Chapter 2 : Mission envelope

CHANGE TRACEABILITY Chapter 2

Here below are listed the changes between issue N-2 and issue N-1:

Here below are listed the changes from the previous issue N-1:

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N°§	Sentence	Planned Resolution
§2.2.1	Notice : The lower limit of the flight domain is estimated to 20 deg (TBC).	

Chapter 2: Mission envelope

The first part of this chapter deals with the PROTEUS mission capabilities: the potential launch vehicles, the achievable orbits, the possible pointing modes with the associated orbit kinds. The second part is dedicated to the User in order to help him analyse his mission at a first level (phase A), choose an orbit type, and assess the required propellant capacities in order to achieve the necessary orbital manoeuvres. In any case, the User is strongly encouraged to contact either ALCATEL SPACE or CNES in order to detail the mission further on, thus benefiting from the greatest experience in mission analysis and design.

2.1 LAUNCH VEHICLES

2.1.1 POTENTIAL LAUNCH VEHICLES

The PROTEUS platform is compatible with the following launch vehicles: Ariane 5, Athena 2, Cosmos, Delta 2, LM-2D, PSLV, Rockot, Soyuz and Taurus. Their main characteristics are summarised in Table 2.1-1, for information only. The User shall refer to the corresponding launcher manual applicable at its study moment

This list takes into account all developed launch vehicles in the 500 to 1000 kg class. It could be updated if new launch vehicles became available. Assuming this large panel of compatible launch vehicles, it will be easy to adapt PROTEUS with other launch vehicles like Ariane 4 (Europe), Atlas (USA), etc...

Launch vehicle	Country / Launch sites	Launch service provider	First flight	Usable volume diameter (mm)	Comments
Ariane 5	Europe/Kourou	Arianespace	1998	4570 or 4800	Multiple launch
Athena 2 (LMLV2)	USA/Cape Canaveral, Vandenberg	Lockeed Martin	January 1998	1984	
Cosmos	Russia/Plesetsk	Cosmos international	1970	2200	
Delta 2 (upper position)	USA/Cape Canaveral, Vandenberg	Boeing	1997	2743 (upper position) 2360 (lower position)	Dual launch
M-2	China/Japan	GWIC	1992	2360/1715	
PSLV	India/Shriarikota	ISRO	1993	2900	
Rockot	Russia/Plesetsk	Eurockot	1990	1983	Dual launch foreseen
Soyuz	Russia/Baikonur, Plesetsk	Starsem	in the sixties	3395	
Taurus	USA/Cape Canaveral, Vandenberg, Wallops	OSC	1994	2055	Taurus versions equipped with a 2.34 m fairing *

Table 2.1-1: Main launch vehicles compatible with PROTEUS platform

* The Taurus, Taurus XL, Taurus XLS can be equipped with a 2.34 m diameter fairing.

2.1.2 LAUNCH VEHICLE ADAPTER

The standard PROTEUS bottom frame has a standard interface with 60 M5 screws on a 943,6 mm diameter.

The launch vehicle adapter is the interface hardware between the platform bottom frame and the launch vehicle. It is bolted on the platform and is maintained by a clamp band on the launch vehicle side. At the launch vehicle/satellite separation, the clamp band opens and the satellite (with the launch vehicle adapter staying fixed at the platform) separates off from the launch vehicle (cf. chapter 1.5). The launch vehicle adapter is a thick ring with a 943.5 mm diameter (for Delta 2, Taurus, Athena 2); but it can be different depending on the launch vehicle (interface diameter). ALCATEL SPACE and CNES are logically responsible for the launch vehicle adapter mechanical and thermal design; it shall be negotiated with the launch vehicle authorities case by case.

2.2 ACHIEVABLE ORBITS

2.2.1 FLIGHT DOMAIN

The allowed orbits are limited by several constraints detailed hereafter. The resulting orbit envelope is shown on Figure 2.2-1.

