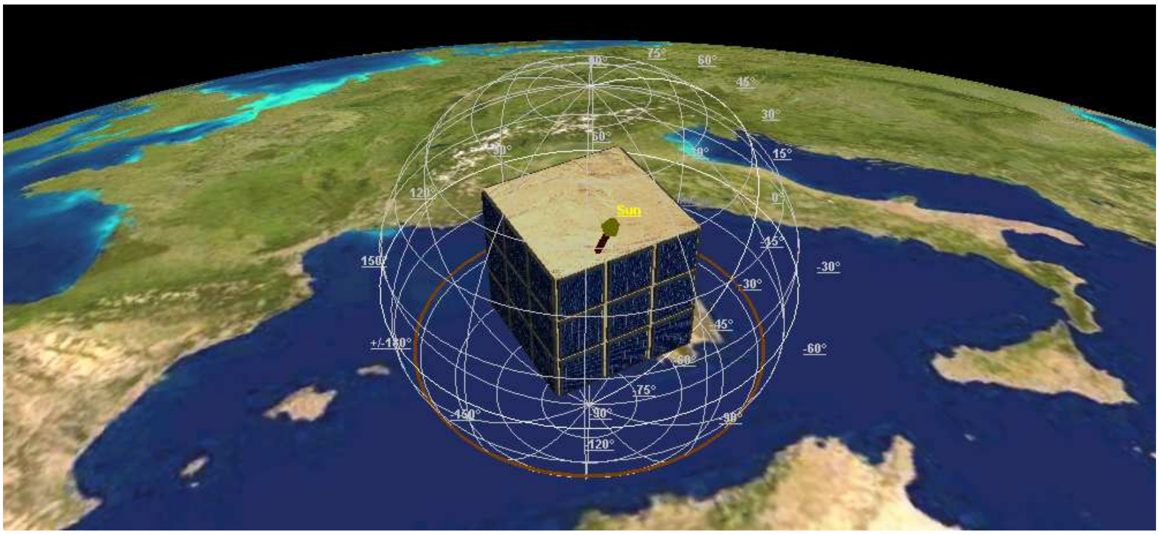
Preliminary design of a micro satellite for Earth observation



The purpose is to make a preliminary design of a micro satellite, by integrating the following design fields :

- orbitography and mission analysis
- radio communication
- thermal control system
- power supply system
- attitude control system

Technical spécifications :

The mission consists in taking 2000 per 2000 pixel pictures (12 bits/pixel) of the Earth in an uncompressed format. The pictures are recovered by ground stations during sunlight periods of the satellite only . The following specifications are given :

- Launch by ARIANE 5 ASAP (Ariane Structure for Auxiliary Payload)
- Orbit :
    - Circular Low Earth Orbit
    - Altitudes of 800 km, 1000 km, 1200 km, 1400 km
    - Inclination of 28°, 52°, polar, sun synchronous
- Attitude : one axis toward the Earth
- 3 ground stations :
    - Cayenne
    - Toulouse
    - La Réunion

Available softwares :

- SATORB
    - Orbitography, mission Analysis, coverage
    - Radio communication
- SIMUSAT
    - Power supply subsystem design
    - Attitude control subsystem
    - Simplified thermal analysis
- EXCEL

Part 1 : Orbitography, satellite visibility

Softwares : SATORB, EXCEL

The purpose of this part is to get familiarized with the constraints imposed by a low Earth orbit mission in terms of visibility and pointing from ground stations.

1-1 Create a new project in the SATORB software

Create a new project (Satorb:Create new project)
Define the satellites *(New/Circular ou New/Héliosynchronous)* for a chosen inclination 28°, 52°, polar, heliosynchronous (Keplerian Propagator).
Define the three ground stations. The minimum elevation needed to see the satellites is fixed to 5°.

1-2 Visibility : Calculation of the mean duration

For every satellite and ground station, print the visibility report *(Analysis/Access/Access)*.
Performing the simulation over 4 to 5 day is sufficient to obtain a reliable mean value. Calculate and analyze the daily duration of visibility for each satellite altitude and each ground station.

1-3 Coverage

Analyse the coverage as a function of the latitude *(Analysis/Coverage/Custom region)* by selecting a zone of around 30° large in longitude, from the equator to the pole. A simulation of 2 to 3 days is sufficient to have a good estimation of the mean coverage.
By selecting a zone that includes the three ground stations, verify the previous results.

1-4 Analysis of the orbit calculation accuracy

Define a new satellite identical to one of the previous ones, using the SGP4 (NORAD) propagator instead of the Keplerian propagator.
Analyze (during an orbital period) the differences between the two propagators *(Analysis/Ephemeris/Position_Velocity)*.
What is the evolution of the position difference on several orbits ?

