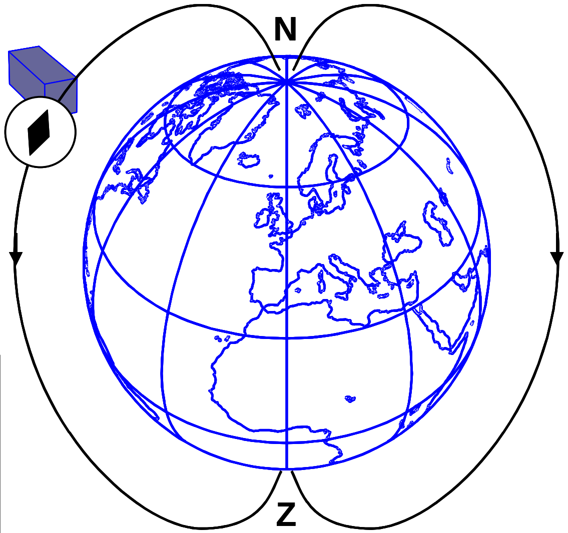


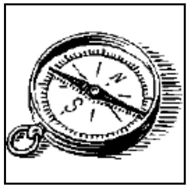
⇒ **Position determination**

3 PROBA - 2

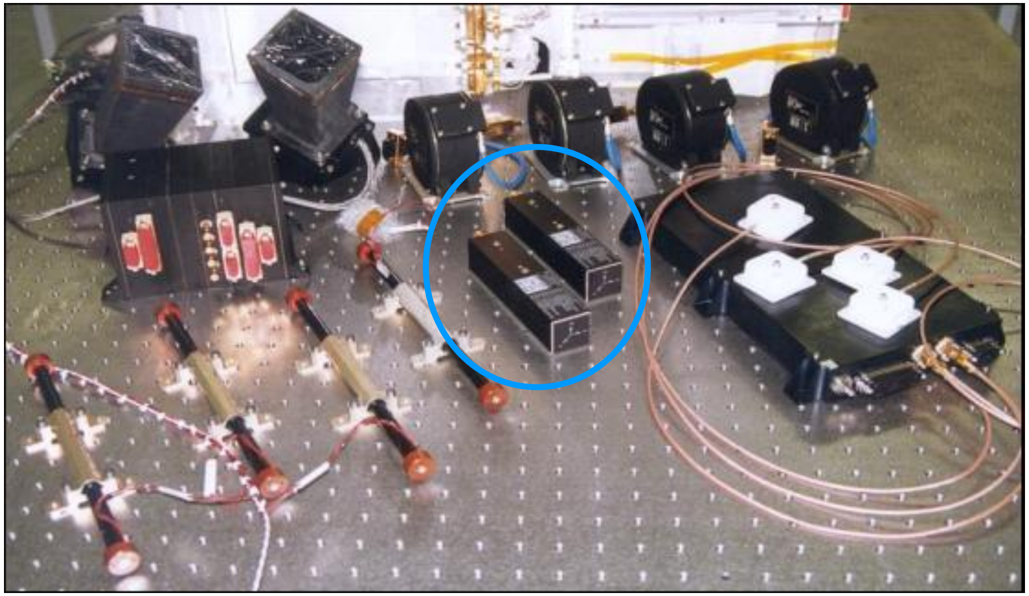
Magnetometer (2)

Measure continuously the
Earth Magnetic field and
determine itselve where the North is.



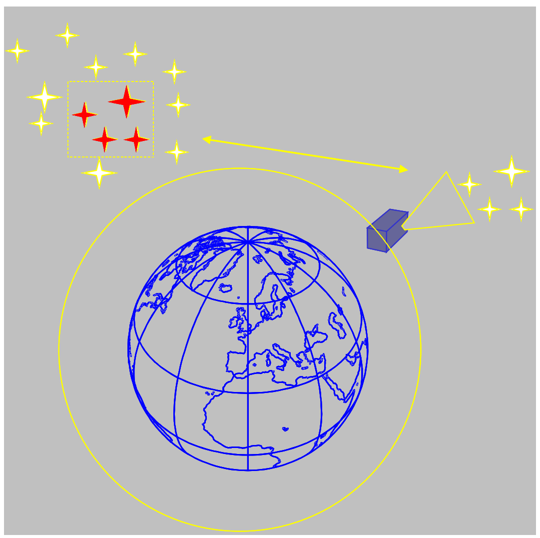


⇒ Position determination

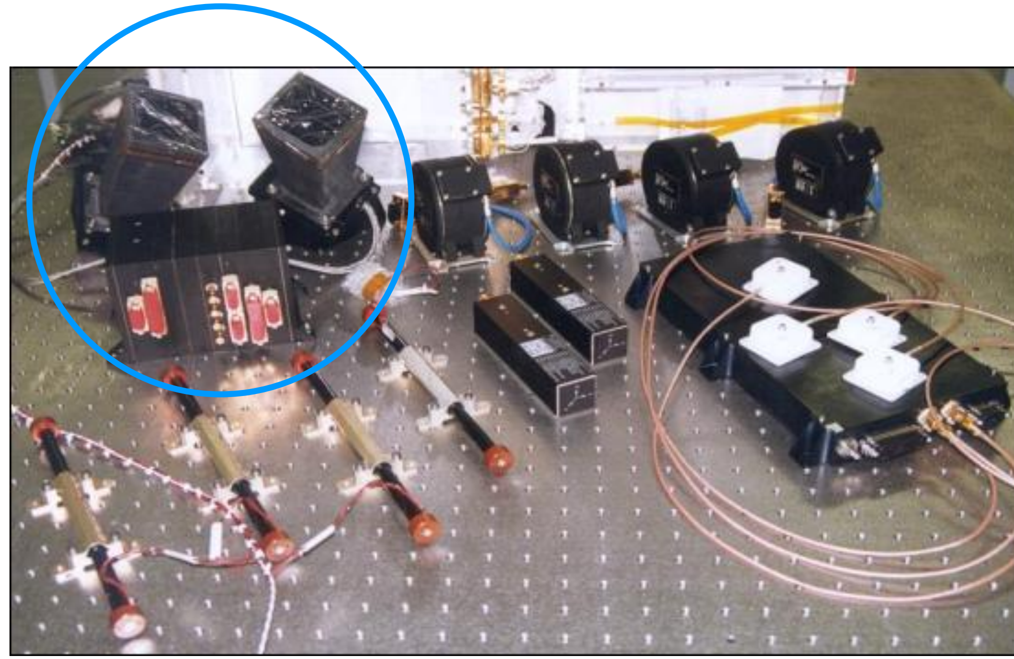


Star-Tracker (3)

•Takes pictures of stars
• Compare it with its internat catalog.
• Compute the satellite orientation/position

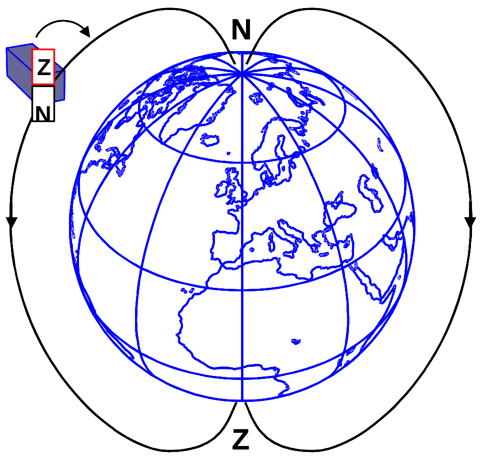


⇒ **Orientation/Position**



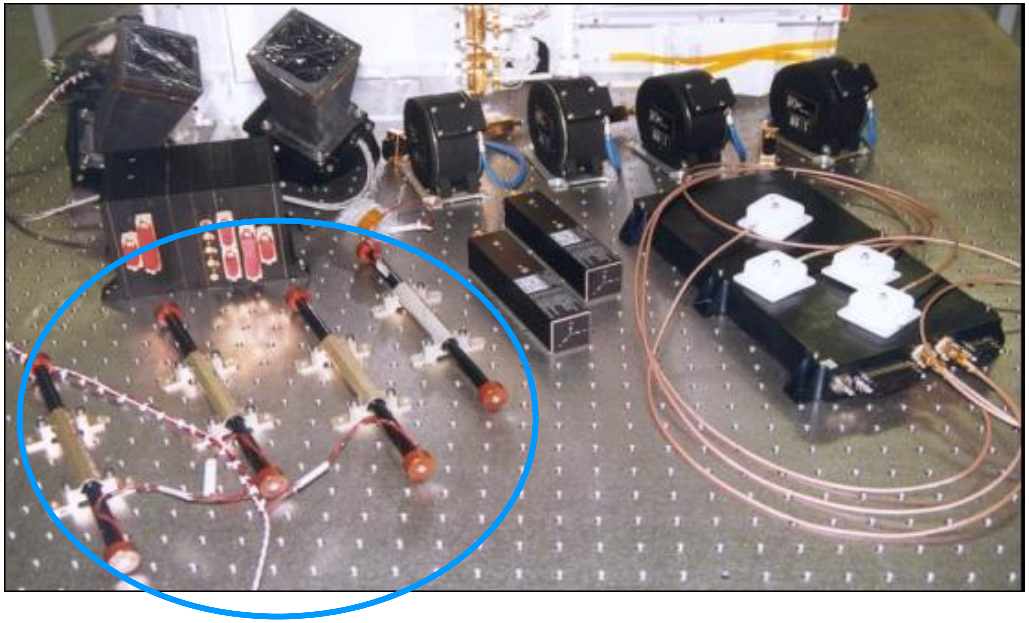
3 PROBA - 2

Magnetotorquer (4)