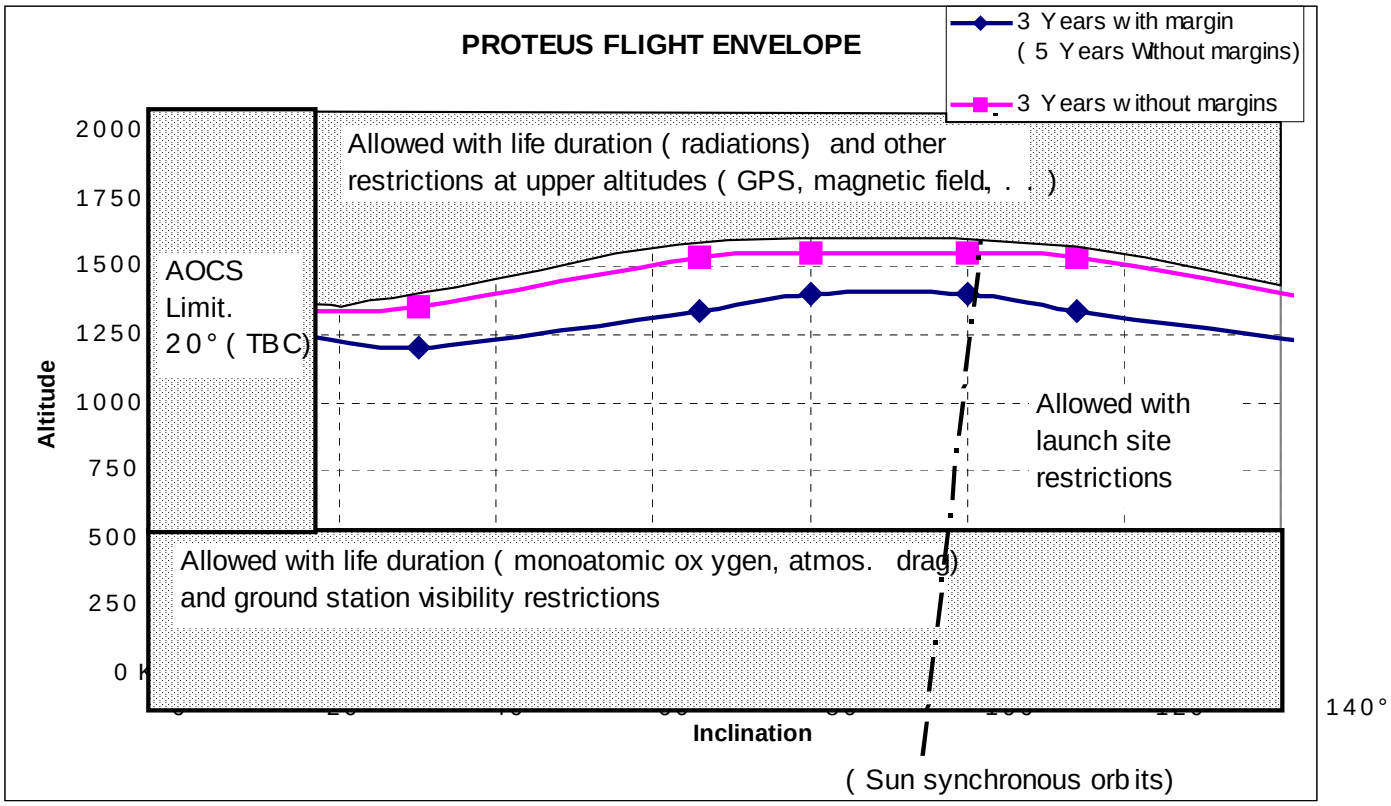

Figure 2.2-1: Orbit envelope

Notice : The lower limit of the flight domain is estimated to 20 deg (TBC).

2.2.2 CONSTRAINTS RELATIVE TO ORBIT ALTITUDE

2.2.2.1 Environment

The lower altitudes limit of the flight domain is determined by the atmospheric drag and the mono-oxygen effects.