1-5 Consequences of the calculation accuracy on the ground antenna pointing

We suppose a temporal error of 4 seconds on the location of the satellite we are tracking.
Edit the report of one of your stations *(Analysis/Access/Azimuth_Elevation_Range)* for a long enough visibility (for instance giving a maximum elevation around 60°/70°) with a time step of 4 seconds.
Plot the tracking error during the visibility period.

Part 2 : Radio communication

Softwares : SATORB, EXCEL

2-1 Specifications

Mission objectives :

Obtaining 2000 per 2000 pixel scenes (12 bits/pixel) of the Earth in an uncompressed format. The pictures are collected by ground stations only during sunlight periods of the satellite.

Global specifications :

Link	Satellite	Ground Stations
Frequency : 2.270 GHz (SHF) Bit Error Rate : 10^{-6} Modulation : BPSK No data encoding Rate : 250 Kbps (25% used for bus control and telemetry) Demodulator degradation : 1.5 dB Atmospheric and rain losses : 0.2 dB Margin : 5 dB	Antenna providing a 3 dB coverage for a minimum elevation of 5° Maximum depointing : 5° Antenna efficiency : 0.6 Internal losses : 1 dB Power amplifier ratio : 20% (emitted power/input power)	Antenna diameter : 1.80 m Antenna efficiency : 0.5 Noise temperature : 200°K

2-2 Power emission calculation for the downlink

From the specifications, calculate the needed power emission according to the different orbit altitudes.
Deduce the mean power the power supply system need to furnish during sunlight periods of the satellites.
Define a downlink in SATORB software and compare the obtained results.

2-3 Effects of an antenna depointing

During a visibility, analyze the consequences of an antenna depointing error on the link budget.
Deduce the maximum temporal error allowed on the location of the satellite to have a positive RF margin.

2-4 Reception capacity

Estimate the mean number of pictures collected per day for each altitude.

2-5 Uplink sizing

Calculate the power of the ground station transmitter using the following data :

Link	Satellite	Ground Stations
Frequency : 2.092 GHz (SHF) Bit Error Rate : 10-6 Modulation : FSK No date encoding Rate : 10 Kbps Atmospheric and rain losses : 0.2 dB Margin : 5 dB	Patch antenna Gain : 3dB Noise temperature : 400°K Losses : 2 dB	Antenna diameter : 1.80 m Antenna efficiency : 0.5

Part 3 : Thermal studies, disturbing torques

Software : SIMUSAT
Data are saved in files such as :

- *.pwr for thermal studies and energy sub system
- *.att for the attitude control subsystem
- *.orb for the orbit

3-1 Satellite data :

Enter the geometrical datas, used for the energy subsystem *(Système énergie/Configuration)* and the attitude control subsystem *(Contrôle d'attitude/Configuration)*. The shape of the satellite is a 0.8 x 0.8 x 0.8 m cube
Enter the thermal data (Système énergie/Configuration/Définition du modèle thermique).
For the thermal studies, the modelization of the satellite is a cube with 6 faces : Earth (-Z), Space (+Z), solar panels (+X,-X,+Y,-Y).
Enter the data to be taken into account for the torques calculation. *(Contrôle d'attitude / Configuration)*
The center of gravity is located at a distance of 2 cm from the geometrical center (on the X and Z axis) and the magnetic moment is equal to 10 Am^2 on each axis.

3-2 Mission data

3-2-1 Orbit and epoch

In order to analyze the results, choose the orbit and time characteristics allowing to place the Sun in the orbital plane (for example at the spring equinoxia with the right ascension of the ascending mode equal to zero)

3-2-2 Pointing

Z side direction towards the center of the Earth $\psi = 0$, $\theta = 0$, $\phi = 0$

3.4 Disturbing torques :

Analyze the order of magnitude of the disturbing torques :

- aerodynamic
- solar
- magnetic
- gravity gradient

3.5 Thermal analysis :

3-5-1 Calculate and analyse the thermal flux :

- solar
- albedo
- earth IR

The calculation may be done on a one orbit duration.

3-5-2 Calculate the temperatures

Thermical nodes suggested characteristics :
Specific heats :

- central body : 50000 J/ °
- solar panels : 2500 J/°
- upside (space) and downside (earth) panels : 1500 J/°

Conductive couplings between thermical nodes :

- links panels / central body : 4W/m²/°

Absorptivity and emissivity for some covers :

	α	ε
White paint	0.2	0.9
Black paint	0.95	0.9
MLI	0.25	0.05
Solar panels	0.8	0.9

Analyse the temperatures of the differents parts (central body, panels) and, if necessary, modify the thermal characteristics to maintain the temperature within given limits (T < 60°C for the solar panels). The simulation has to be done during 4 to 5 orbits in order to reach a good thermal balance.