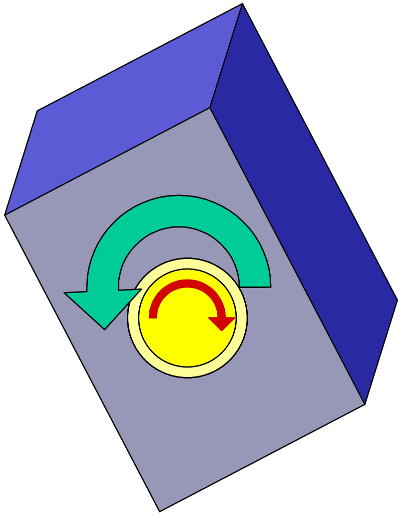
- Magnetic Coil
- Align itself to Magnetic lines

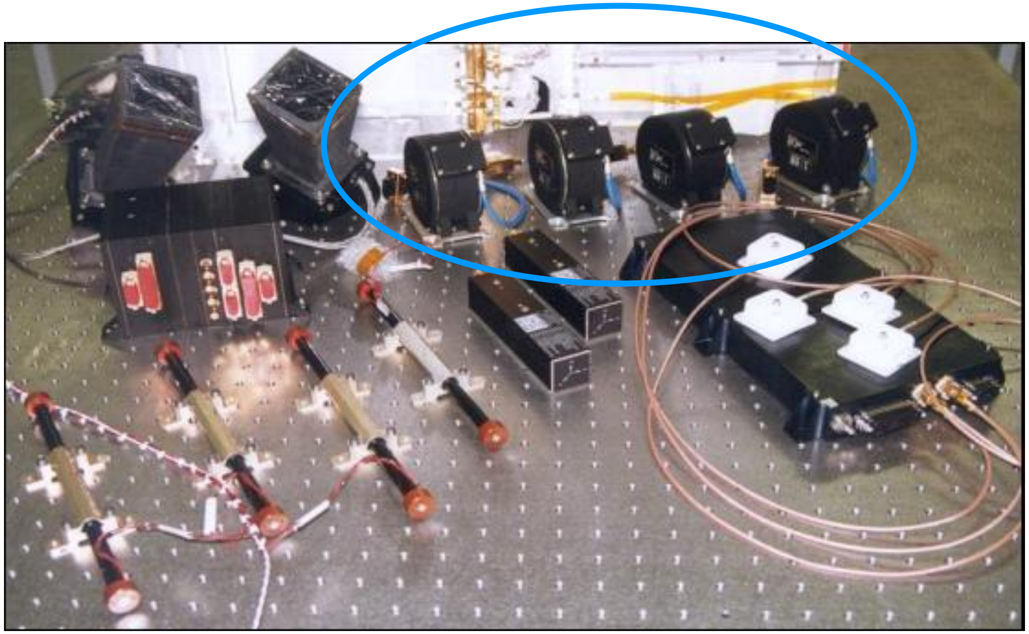
⇒ **Orientation/Manoeuvre**



3 PROBA - 2

Rection Wheel (4)





Accelerates or decelerates while momentum conservation implies the PROBA to rotate the other way.

⇒ **Orientation/Manoeuvre**

3 PROBA - V

3 PROBA - V

- **Mission**

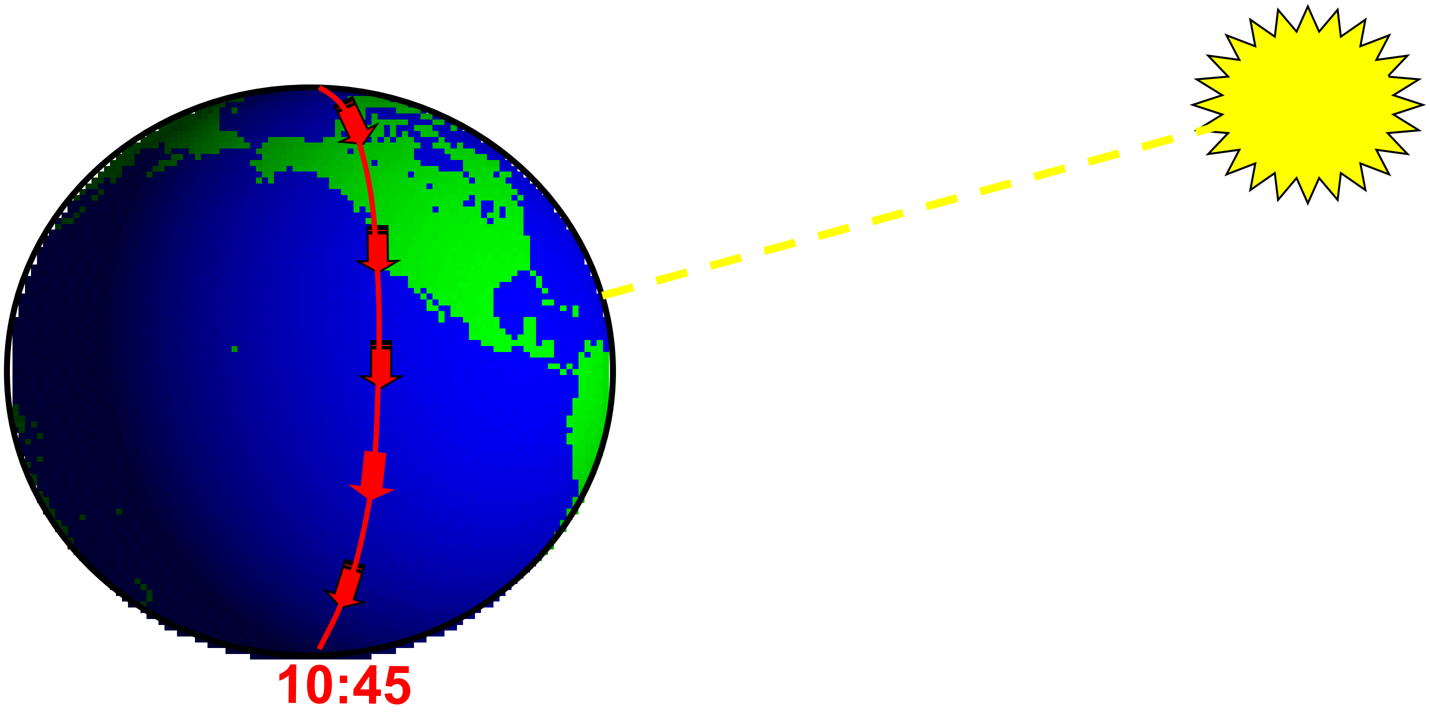
1. Providing **Daily Vegetation Global Monitoring** Capability to Scientific Community

	Key Performance data
Orbit	Quasi SSO (slightly drifting LTDN – injection 10:45 AM) i = 98.73 deg e = 0 Altitude = 820km
Mission Lifetime	2.5 year
Daily global coverage	Latitudes 35° to 75°N Latitudes 35° to 56°S
Coverage after two days	Latitudes between 75°N and 56°S

3 PROBA - V

- **Mission**

1. RAAN selected for 10:45 Local Time Descending Node.



3 PROBA - V

- **Mission**

1. Launcher is **VEGA**
   - Semi-major axis accuracy of 15km
   - Inclination accuracy of 0,15 deg
   - Better than PSLV
     - Inclination accuracy of 0.2°
     - Altitude accuracy of 35km

2. Launch site: **Kourou**
3. Launch Date: **7th of May 2013**

3 PROBA - V

- **Mission**

1. Ground station selection
   - Svalbard / Kiruna / Fairbanks – Payload data downlink
   - REDU for mission control

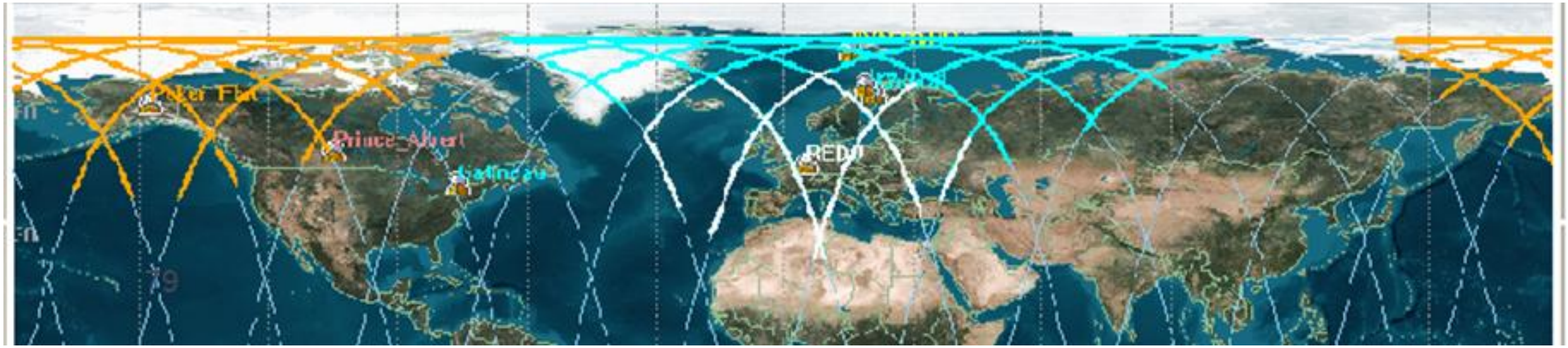
	Svalbard	Redu	Kiruna	Alaska
Location	Norway	Belgium	Sweden	Fairbanks
Controlled by	KSAT	ESA	SSC	USN
Co-ordinates	78.13 ºN, 15.23ºE	50.01°N, 5.14°E	67.85°N, 20.96°E	64.9°N, 147.9°W
Antenna diameter [m]	11/13	2.4	15/13	10
Bands available	S/X	S	S/X	S/X
Min Elevation [Deg]	5	5	5	5
EIRP [db]	64	72.5	71/69	24/37
G/T, sensitivity [db]	35.4	29 (S)	21.4/35.6	22/32
Altitude [m]	455	386.6	402	149