The atmospheric drag is usually compensated by periodic manoeuvres in order to maintain the altitude and/or the semi-major axis or the orbit. In this area, the propellant capacity of the satellite limits the orbital manoeuvres capacity and has a direct effect on the satellite lifetime.

The mono-atomicoxygen contained in the upper atmosphere reacts with satellite materials, especially Kapton and Silver and causes the erosion and the weakening of these materials.

MLI external layer is very sensitive to mono-atomic axygene dose. Standard PROTEUS design allows minimum altitude of about 600 km.

The mono-atomic oxygene dose specified for the Proteus generic Star Tracker is such that STR is not designed for missions at an altitude under 600 km.

Standard Solar Arrays (without protection against atomic oxygen) can stand environment met at altitude around 600 km. With a coating protection on the solar arrays, the flight domain can cover lower altitudes.

Due to the exponential relationship between these effects and the altitude, a minimum altitude of 600 km is recommended.

The upper altitudes (above 1500km, roughly) are sensitive to radiation, LET (Linear Energy Transfer) and Trapped Proton fluxes.

The sizing has been performed taking into account a cumulated radiation dose over 5 years on a reference orbit : 1336 km/66° (without margins), cf. red curve on figure 2.2-1.

2.2.2.2 Global Positioning System (GPS) constraint

A GPS is used on board to have a time, position and velocity reference.

High altitudes limit the GPS constellation satellites visibility, but GPS satellites are far above the satellites using the PROTEUS platform: 20 000 km versus about 1500 km and therefore are not a limiting constraint for PROTEUS nominal flight envelope.

A specific study (on request/orbit dependent) will be done for circular orbits or elliptical orbits with an apogee higher than 1500 km or in case of inertial pointing (mission dependent).

2.2.2.3 Attitude and Orbit Control System (AOCS) constraint

The PROTEUS platform AOCS uses magnetic torquers for reaction wheels desaturation in normal mode and attitude acquisition in Safe Hold Mode. This equipment can generate a torque perpendicular to the Earth magnetic field. Unfortunately, at the equator, the Earth's magnetic field is nearly perpendicular to the equatorial plane, so one axis becomes poorly controllable for low inclination orbits. Below 20° inclination and depending on satellite inertia, the mission feasibility has to be checked on a case by case basis.

The magnetic field strength decreases with altitude and an other limitation could appear for very high orbits. As for GPS constraint, a specific study will be led for circular orbits or elliptical orbits with an apogee higher than 1500 km.

2.2.2.4 Telecommunication constraints

The downlink and ground command budget has been evaluated for a circular orbit with a 1336 km altitude and a minimum elevation angle equal to 10 deg, which is nearly the highest altitude allowed.

For higher orbits and the same site angle, the satellite to ground station distance is greater, but the visibility duration per day with a single ground station increases. RF link budget performances could be maintained assuming a higher minimum elevation angle and/or a data rate reduction.

For lower altitudes, the visibility duration per day decreases dramatically and a second ground station could be necessary (mission dependent on TM flow and ground station minimum elevation constraint)

2.2.3 CONSTRAINTS RELATIVE TO ORBIT INCLINATION

The achievable orbits are limited by the launch azimuth allowable from a given launch site. PROTEUS can correct the launch vehicle injection errors, but manoeuvres to change significantly the orbit inclination are propellant consuming. The achievable orbits depending on the launch vehicles and launch sites are shown on Figure 2.2-2.