Part 4 : Energy subsystem

Software : SIMUSAT

4-1 System specifications :

- non regulated architecture (24V – 38 V)
- nickel-cadmium batteries (3A.h), MAX DOD = 30 %
- maximum discharging current of the battery = Cbat/2
- power during eclipse = 3O W, Power at sunlight = 35 W
- solar cells : AsGa (4 cm X 4 cm) on the 4 lateral sides

4-2 Dimensioning the solar panels and battery

4-2-1 Battery :

Calculate the number of cells in serie (line) taking into account the tension values.
Calculate the maximum eclipse duration
What is the value of the whole charging Qd of the battery during an eclipse ?
Deduce the minimum capacity (taking into account the max DOD)
Calculate the number of cells in parallel and the predicted max DOD.
Calculate the maximum power which is needed to charge the battery.

4-2-2 Solar panels :

Taking into account the maximum temperature of the solar cells, calculate the minimum number of cells in serie (line) of a chain (the functioning point of cells must be on the left of the max power point). Moreover, this calculation will be done at the end of life in order to take into account the degradation due to space radiations (total dose of 4.10^{14} electrons of 1 Mev at the end of life)
Calculate the maximum number of cells chains which can be put on each lateral face of the satellite.
Calculate the cells filling ratio on the lateral sides of the satellite.
Calculate the sunlight which is received on each panel on the 21st of March, as the Sun is in the orbital plane, and the maximum current and power produced by the panels (at this date).

4.3 Simulation

4-3-1 Simulation specifications

- Initial date : 21/03/2003 (launch of the satellite)
- Max duration eclipse : the Sun is in the orbital plane (21st of March) $\Rightarrow$ right ascension of the ascending mode equal to zero.
- Duration of life : 3 years $\Rightarrow$ simulation date = 21/03/2006 (end of life)
- ROL ($\psi = 0$, $\phi = 0$, $\theta = 0$) :
    - o +X = velocity vector
    - o Y perpendicular to the orbital plane.
    - o Z axis towards the Earth
- Sunlight power = Power during eclipse + emitted power (mean value during the sunlight duration).

4.3.2 Working plan

Simulate the energy subsystem behaviour with the previous defined conditions.
Identify the 3 battery charging modes and verify the validity of the previous calculations (4.2.1)
Calculate the maximum available power (without charging the battery) taking into account the mean power which is used by the emission system : the energy bilan must be balanced on several following orbits (the battery must be totally charged at least 5mn before the beginning of an eclipse).
Calculate the influence of the season on the energy balance. (By defining an orbit at the date of 21st of June, when the Earth-sun distance is maximum)

Part 5: Attitude Control Sub-system

5 - 1 Introduction

The purposes of this session are :

- to define the performance requirements induced by the mission on the ACS
- to inventory the environmental disturbing torques
- to assess the adequacy of several potential architectures of the ACS
- and finally, to select an architecture for the ACS and to dimension the actuators and the captors

Of course, mass and consumption minimization constraints are relevant for the ACS. A catalogue of pieces of equipment, all of them supposed available on the shelf, is provided in appendix.

5 - 2 Performance requirements for the ACS

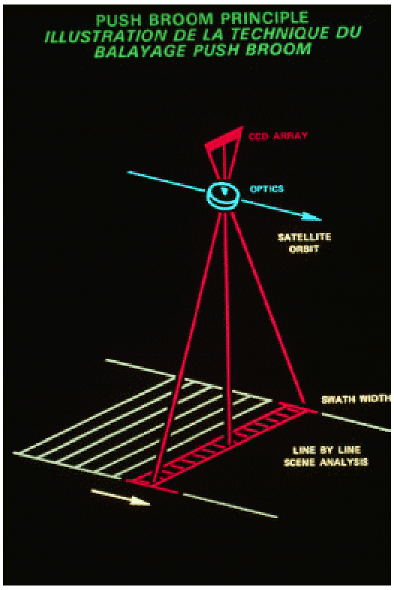
The main features of the optical mission are described here below. They drive the performance requirements, in terms of pointing and angular speed.
NB : In order to make the computations easier, the performance requirements may be evaluated at the centre of the instrumental field of view.

5-2-1 Pointing requirements

The instrument must be orientated towards the target scene. The pointing requirements rise from this constraint.
The instrumental field of view is 2000 km wide, with a 10 km margin allocated to pointing errors. In other words, at the centre of the field of view, the line of sight must be orientated within ± 5 km from the target, in the direction that is perpendicular to the satellite track. Compute the pointing requirement around the roll axis.
The position of the images along the satellite track must be controlled as well. The main error sources are the accuracy of the prediction of the satellite orbital position, the accuracy of the date of imaging, and finally the pointing accuracy around pitch axis.
Compute the pointing requirement around the pitch axis, for an allocated position error of ± 5 km.