Source: ESA ESTRACK manual

3 PROBA - V

- **Mission**

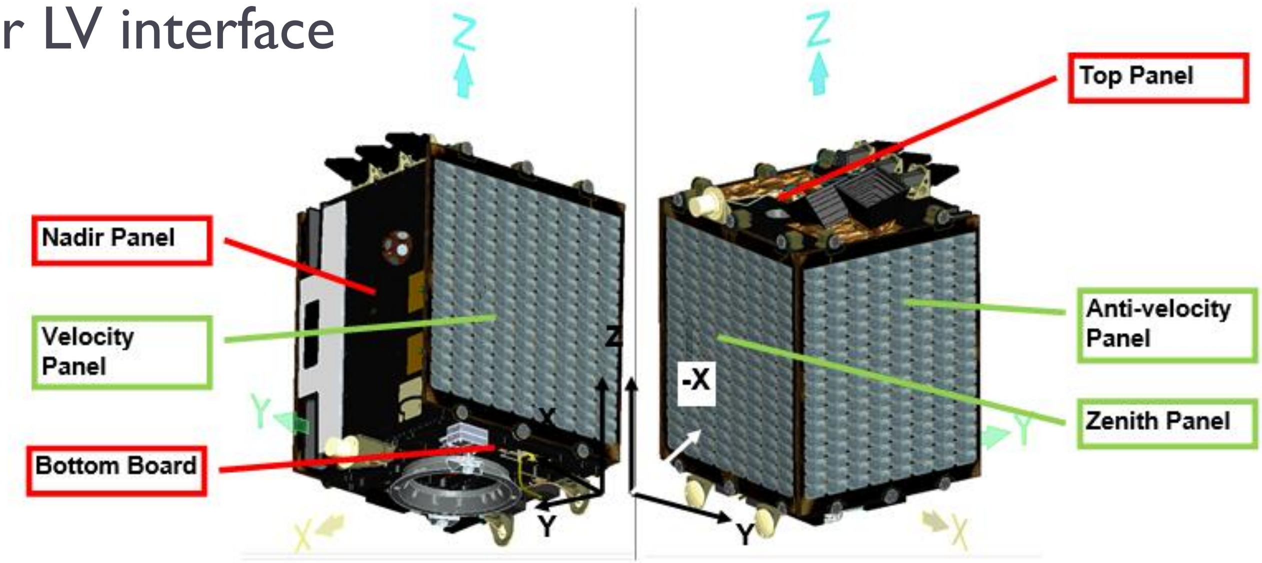
1. Ground station selection
   - Fairbanks & Kiruna overlap! No steerable antenna on board
   - Necessity to interrupt connection



3 PROBA - V

- **Satellite Design - Configuration**

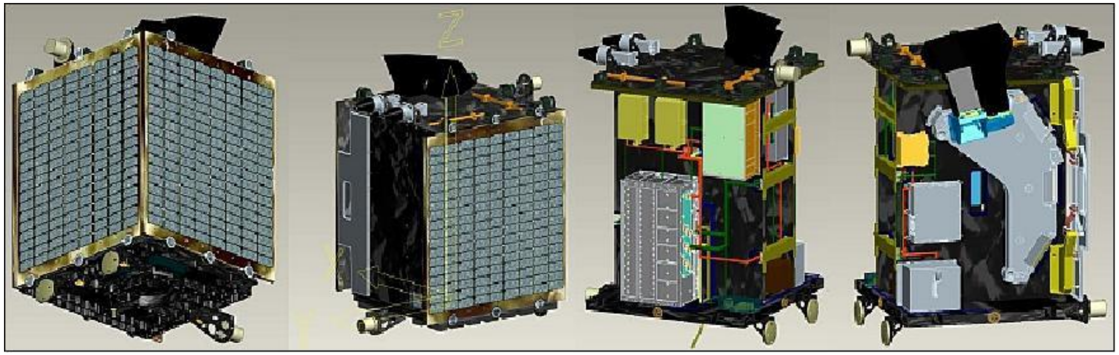
1. X-band antenna toward Nadir
2. Solar Array on Velocity, Zenith & Anti Velocity
3. Star Tracker looking as much as possible towards deep space
4. Bottom board for LV interface



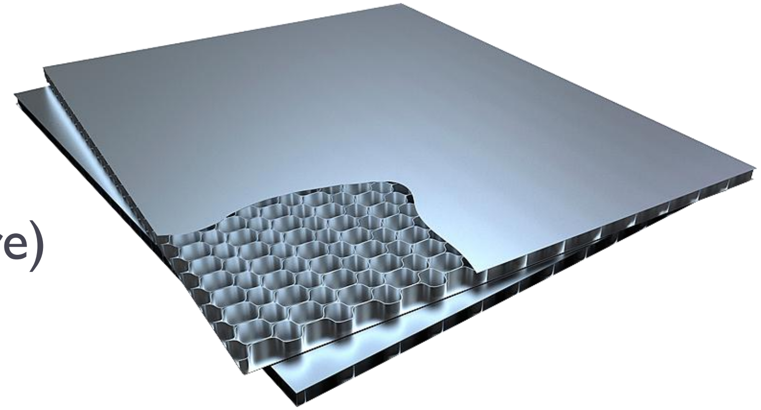
3 PROBA - V

- **Satellite Design - Structure**

1. Single H structure
2. Stiffening beams



3. Honeycomb panels
   - Aluminium core
   - Aluminium edge
   - Alumiunium facesheet (Primary structure)
   - CFRP facesheet (Secondary structure)



3 PROBA - V

- **Satellite Design - Mechanics**

1. Spacecraft Mass = 148 kg with given margin philosophy

	Mass (w/o margin)	Margin	Mass w/ margin	Note
Unit 1	5kg	5%	5.25kg	No modification
Unit 2	2kg	10%	2.2 kg	Modification
Unit 3	3kg	20%	3.6 kg	New development
TOTAL	**10 kg**		**11.05 kg**	**w/o Sys Margin**
	System Margin	**20%**	**13,26kg**	**w/ Sys Margin**

2. Balance Mass of for CoG Location Requirement

Launcher	Lateral Tolerance [mm]	Longitudinal Tolerance [mm]
VEGA	10	< 500

3 PROBA - V

- **Satellite Design - Power**

1. Power budget approach could be:
   - Rely on Solar Array (SA) in Sun and on Battery in Eclipse
   - Rely on both Battery and SA when available.

What are the advantages & disadvantages of these approaches?

3 PROBA - V

- **Satellite Design - Power**

1. Power budget approach could be:
   - Rely on Solar Array (SA) in Sun and on Battery in Eclipse
   - Rely on both Battery and SA when available.

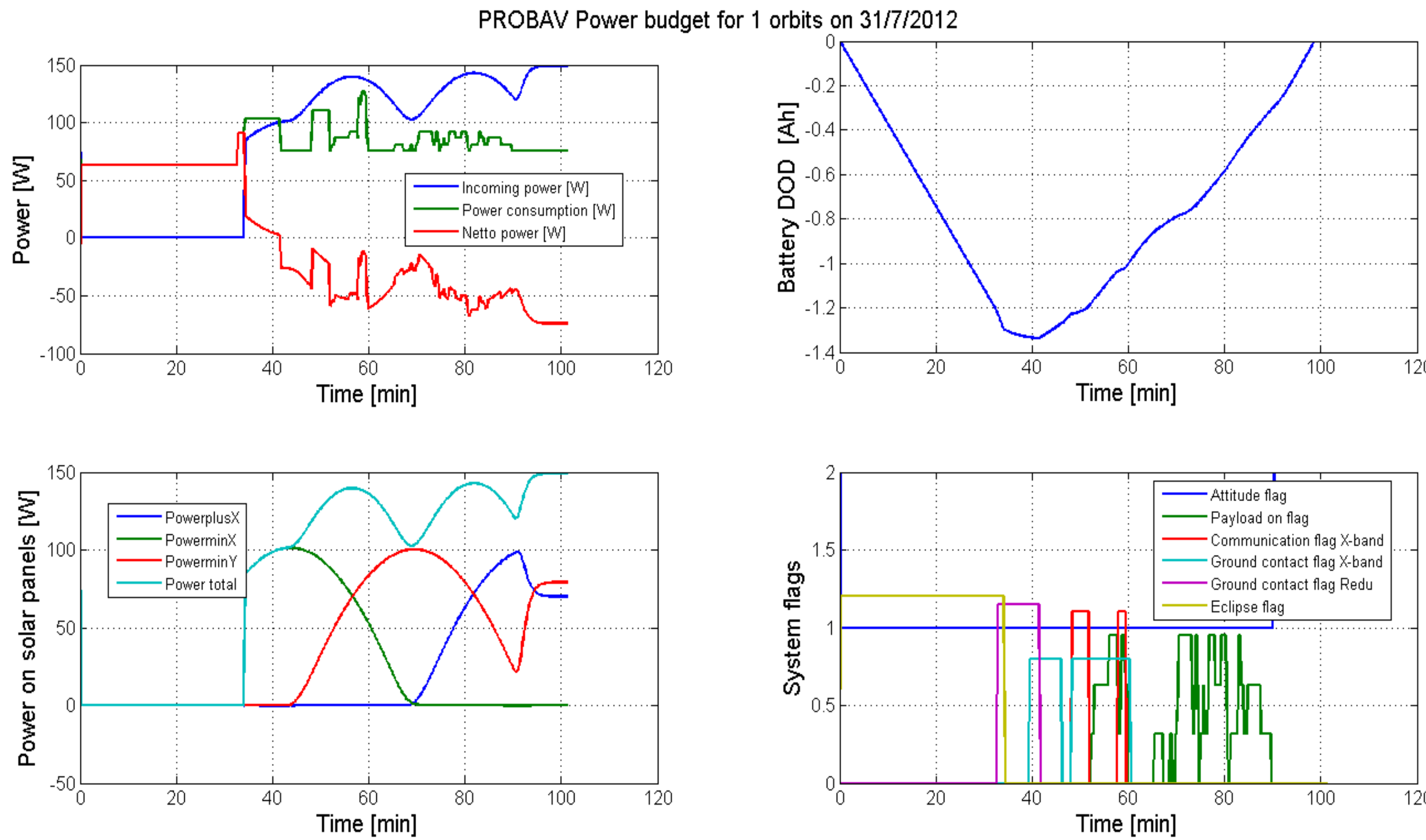
	IMAGING	X-BAND	STD-BY
PDHU	20W	20W	20W
COM	5W	40W	5W
PAYLOAD	32W	5W	5W
...			
TOTAL	85 W	120 W	40W
Sys Marg	**5%**	**5%**	**5%**
TOTAL	**90W**	**125W**	**45W**