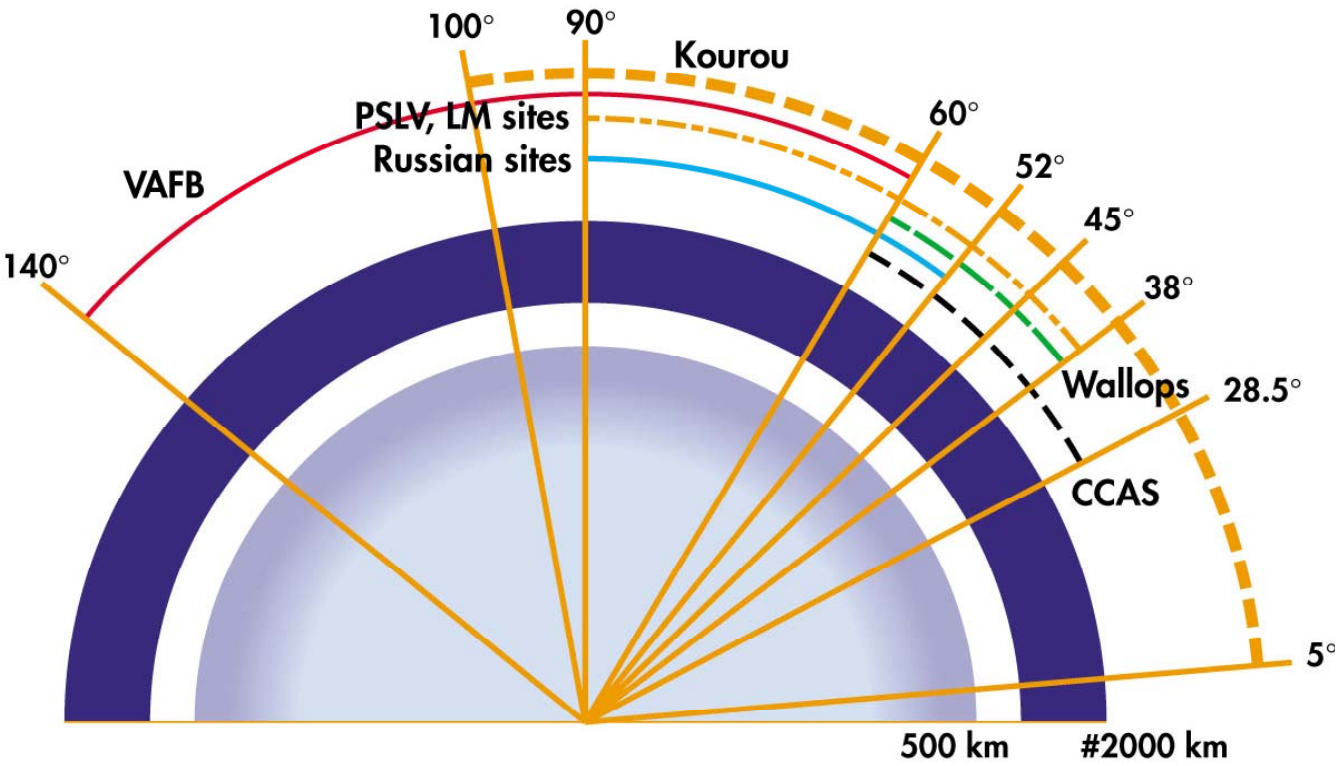

Figure 2.2-2: Achievable orbits versus launch sites and vehicles

2.3 IN FLIGHT ORIENTATION AND POINTING

2.3.1 ACHIEVABLE POINTING

The standard PROTEUS platform allows five main kinds of pointing:

Earth pointing with a fixed yaw on a sun synchronous orbit or on a low inclination orbit (about 20 deg),

Earth pointing with a yaw steering on every orbit,

inertial pointing,

sun pointing on a sun synchronous orbit,

other non standard pointing modes which can be studied upon request.

The possible manoeuvres around these pointings shall be compatible with the platform reaction wheels torque capacity and the moment of inertia acceptable for the reaction wheels.

The AOCS dynamic range is compatible with the following pointings : Earth pointing, Nadir pointing, track pointing, yaw steering, inertial pointing.

Table 2.3-1 summarises the conceivable satellite pointings considering the PROTEUS flight domain.