5-2-2 Angular speed requirements

The observation mission relies on a CCD detector, in a push-broom way :
As the satellite goes by on its orbit, a set of square detectors, laid out in a line, collect one after the other the different rows of an image. The projection on ground of a single detector constitutes a pixel of the image.
The size of each square detector and the sampling period are tuned in such a way that the sub-satellite points moves of one pixel during one sampling period.



1) The sampling rate on ground (or the pixel) is 1 km at nadir. Compute the sampling period of the detectors. Remember that the orbital pulsation of a circular orbit is :

$$\omega_0 = \sqrt{\frac{\mu}{(R_E + H)^3}}.$$

The terrestrial radius R_E is 6378 km, and the terrestrial constant of gravitation μ is 398600 $km^3 s^{-2}$.
2) Image quality requirements are related to the sharpness and the sampling regularity on ground. They arise a requirement on the regularity of the size of the pixels within 10% as a maximum, row-wise and column-wise as well. Deduce the maximal acceptable angular speed around roll and pitch axes.
3) In a qualitative way, what do you think about performance requirements around yaw axis, compared to roll and pitch requirements?
Thereafter, and as a simplifying hypothesis, we will take into account the same requirements on the yaw axis than on the pitch and roll axes.

5 - 3 Environmental disturbing torques

We are on a low Earth orbit. The question is to inventory the main environmental disturbing torques and to evaluate:

- their order of magnitude on one hand (looking for raising values)
- their evolution with respect to time on the other hand (constant or periodical).

Simple computations are to be done, relying upon the provided formulas. Then the obtained results are to be correlated with those that were obtained with SIMUSAT.

5-3-1 Magnetic torque

The formula for this torque is : $T_{magn} = \overrightarrow{M} \wedge \overrightarrow{B}$

(M stands for the magnetic momentum and B for the magnetic field).
An hypothesis of a 10 Am^2 magnetic momentum, on the 3 axes is taken. The magnetic field is about 2.10^{-5} Teslas at altitudes between 800 and 1400 km.

5-3-2 Aerodynamic torque

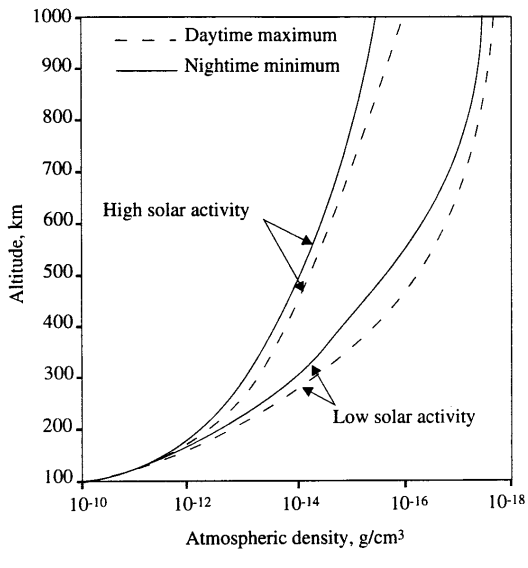
The formula for this torque is:

$$T_{aero} = ½ \rho V^2 S C la$$

(ρ stands for the atmospheric density, S for the wet surface, C for the aerodynamic coefficient, V for the linear speed of the satellite and la for the lever arm)

Take into account:
- a 1 cm shift between the centre of pressure and the centre of gravity, along the Earth pointing axis, towards the Earth
- an aerodynamic coefficient about 2,5.

The opposite figure shows the atmospheric density with respect to the altitude, in g/cm^3 (be careful, it is a non standard unit)



5-3-3 Solar torque

The formula for this torque is: $T_{sol} = P_{sol} S la$

Take into account
- the same shift between the centre of pressure and the centre of gravity than here above
- a value of about 5.10^{-6} N/m^2 for the solar radiation pressure

5-3-4 Gravity gradient torque

The formula for this torque on a circular orbit is:

$$\vec{C} = 3\omega_0^2 \cdot \vec{u} \wedge I_{sat/sat}\vec{u}$$

($\vec{u}$ stands for the unit vector associated to the Satellite-Centre of the Earth direction and Isat the inertia matrix of the satellite with respect to its centre of gravity)
Take into account:

- a 2,5 kgm^2 diagonal term of the inertia matrix for x and z axes
- 0 for the other diagonal terms

Compare the result with those obtained with the simulator. Explain the possible variations. How could you improve the situation?

5 - 4 Assessment for different possible architectures for the ACS

5-4-1 Spin stabilization

In a qualitative way, do you think that the mission is compatible with spin stabilization ? Why ?