Typical power budget

3 PROBA - V

- **Satellite Design - Power**

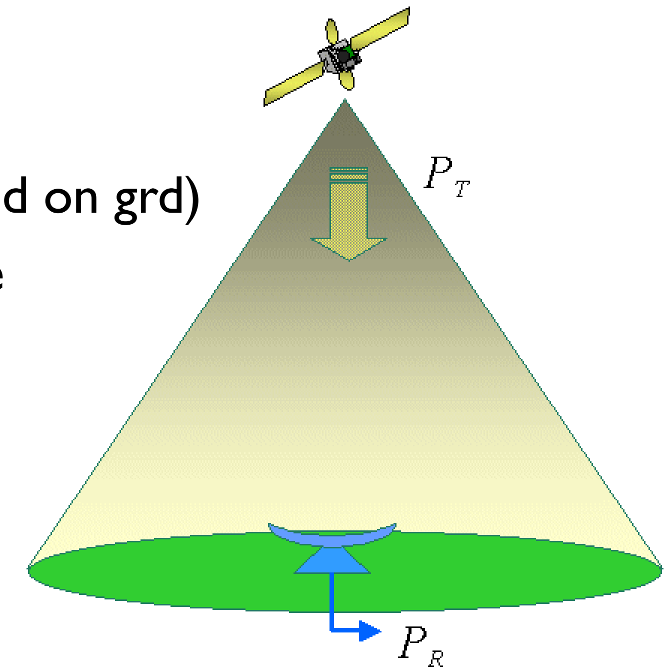
1. Power > 0 (consumption) & Power < 0 (recharge)



3 PROBA - V

- **Satellite Design – RF COM**

1. Downlink (S-Band) – 2235 MHz
   - Data rate = 142kbps (BPSK modulation)
   - Symbol rate = 329 ksps (Convolutional-Reed Solomon Coding)
   - **Two things to look at:**
     - Flux margin (avoid too much power received on grd)
     - Telemetry Recovery margin (avoid too little power to read the telemetry) (Eb/N0)
   - Good practice = 3 dB



3 PROBA - V

- **Satellite Design – RF COM**

1. How to make quick check for Flux Density?
   - +Power @ Transmitter (dBW)
   - - Circuit Loss (3dB)
   - +S/C Gain of Antenna (dBi)
   - - 10*LOG(4*π*(slant_km*1000)²)

   EIRP (first three lines)

   = Power Flux @ G/S (dBW/m²)
   - -10*LOG(bit_rate_kbps)+6+30

   = Power Flux Density @ G/S (dBW/m²/4kHz) shall be < Required

3 PROBA - V

- **Satellite Design – RF COM**

1. How to make quick check for TM Recovery margin?
   - +Power @ Transmitter (dBW)
   - - Circuit Loss (3dB)
   - +S/C Gain of Antenna (dBi)
   - - Path Loss (dB) - Atmosphere & Polarization mismatch (1dB)
   - + G/T of the Antenna (dB/K)
   - - Boltzmann constant (dBW/HzK)

= C/N0 (carrier to noise ratio) (dBHz)

- - TM Demodulation Loss (dB) depends on complexity
- - TM BitRate (dBHz)

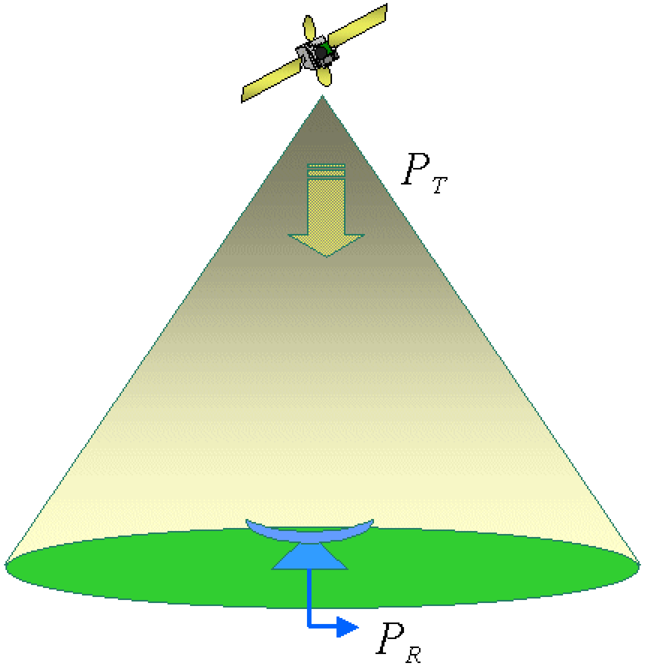
- Eb/N0 (dB) shall be > Required

3 PROBA - V

- **Satellite Design – RF COM**

1. Downlink (S-Band) – 2235 MHz
   - Data rate = 142kbps (BPSK modulation)
   - Symbol rate = 329 ksps (Convolutional-Reed Solomon Coding)

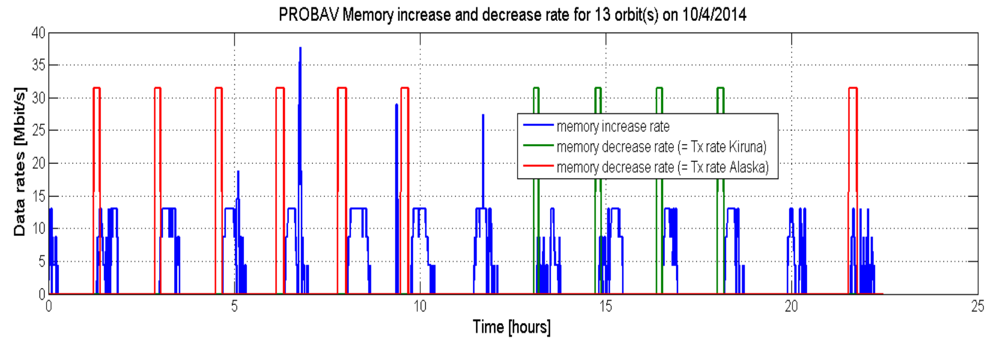
2. Uplink (S-Band) – 2058 MHz
   - Data rate = 64kbps
   - Carrier Recovery margin of 20dB
   - Telecommand Recovery margin of 6dB

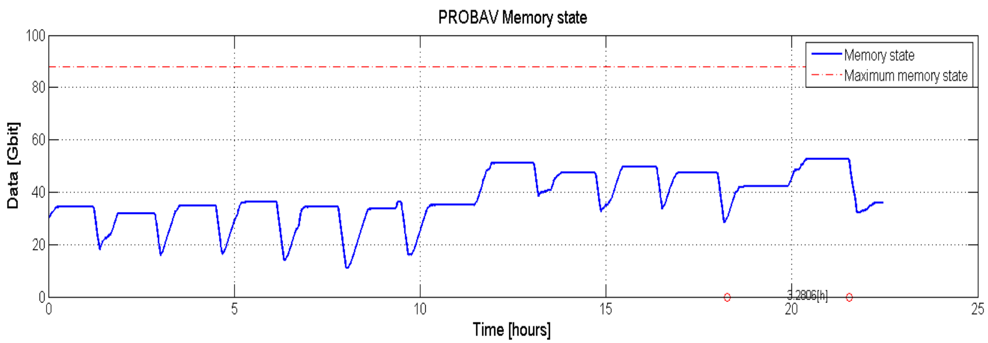


3 PROBA - V

Satellite Design – Memory

1. Memory
   - Mass Memory size = 90Gbit
   - Required < 55 Gbit
   - Total Generated < 230 Gbit





2. Ground contact
   - 20 contact per day – 8 skipped (assumption)
   - Max time delay = 3,2 hours

3 PROBA - V

- **Satellite Design – AOCS**

1. Guidance

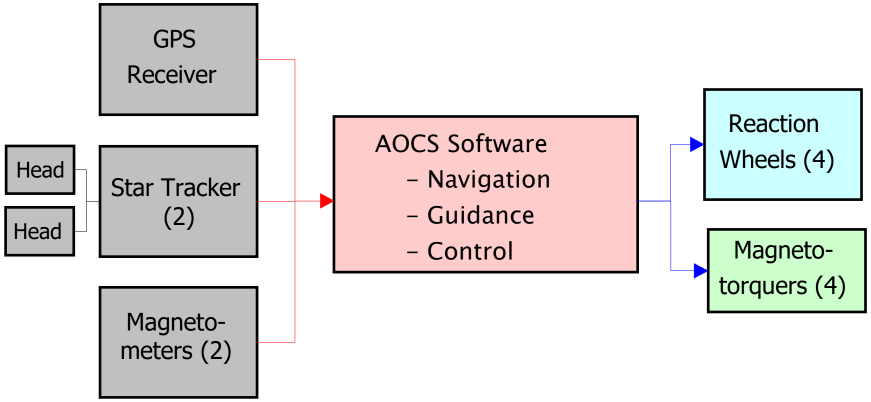
   = *determination of the desired path of travel from the satellite's current location to a designated target*

2. Navigation

   = *determination of the satellite's location, velocity and attitude*

3. Control

   = *manipulation of the forces needed to track guidance commands while maintaining satellite stability*