POINTING	EARTH CENTERED				INERTIAL POINTING
MOVMENT AROUND POINTING AXE	Yaw steering	None			None
ORBITAL PLANE	Arbitrary	Low inclination	Sun synchronous		Inertial
ASCENDING OR DESCENDING NODE LOCAL TIME	N.A.	N.A.	9H30 to 14H30 21h30 to 2h30	5H to 7H & 17H to 19H	N.A.

Table 2.3-1 : PROTEUS satellites pointings

The pointing chosen according to the mission needs imposes:

the satellite orientation in routine mode, so the associated mechanical configuration of the payload and the set up of some equipment components such as the star trackers, the antennas.

(satellite lay out, centering, inertia)

thermal limitations for the satellite,

solar arrays efficiency and so the power limitations for the satellite and the payload,

the AOCS approach,

telecommunications link constraints.

For PROTEUS based missions, these constraints lead to consider preferably the standard satellite configurations described hereafter.

2.3.1.1 Earth pointing / fixed yaw / sun synchronous

The satellite configurations presented in this paragraph can be applied for fixed yaw but also for angular variations of a few degrees around the pointing axis.

2.3.1.1.1 Sun synchronous orbits

2.3.1.1.1.1 Definition

The orbital plane keeps a constant angle with the Earth-Sun direction during the year. Under some inclination and altitude conditions, the orbital plane drift is equal to the Earth movement around the Sun (0.986 deg per day).

Sun synchronous orbits allows to get:

a constant solar local time at a reference location, which determines a constant illumination (if the seasonal variations are not considered),

a sweeping over the whole surface of the Earth; the orbit is nearly polar (orbit inclination around 98 deg).

They are usually circular with the frozen perigee and at a constant altitude. This orbit kind is typically selected for Earth observation.

2.3.1.1.1.2 Satellite pointing

For PROTEUS, the possible sun synchronous orbits are as follows:

sun synchronous orbits with an ascending or a descending node between 9:30 am and 2:30 pm or between 9:30 pm and 2:30 am.

sun synchronous orbits with an ascending or a descending node close to 6 am or to 6 pm. In this case, the nominal satellite configuration is classical.

For the sun synchronous orbits, two satellite pointing modes are possible:

the +Zs axis can be oriented towards the Earth; it is called « nominal satellite configuration »

the +Xs axis can be oriented towards the Earth; it is called « vertical satellite configuration ». This second configuration shall be negotiated. Currently, the main identified critical points for this configuration are :

TMTC link

Thermal control (mainly battery and DHU)

Lead to important attitude slew for orbit manoeuvers and loss of pointing

GPS antenna field of view (only on $-Z_s$ side and $-X_s$ sides)

STA accommodation

Sun synchronous orbits with an ascending or a descending node between 9:30 am and 2:30 pm or between 9:30 pm and 2:30 am

a) Nominal configuration

The nominal satellite configuration (cf. Figure 2.3-1) is such that the satellite +Zs axis points towards the Earth. The +Ys axis is aligned with the solar array rotation axis and it is oriented such that the Sun is in the - Ys hemisphere. The +Xs axis is aligned with the launch vehicle axis and is oriented following the velocity or anti-velocity direction depending on local time. The axis direction is imposed by the right handed orthogonal reference frame.