5-4-2 Stabilization with an on-board kinetic momentum

Remind the main advantage of stabilization with an on-board kinetic momentum: to control the roll axis allows controlling the yaw axis as well, without any captor or actuator on the yaw axis.
1) On which axis would you lay the kinetic momentum? Why? Explain, with a simple reasoning, the coupling between roll and yaw axes.

2) It can be established that the performance around yaw axis is: $\psi < \dfrac{Tz}{\omega_0 \cdot Hr}$

(Tz = disturbing torque around yaw axis, Hr = wheel kinetic momentum on y axis).
Compute the wheel kinetic momentum that would be required to ensure stabilization with an on-board kinetic momentum. What is your conclusion?

5-4-3 Gravity gradient stabilization

A gravity gradient stabilization consists in adding a mast along the z-axis, with a mass at the end of the mast. In this way, the inertia around the x and y axes are significantly increased, but the inertia around the z-axis remains almost unchanged.
Using the same formula than here above, with the following hypotheses:

- a small pointing error around roll or pitch axis
- diagonal terms of the inertia matrix negligible (to be checked afterwards)

Demonstrate that the gravity gradient torque acts as a drawback torque around roll or pitch axes.
Dimension the mast that would ensure a gravity gradient stabilization. What is your conclusion?

5-4-4 Three axes stabilization

A three axes stabilization is then necessary. Are any secondary actuators needed? Why?

5 - 5 Actuators dimensioning

Propose a configuration for the actuators.
Simplifying hypothesis : whatever the orbit, consider that the magnetic field provides an average unloading capacity of 0.03 Nms/Am^2, during one orbital period.

5 - 6 Captors dimensioning

Propose a configuration for the captors, and a lay out on the satellite.

5 - 7 Budgets

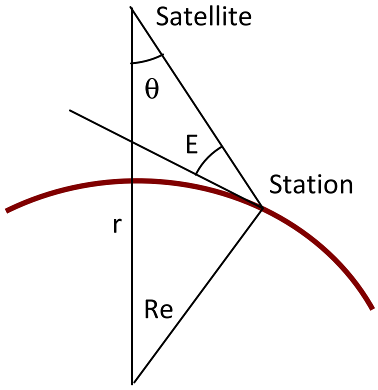
Assess pointing, mass and power budgets of the satellite.
Is the proposed architecture compatible with the resources of the satellite?
If not, how could you reiterate on this architecture?

Annexe 1 : Form for RF link budget calculation

Salellite antenna half aperture angle :

$\sin\theta = \cos E . Re/r$

$\theta_{3db} = 2.(\theta + \Delta\theta)$ (θ_{3db} is the total aperture)



Satellite antenna gain :

$Gmax = \eta\ (70\pi/\theta_{3db})^2$ θ in degrees

$G(\theta) = G_{max} - 12.(\theta / \theta_{3db})^2$ G, G_{max} in dB

Space loss :

$LFSL = (4\pi\ R/\lambda)^2$

Ground station antenna gain :

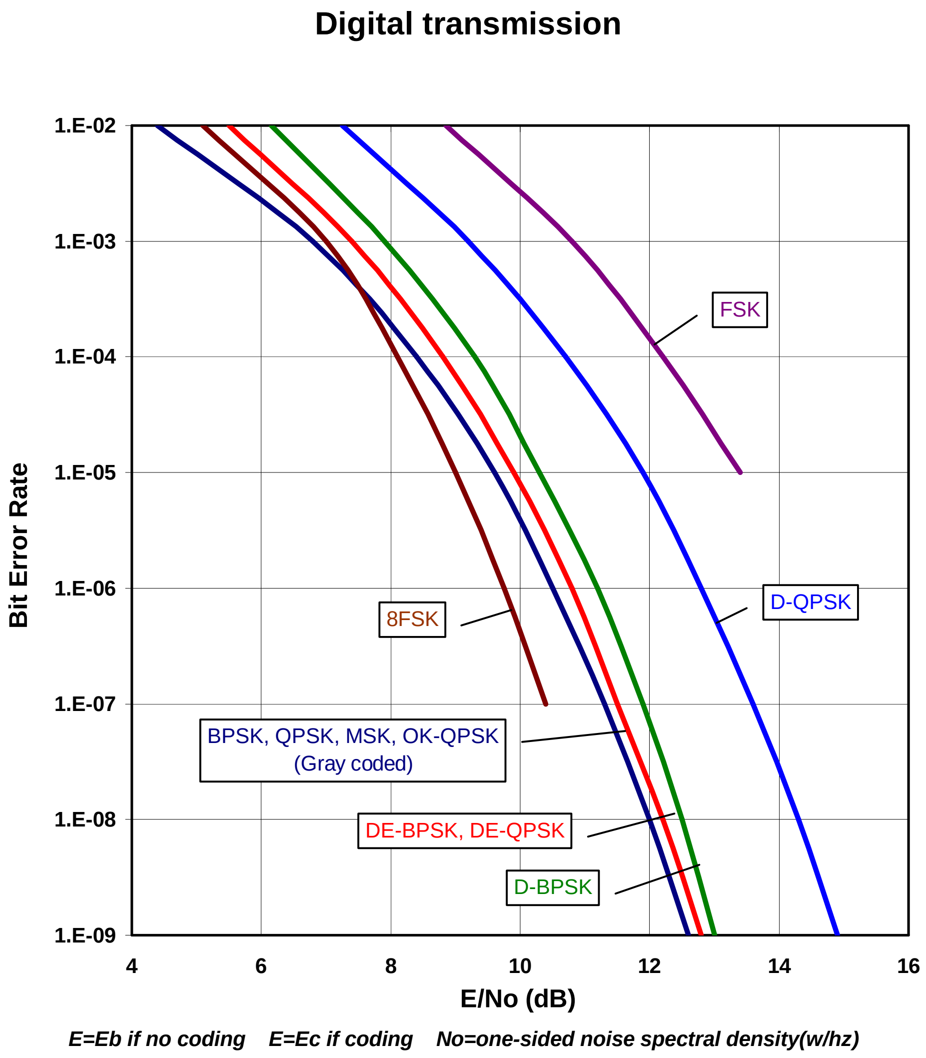
$GR = \eta\ (\pi\ D/\lambda)^2$ $\theta_{3db} = 70\ (\lambda / D)$ $Perte = -12.(\Delta\theta / \theta_{3db})^2$