3 Phase A

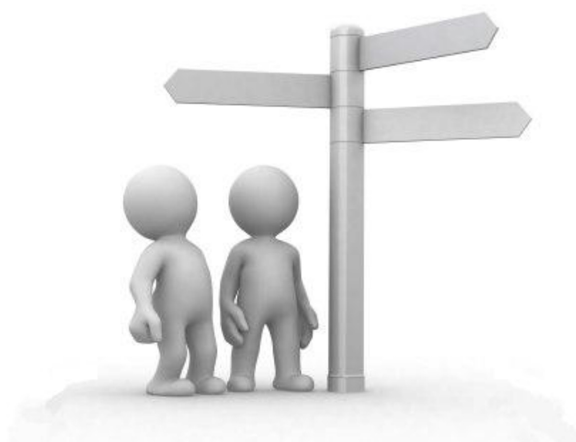
- **What are the typical sources of error?**

3 Phase A

- **What are the typical pointing requirement one usually finds?**

3 Phase A

- **How much deviation does it give on ground if the satellite flies at 800 km?**



(give it in 2-sigma & 3-sigma confidence)

3 Phase A

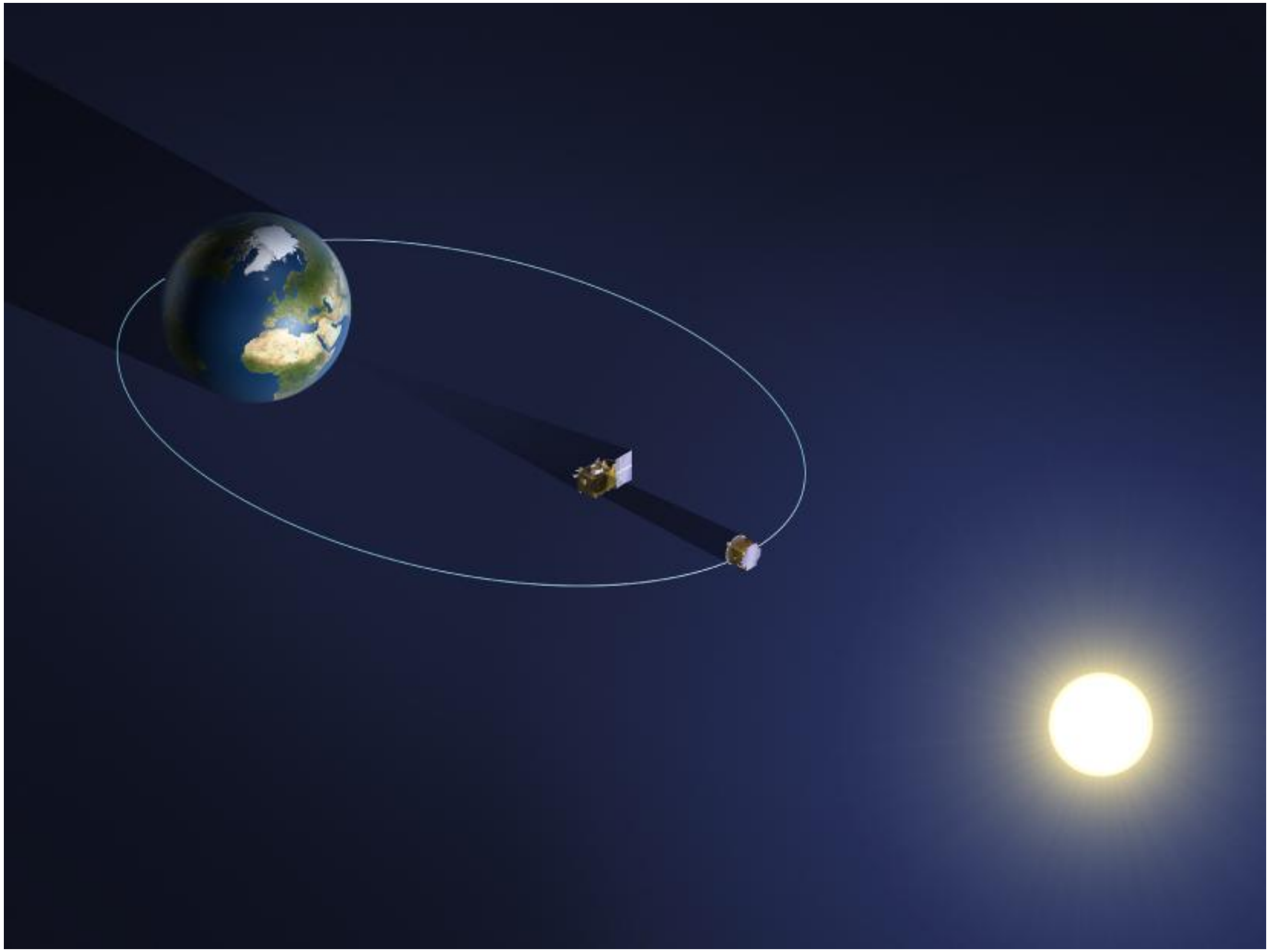
- **How do you think it can compare to a 6U cubesat flying at 500 km?**

3 PROBA - V

- **Launched on the 07/05/2013 from Kourou**

4 CASE STUDY

3 SOLAR OCCULTATION MISSION



3 SOLAR OCCULTATION MISSION

- **Mission**

1. ESA asks you to put **two spacecrafts** in High Elliptical Orbit (HEO) with 20 hours period so that one can occult the Sun while the other one collects picture of its Corona (scientific purpose)

Parameter	Value
Orbit type	HEO
Perigee altitude	600km
Apogee altitude	60,000km
Inclination	59°
Eccentricity	0.8

3 SOLAR OCCULTATION MISSION

- **What are the advantages and disadvantages of such an orbit?**

3 SOLAR OCCULTATION MISSION

- **What would you put in your spacecraft ?**

Subsystem	Design
Structure	Aluminum (density = 2700 kg/m^3) CFRP (density = 1800 kg/m^3) Invar (density = 8000 kg/m^3)
Thermal	Passive OR Active ?
Mechanism	Body Mounted OR Deployable SA?
Power	Large OR Small SA ? Large OR Small Battery?
GNC	Sensor? + Why ? Actuator? + Why?
RF	High/Low Gain COM Antenna?

3 SOLAR OCCULTATION MISSION

- **How would you Launch the two spacecrafts so that they reach the same orbit?**

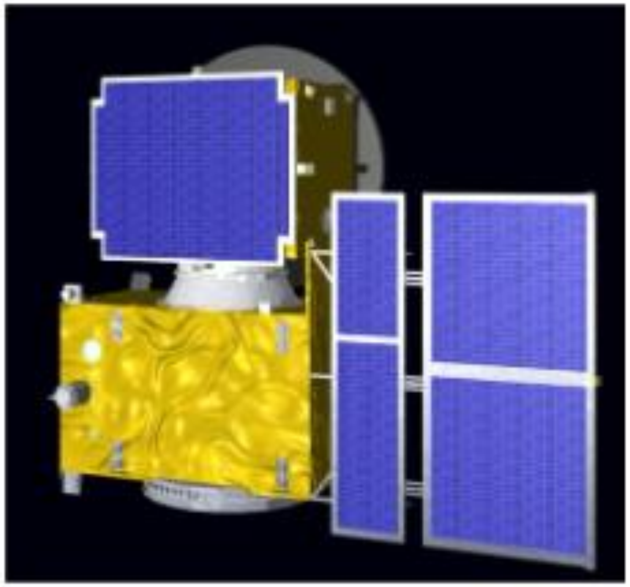
(Two Launches? One Launch?)

(Together? Separated?)

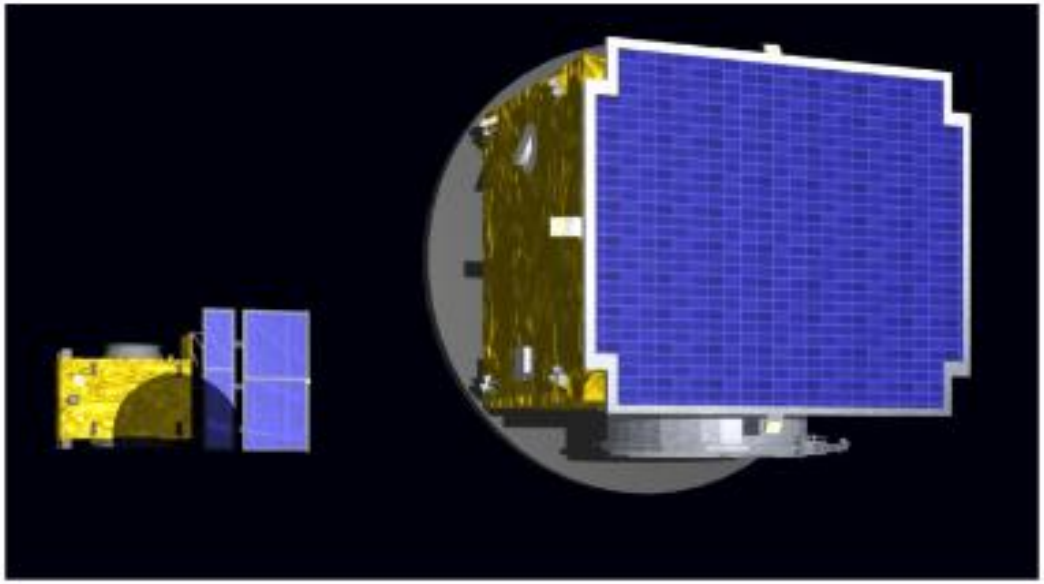
3 SOLAR OCCULTATION MISSION

- **Welcome in Phase A !**

3 Phase A… And the usual problems



STACK (SC1+SC2)



Spacecraft 1 (SC1) & Spacecraft 2 (SC2)

3 Phase A… And the usual problems

1. Mass Budget
   - "Total dry mass including I/F ring with L/V shall be less than 360kg for SC1 and 300kg for SC2".
   - Current estimate: SC1 = 375kg & SC2 = 270kg

2. Power Budget
   - " Total maximum power consumption shall be less than 300W (with margins) for both spacecrafts"
   - Current estimate: SC1 = 350W & SC2 = 250W