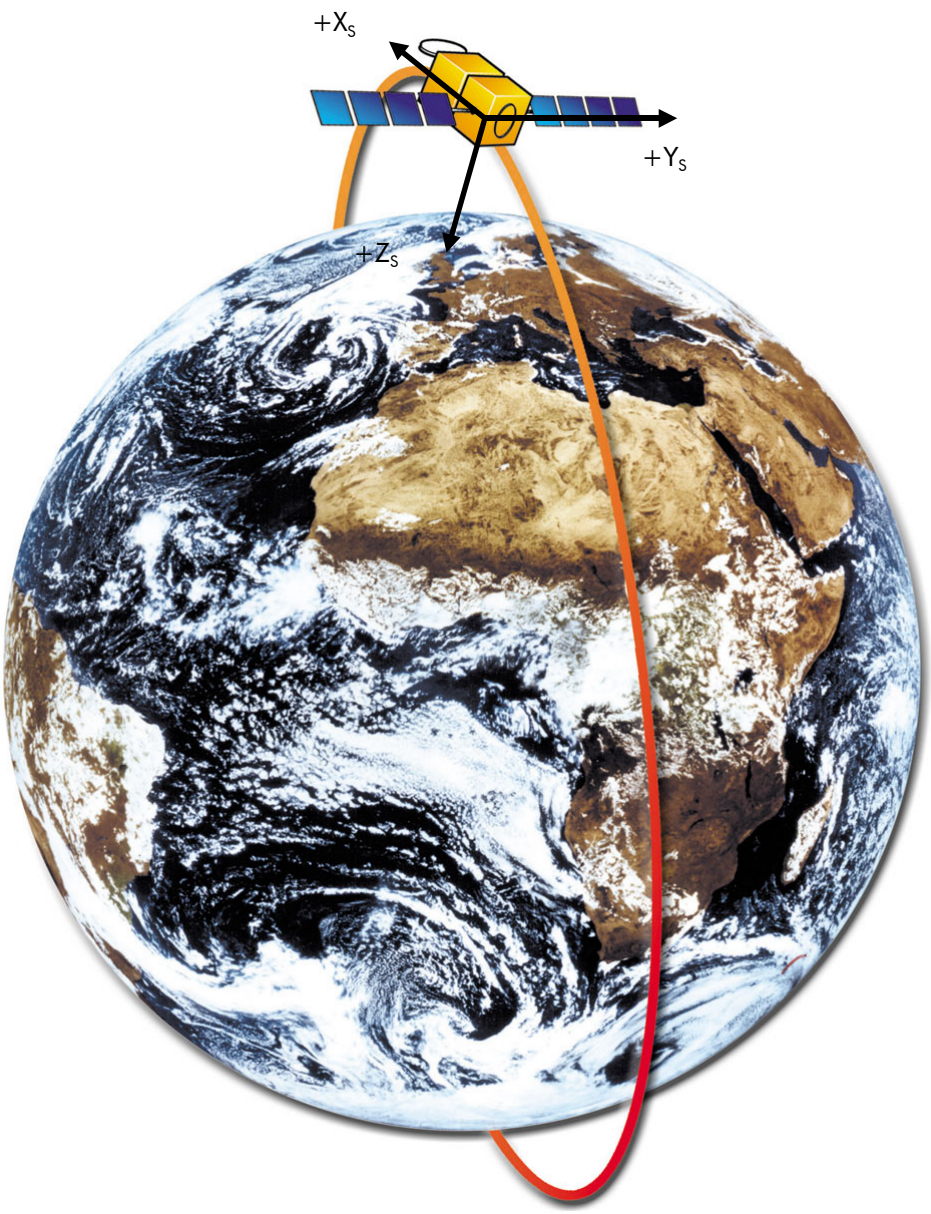

Figure 2.3-1 : Nominal satellite configuration for the sun synchronous orbits (orbits around noon or midnight on the drawing)

b) Vertical configuration

For this orbit kind, the payload accommodation with the platform is such that the satellite adopts the vertical configuration (cf. Figure 2.3-2); that means the +Xs axis is pointed towards the Earth. The +Ys axis is aligned with the solar array rotation axis and it is oriented such that the Sun is in the - Ys hemisphere. The +Z axis is aligned with the launch vehicle axis and is oriented following the velocity or anti-velocity direction depending on local time. The axis direction is imposed by the right handed orthogonal reference frame.