Power at receiver input :

$C/N0 = Eb/N0 + 10\log(\text{data rate})$ data rate in bits/s, C/N0, Eb/N0 en dB

$C/N0 = PT\ GT\ (GR/TR)\ (1/K) / L / LFSL$

K	Boltzman constant (K=-228,6dB)
PT	Emitter power
GT	Emitter gain
GR	Receiver gain
TR	Receiver noise temperature
L	Miscellaneous losses (atmosphere, rain etc…)
LFSL	Space loss


Digital transmission
Bit Error Rate
1.E-02
1.E-03
1.E-04
1.E-05
1.E-06
1.E-07
1.E-08
1.E-09
4
6
8
10
12
14
16
E/No (dB)
FSK
D-QPSK
8FSK
BPSK, QPSK, MSK, OK-QPSK
(Gray coded)
DE-BPSK, DE-QPSK
D-BPSK
E=Eb if no coding E=Ec if coding No=one-sided noise spectral density(w/hz)

Annexe 2 : Equipments list for attitude control system

A2-1 Actuactors

Wheels :

Max couple (mNm)	Max angular momentum (Nms)	Mass (kg)	Max absorbed power (W)
25	0,4	2,2	9
10	2,2	2,9	28
5	0,12	0,75	3,8
5	0,04	0,6	3

Notice :
The power used by a wheel is proportional to the torque it provides through its angular momentum. The maximum power consumption is a rare case of operation (for example, maximum positive angular momentum znd mzximum negative torque).

Magnetotorquers :

Magnetic momentum (Am^2)	Length (cm)	Mass (kg)	Max absorbed power (W)
12	41	0,4	0,7
12	28	0,5	0,9
30	57	0,7	0,9
110	41	4,5	5,4

Gravity gradient boom :