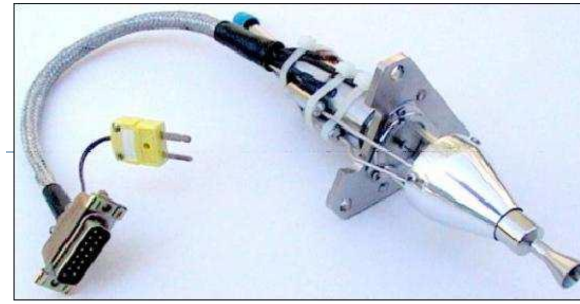
3 Phase A… And the usual problems

1. Link Budget
   - "The SC1 shall be able to downlink data at a symbol rates of 2Msps"
   - "Both SC shall be able to receive TC data at a symbol rates of 64ksps"
   - Current estimation with 2.4 meter antenna (REDU)
     - Downlink not closed for 2Msps (@apogee) for S/C1
     - Uplink not closed for 64ksps (@apogee) for neither SC…

3 Phase A

- **What can we do?**

3 Phase A

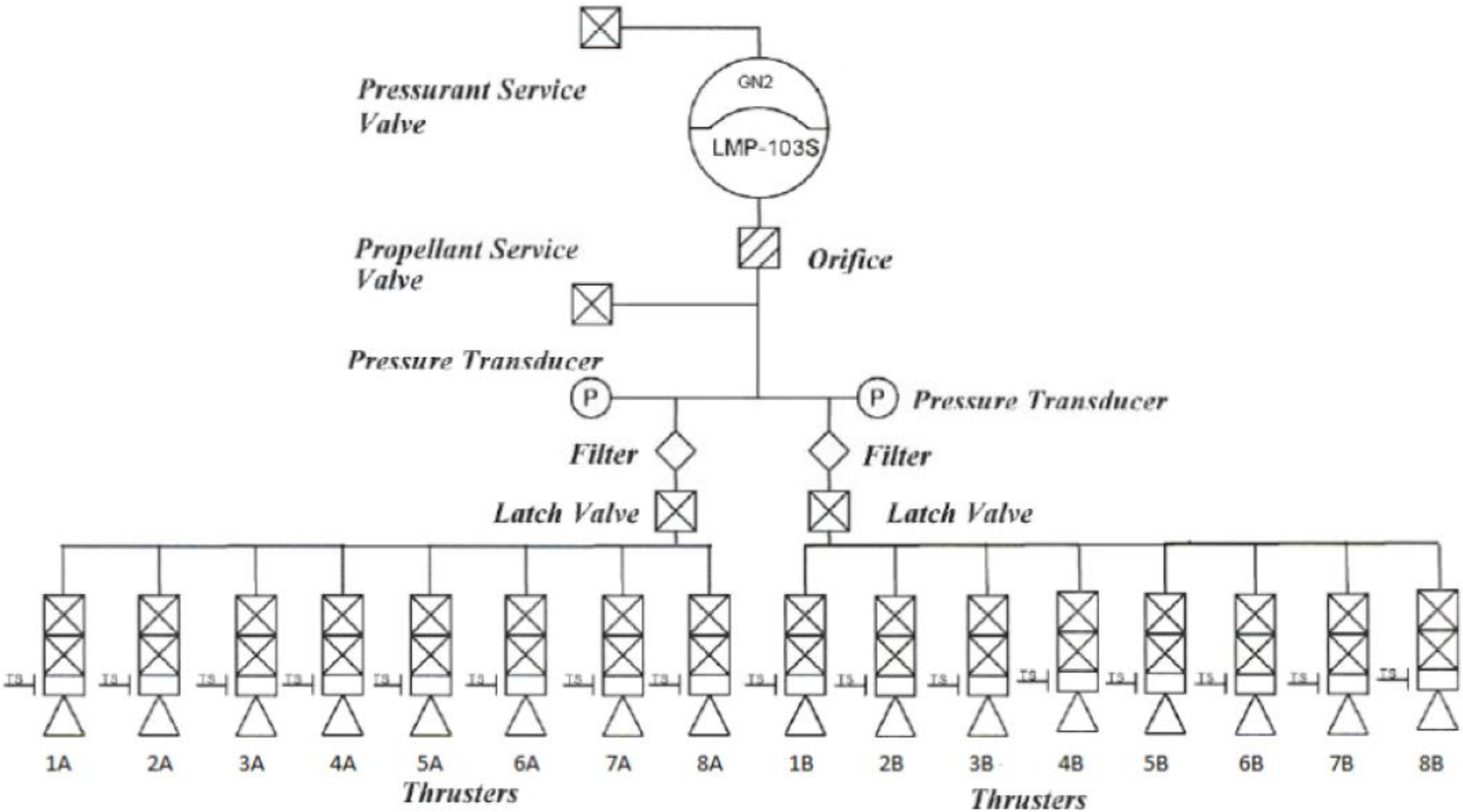


- **For Information**

1. SC1 has two propulsion subsystems
   - 2x8 thrusters to raise orbit (Mono-propellant)
   - Sixteen thruster for GNC and FF (Cold Gas)
2. SC2 does not have any propulsion
3. Monopropellant shall be maintained above 10°C (always) and thus currently consumes 90W just for heating.
4. Monopropellant thrusters shall be pre-heated to give their maximum performance (1N and 202 sec of Isp)
5. Monopropellant sub-system weight 27kg
6. Payload is currently kept warm constantly (40W)

3 Phase A

- **For Information**



3 Phase A

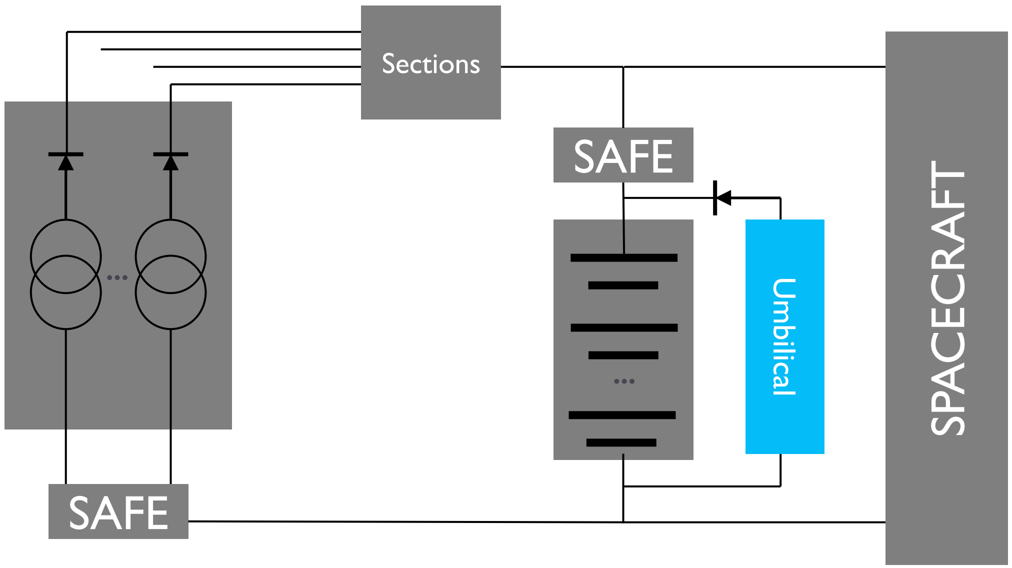
- **What do we need to update if the Power consumption is modified?**

3 Phase A – Electrical Architecture

1. Power Generation & Storage
   - Solar Array
   - Battery

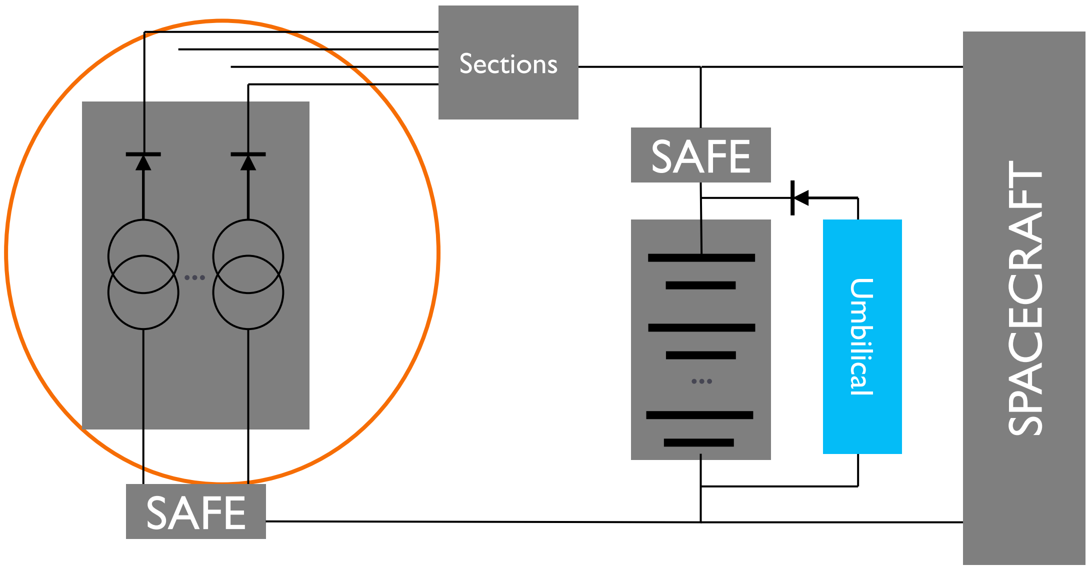
2. Power Conditioning Distribution Unit (not displayed)