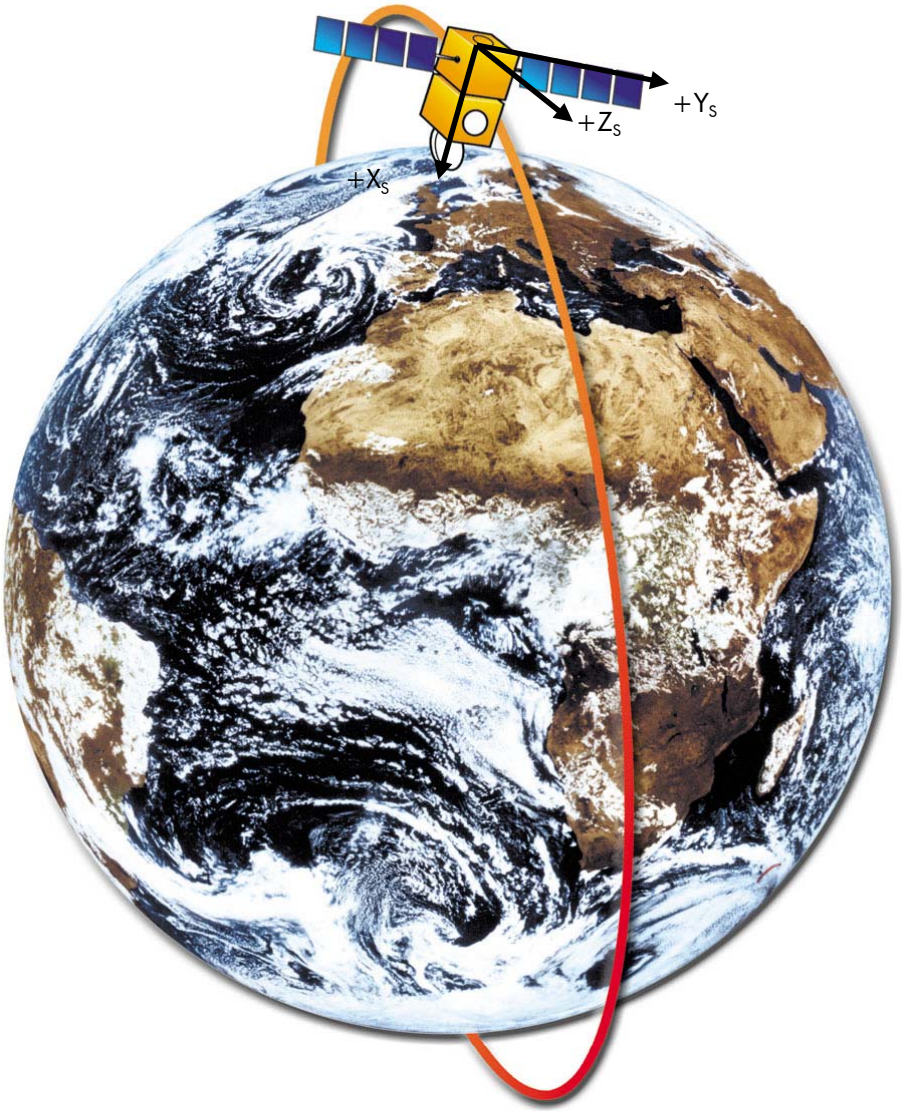

Figure 2.3-2 : Vertical satellite configuration for the sun synchronous orbits (example : orbits around noon or midnight on the drawing)

Sun synchronous orbits with an ascending or a descending node from 5 am to 7 am or 5 pm to 7 pm.

In this case, the Sun/orbital plane angle is equal to the orbit inclination (nearly 98 deg), plus a seasonal variation equal to the solar declination (+23.5 deg maximum at solstice).

The satellite will fly with the +Ys axis parallel to the orbital speed and preferably the Sun is in - Xs hemisphere to have a configuration similar to the Safe Hold Mode pointing.