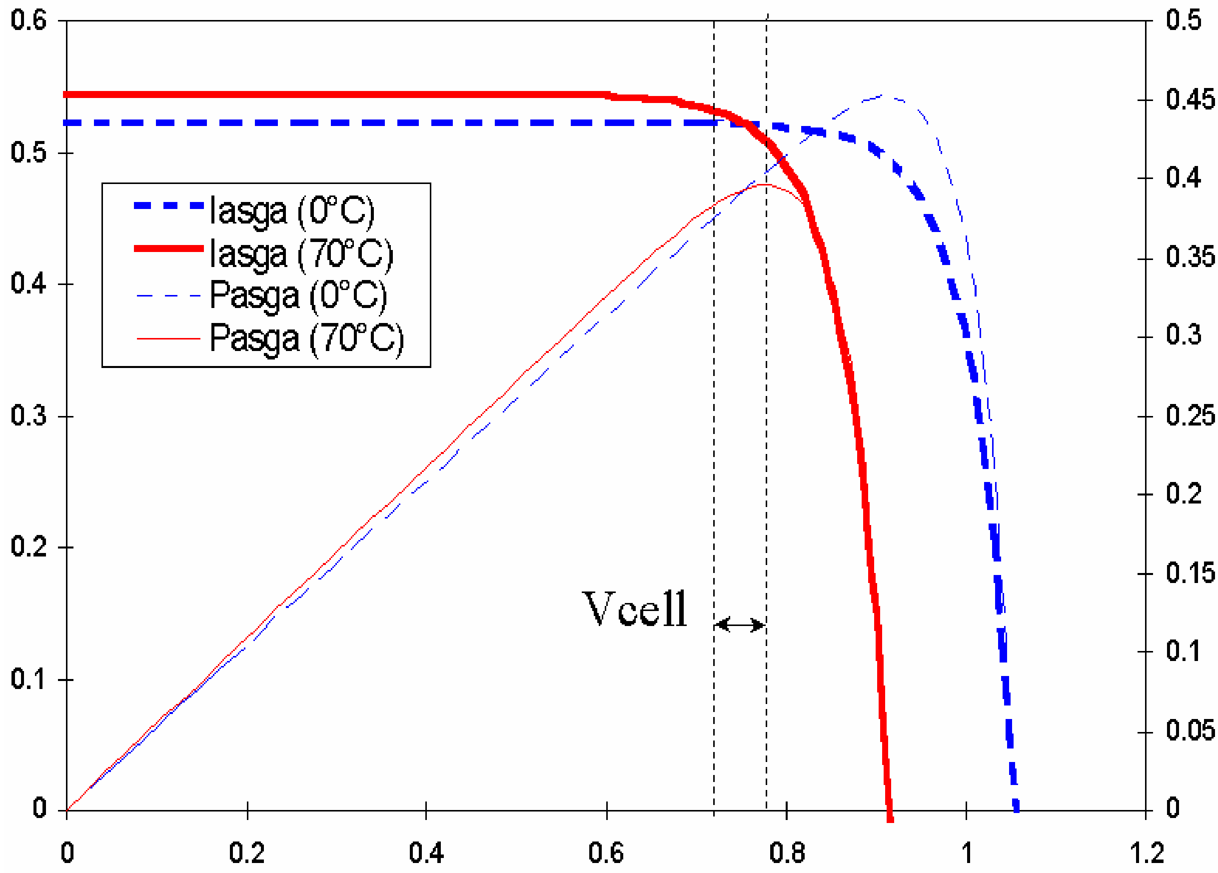
For exemple, on UoSat satellite there is a 3 kg mass 6 meters from the satellite.

A2-2 Sensors

Type	Function	Accuracy	Frequency (Hz)	Field of view	Mass (kg)	Power (W)
Gyrometers unit	Angular rates 3 axes	0,01°/s	10	-	1	4
Analog sun sensor	Angles 2 axes	0,3°	1	128°x168°	0,2	0,3
Analog sun sensor	Angles 2 axes	1°	1	180°x180°	0,07	0
Digital sun sensor	Angles 1 axe	0,02°	2	10°x10°	2	5
Magnetometer	Angles 3 axes	1°	1	-	0,35	0,3
Earth sensor	Angles 2 axes (roll/pitch)	0,13°	2	10°x10°	1,3	0,8
Star tracker	Angles 3 axes	0,05°	4	28°x28° Clearances : Earth 30° Sun 40°	1,5	6