3. Connections
   - Safe & Arm
   - Umbilical Connection



3 Phase A – Solar Array Sizing

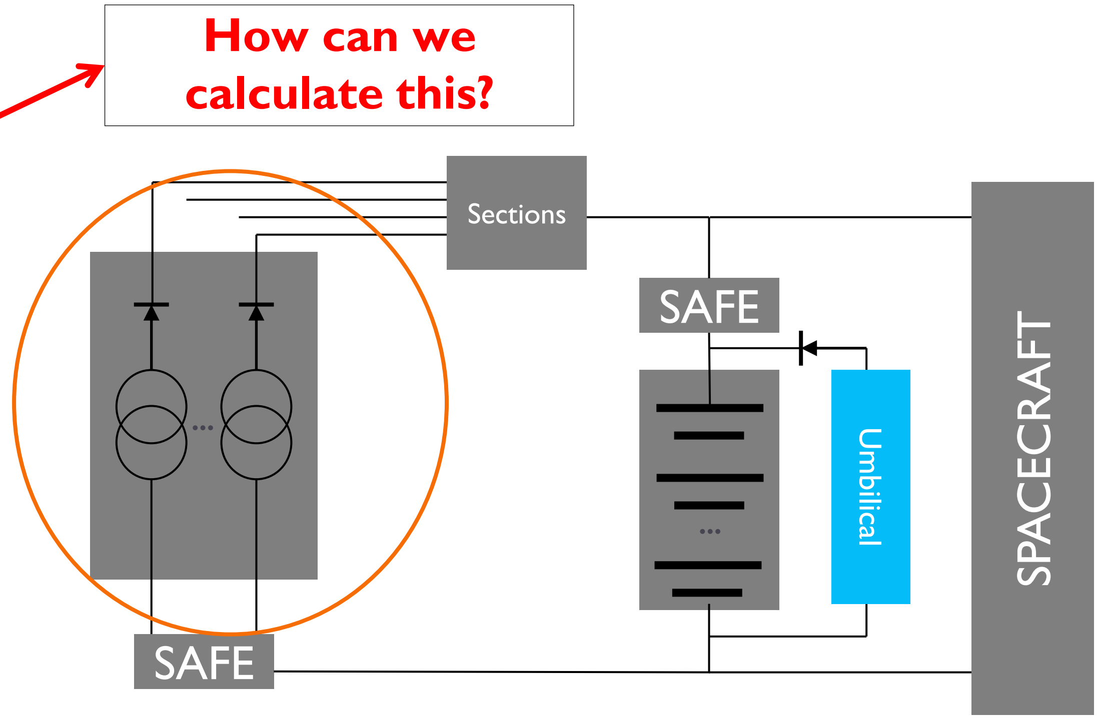
1. **Main components**
   - Cells Series (TBD)
   - String Parallel (TBD)
   - Section (TBD strings)



2. **Secondary components (not displayed)**
   - Shunt Selection
   - Dump Resistor

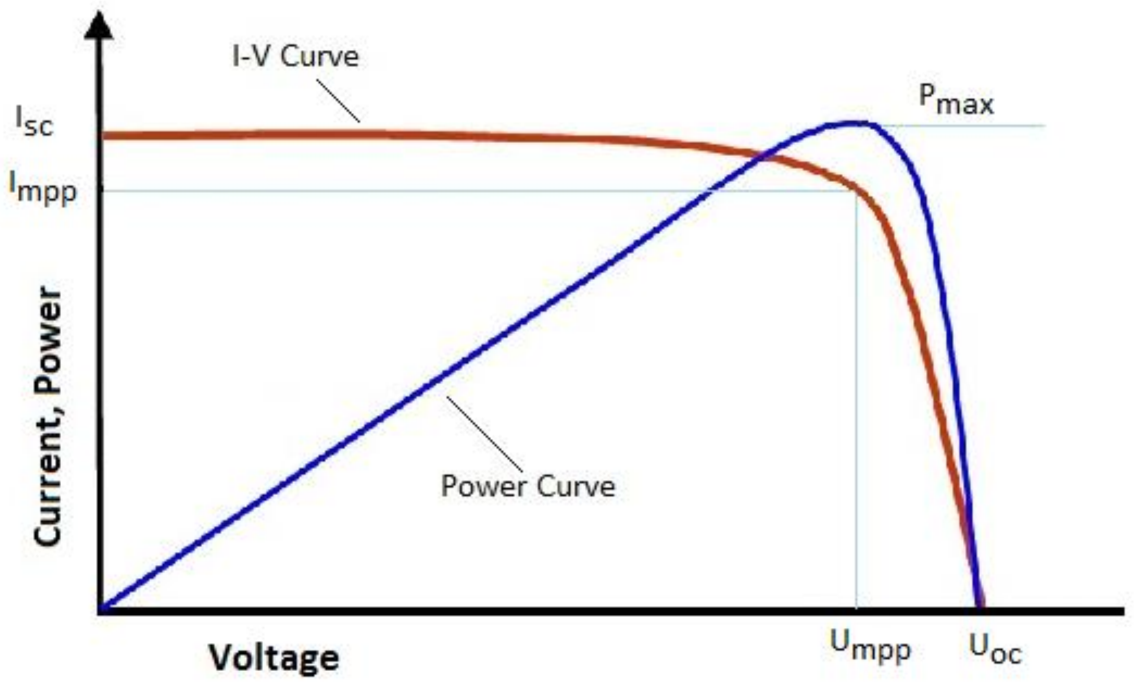
3 Phase A – Solar Array Sizing

1. Main components
   - Cells Series (TBD)
   - String Parallel (TBD)
   - Section (TBD strings)



2. Secondary components (not displayed)
   - Shunt Selection
   - Dump Resistor

3 Phase A – Solar Array Sizing



3 Phase A – Solar Array Sizing

- **Solar Array Design (3G28%)**

1. Evaluate the degradation?
   - Coverglass thickness
   - Degradation of Electrical Parameters
   - Degradation of Temp. Coefficient

Coverglass thickness (µm)	Pmax [#/cm²]	Voc [#/cm²]	Isc [#/cm²]
0,0	5,11E+17	5,99E+17	7,12E+17
25,4	7,64E+14	8,93E+14	8,43E+14
76,2	1,52E+14	1,76E+14	1,35E+14
152,4	7,81E+13	8,99E+13	6,48E+13
304,8	4,16E+13	4,75E+13	3,27E+13
508,0	2,53E+13	2,88E+13	1,94E+13
762,0	1,64E+13	1,86E+13	1,16E+13
1524,0	3,20E+12	9,37E+12	5,43E+12

3G-28% cell: Electrical Parameters drifts in irradiation

EP	BoL	2,00E+13	7,00E+13	2,50E+14	5,00E+14	1,00E+15	3,00E+15
Voc	2650	0,98	0,97	0,95	0,94	0,92	0,9
Isc	498	0,99	0,99	0,99	0,98	0,96	0,87
Vmp	2365	0,99	0,98	0,94	0,93	0,92	0,9
Imp	480	0,99	0,99	0,98	0,97	0,95	0,84
Pmp	1135	0,98	0,97	0,92	0,91	0,87	0,76

3G-28% cell: Temperature Coefficients drifts in irradiation

TC	BoL	2,00E+13	7,00E+13	5,00E+14	1,00E+15	3,00E+15
Voc (mV/K)	-5,96	-6,01	-6,11	-6,15	-6,25	-6,46
Isc (mA/K)	0,348	0,298	0,298	0,340	0,370	0,340
Vmp (mV/K)	-6,01	-6,42	-6,45	-6,26	-6,35	-6,59
Imp (mA/K)	0,316	0,248	0,219	0,220	0,250	0,220
Pmp (mW/K)	-2,54	-2,60	-2,69	-2,66	-2,59	-2,42

3 Phase A – Solar Array Sizing

- **What happens to the Pmax at EoL?**

Consider both radiation and thermal effect

3 Phase A – Solar Array Sizing

- **Example using 3G28%**

I. Evaluate the degradation?

- Coverglass thickness

Coverglass thickness (µm)	Pmax [#/cm²]	Voc [#/cm²]	Isc [#/cm²]
0,0	5,11E+17	5,99E+17	7,12E+17
25,4	7,64E+14	8,93E+14	8,43E+14
76,2	1,52E+14	1,76E+14	1,35E+14
152,4	7,81E+13	8,99E+13	6,48E+13
304,8	4,16E+13	4,75E+13	3,27E+13
508,0	2,53E+13	2,88E+13	1,94E+13
762,0	1,64E+13	1,86E+13	1,16E+13
1524,0	[illegible],20E+12	9,37E+12	5,43E+12

- Degradation of Electrical Parameters

3G-28% cell: Electrical Parameters drifts in radiation

EP	BoL	2,00E+13	7,00E+13	2,50E+14	5,00E+14	1,00E+15	3,00E+15
Voc	2650	0,98	0,97	0,95	0,94	0,92	0,9
Isc	498	0,99	0,99	0,99	0,98	0,96	0,87
Vmp	2365	0,99	0,9[illegible]	0,94	0,93	0,92	0,9
Imp	480	0,99	0,99	0,98	0,97	0,95	0,84
Pmp	1135	0,98	0,97	0,92	0,91	0,87	0,76

- Degradation of Temp. Coefficient

3G-28% cell: Temperature Coefficients drifts in radiation

TC	BoL	2,00E+13	7,00E+13	5,00E+14	1,00E+15	3,00E+15
Voc (mV/K)	-5,96	[illegible]	-6,11	-6,15	-6,25	-6,46
Isc (mA/K)	0,348	[illegible]	0,298	0,340	0,370	0,340
Vmp (mV/K)	-6,01	-6,42	-6,45	-6,26	-6,35	-6,59
Imp (mA/K)	0,316	0,248	0,219	0,220	0,250	0,220
Pmp (mW/K)	-2,54	-2,60	-2,69	-2,66	-2,59	-2,42

3 Phase A – Solar Array Sizing

2. **Evaluate the number of cells ?**
   - Consider worst case (EoL + Hot temperature ≈ 100°C)
   - Max battery voltage to be provided (e.g. 28V)

	Voltage	
Harness voltage drop (incl. connectors)	< 1	V
Solar Array connectors & protections (e.g. diodes)	< 1	V
Maximum Battery Voltage	28	V
Minimum solar cell strings voltage required	28+1+1=30	V
Additional voltage margin	1	V
Required solar string voltage	31	V
EoL cell voltage @ 100°C	1,8	V
Required amount of cell to cells in one string to achieve voltage	18	Cells