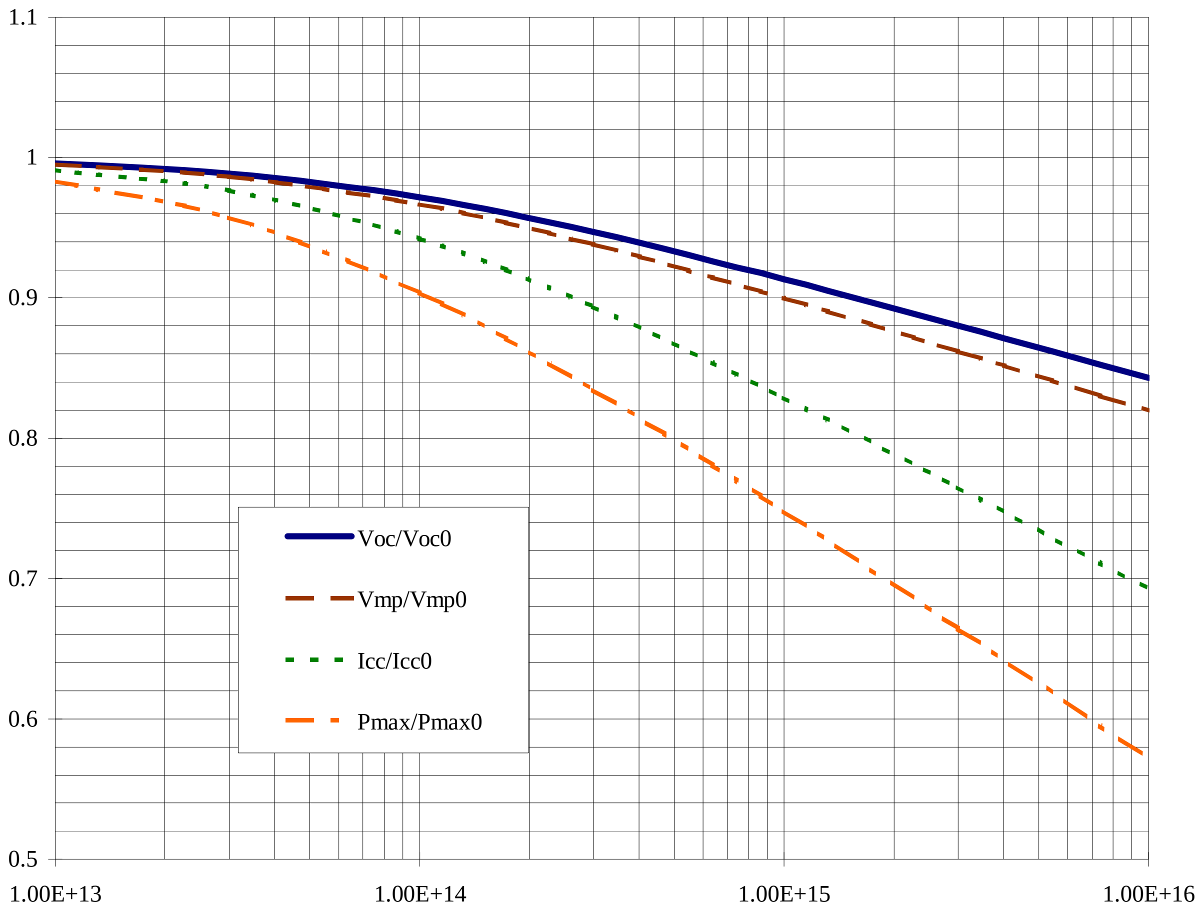
Annexe 3 : Batteries and solar cells data

A-3-1 Solar cell data



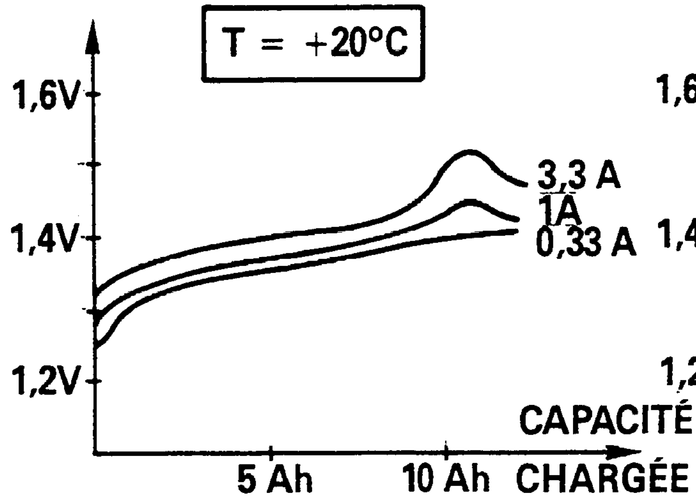
Isc(bol) at 28°C	0.53	A
Imp(bol) at 28°C	0.5	A
Voc(bol) at 28°C	1	V
Vmp(bol) at 28°C	0.86	V
dIsc/dT(eol)	3.00E-04	A / °C
dImp/dT(eol)	2.50E-04	A / °C
dVoc/dT(eol)	-2.00E-03	V / °C
dVmp/dT(eol)	-2.00E-03	V / °C

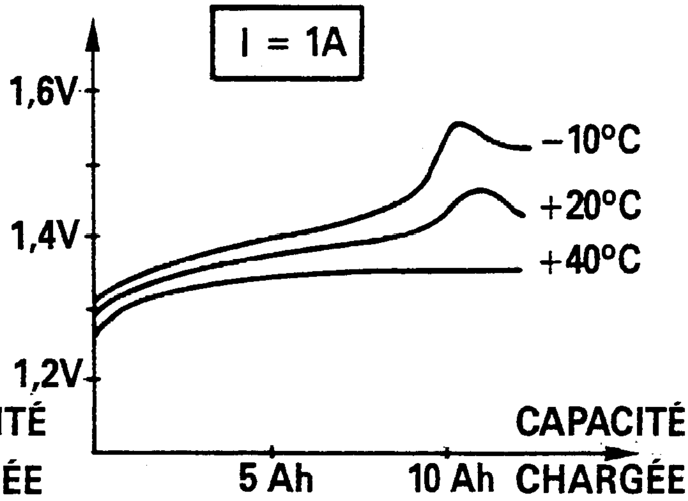
A-3-2 Solar cell degradation due to cumulated radiations



A3-3 Exemple of charge/discharge curve for a Ni/Cd accumulator

A3-2-1 : Charge





A3-2-2 : Discharge