3 Phase A – Solar Array Sizing

3. **Evaluate the number of strings ?**
   - Max current allowed by PDHU (e.g. 20A) @BOL
   - Compute current available at EOL, knowing that di/dT < 0 and accounting for Minimum Solar Cste (di/dC > 0)

	Current	
Assuming best case = direct illumination, BOL, hot case	-	-
BOL string current	0.5	A
BOL maximum amount of string	**40**	**strings**
Assuming worst case = 10% shading, low solar constant, cold case	-	
EOL degradation factor	90%	
EOL string current	0.45	A
Shading degradation (10% of 40 strings shaded)	4	strings
Total current available at EOL	16.2	A

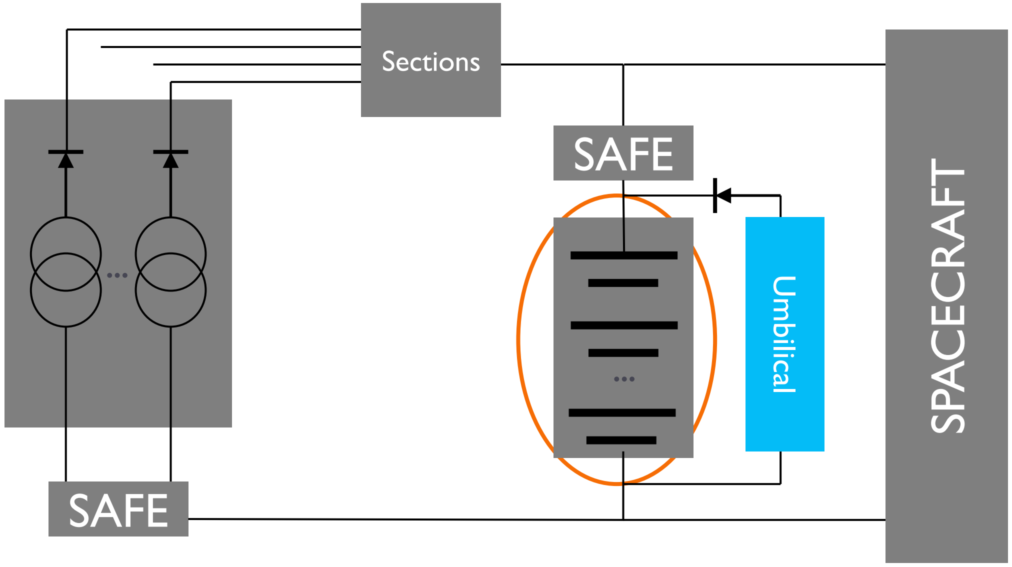
3 Phase A – Solar Array Sizing

- **How do we know that**
- **it is enough to have 40 strings?**

3 Phase A – Battery Sizing

1. Main components
   - Cells Series (TBD)
   - String Parallel (TBD)

2. Secondary components
   - Internat heaters
   - Thermistors

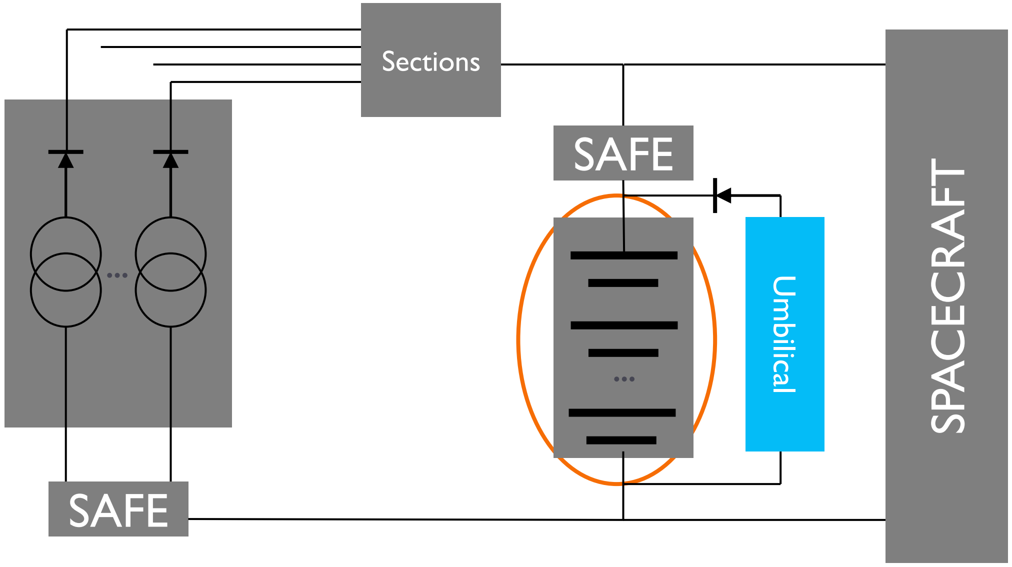


3 Phase A – Battery Sizing

1. Main components
   - Cells Series (TBD)
   - String Parallel (TBD)

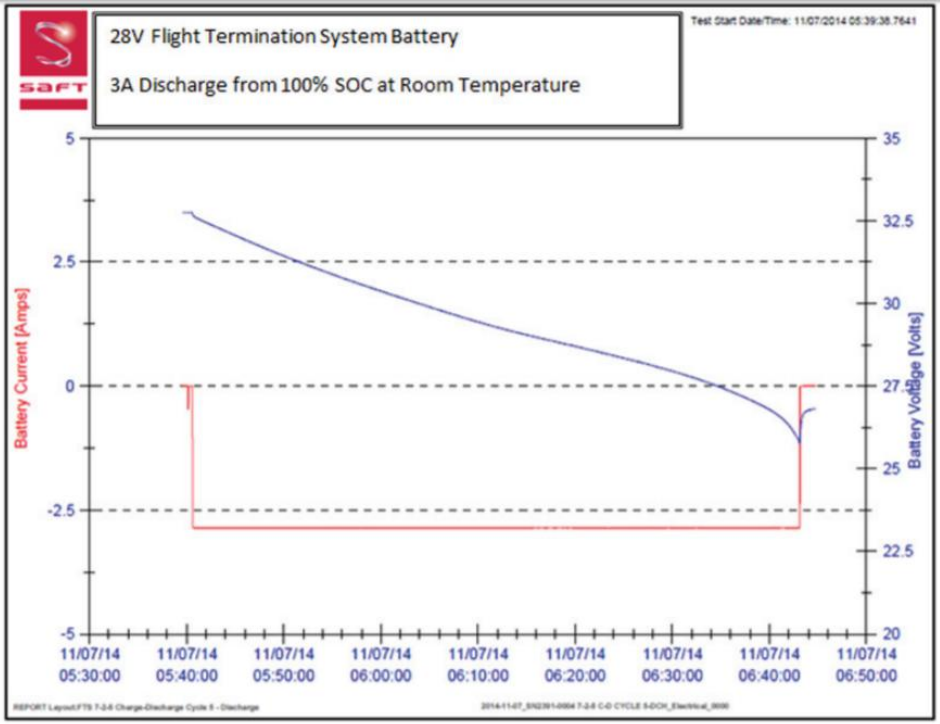
How can we calculate this?

2. Secondary components
   - Internat heaters
   - Thermistors



3 Phase A – Battery Sizing

1. Evaluate the number of cells ?
   - Check Nominal non regulated bus voltage requirement (28V)
   - Find Cells characteristics (e.g. 4V End of Charge - EoL)
   - Combine 7 cells in series to create the voltage



3 Phase A – Battery Sizing

2. Battery shall be used for
   2. Detumbling
   3. Eclipse
   4. Support

Scenario	Used Capacity (Wh)
Detumbling	200
Eclipse	120
Long Eclipse	**665**

Scenario	DoD choice	Required Capacity (Wh)
Detumbling	20%	240
Eclipse	20%	148
Long Eclipse	**60%**	**1064**

3. Evaluate the number of strings ?
   - $Capacity_{used}$ = Power * Time
   - $Capacity_{required}$ = $Capacity_{Used}/(1\text{-}DoD_{Allowed})$
   - Capacity of 1 string = $Nb_{Cells}*Capacity_{Cell}$
   - One String Failure Tolerance

Parameter	Value
Nb_{Cells}	7
$Capacity_{Cell}$	5,4 Wh
Capacity 1 string	37,8 Wh

29 (+1) Strings

3 WHERE ARE WE NOW?

1. Mass Budget
   - “Total dry mass including I/F ring with L/V shall be less than 360kg for SC1 and 300kg for SC2”. ✓
2. Power Budget
   - “ Total maximum power consumption shall be less than 300W (with margins) for both spacecrafts” ✓

3 WHERE ARE WE NOW?

1. Link Budget
   - "The SC1 shall be able to downlink data at a symbol rates of 2Msps"
   - "Both SC shall be able to receive TC data at a symbol rates of 64ksps"
   - Current estimation:
     - Downlink not closed for 2Msps (@apogee) for S/C1
     - Uplink not closed for 64ksps (@apogee) for neither SC...

3 Phase A

- **What can we do?**

3 We have modified requirement

1. Link Budget
   - "The SC1 shall be able to downlink data at a symbol rates of **256ksps and** 2Msps"
   - "Both SC shall be able to receive TC data at a symbol rates of 64ksps"

Note: REDU antenna of 15 meter diameter is now considered.

Operational cost has been increased.

4 CONCLUSION

4 CONCLUSION

1. Follows the Project Life Cycle
   - Starts with Mission Concept
   - Prepares System Requirements
   - System Designs based on Technical & Programmatic Trade-Offs
   - Prepare sub-system & unit requirement + analysis
   - Link all levels together (= unit requirement verifying sub-system requirement, themselves verifying system requirements)
   - Manufacture the S/C
   - Verify Requirements (Review of Design / Analysis / Test)
   - Launch it !

2. Iterative Multi-disciplinary approach + Massive Communication