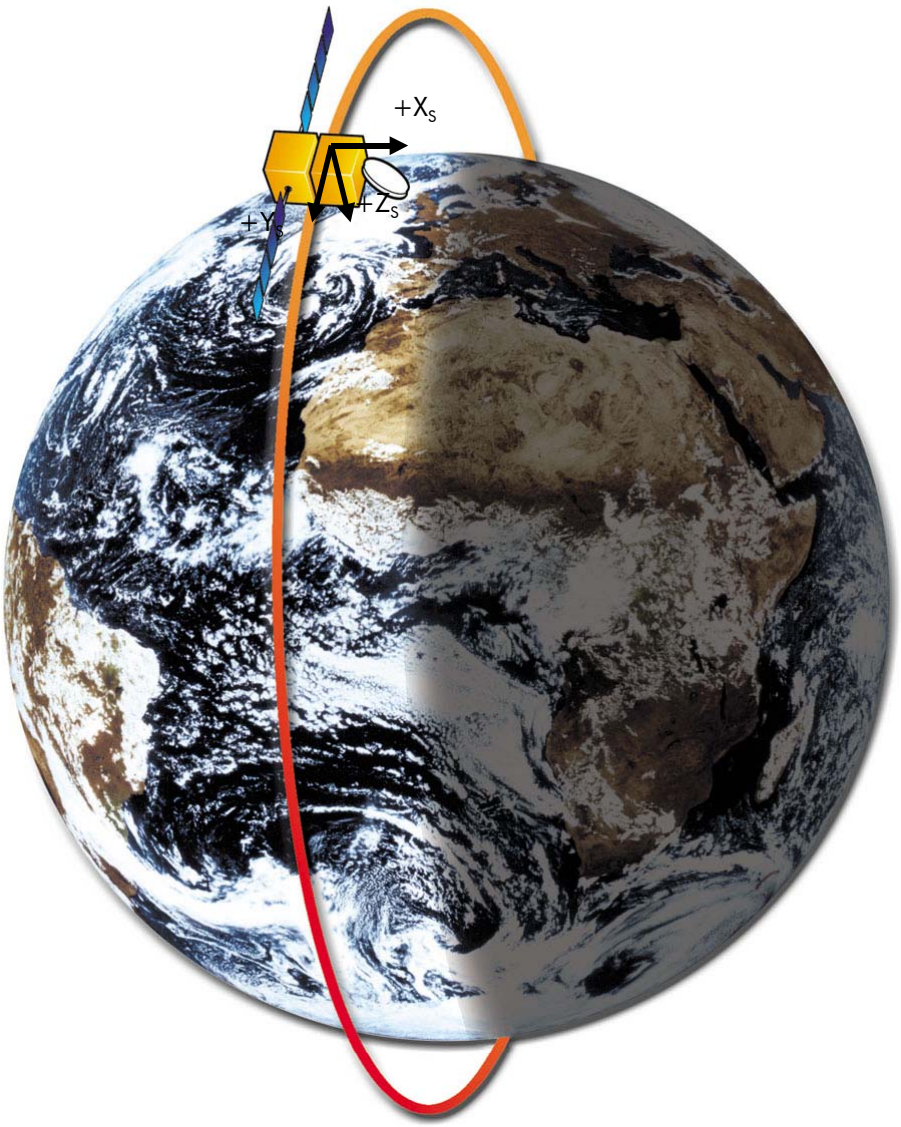

Figure 2.3-3 : 6 am or 6 pm Sun synchronous orbits

2.3.1.2 Earth pointing / fixed yaw / Low inclination orbits (About 20 deg)

Low inclination orbits drift around the polar axis (several degrees a day) and the sun/orbit plane angle varies in the following interval [-(orbit inclination + solar declination);+(orbit inclination + solar declination)]. It is necessary to make a 180° slew around the yaw axis (Zs) when the sun crosses the orbital plane to maintain the sun in one satellite hemisphere. A three axis pointed satellite flies with the +Xs axis along the orbital velocity during half the time and with the +Xs axis in the opposite direction other wise.

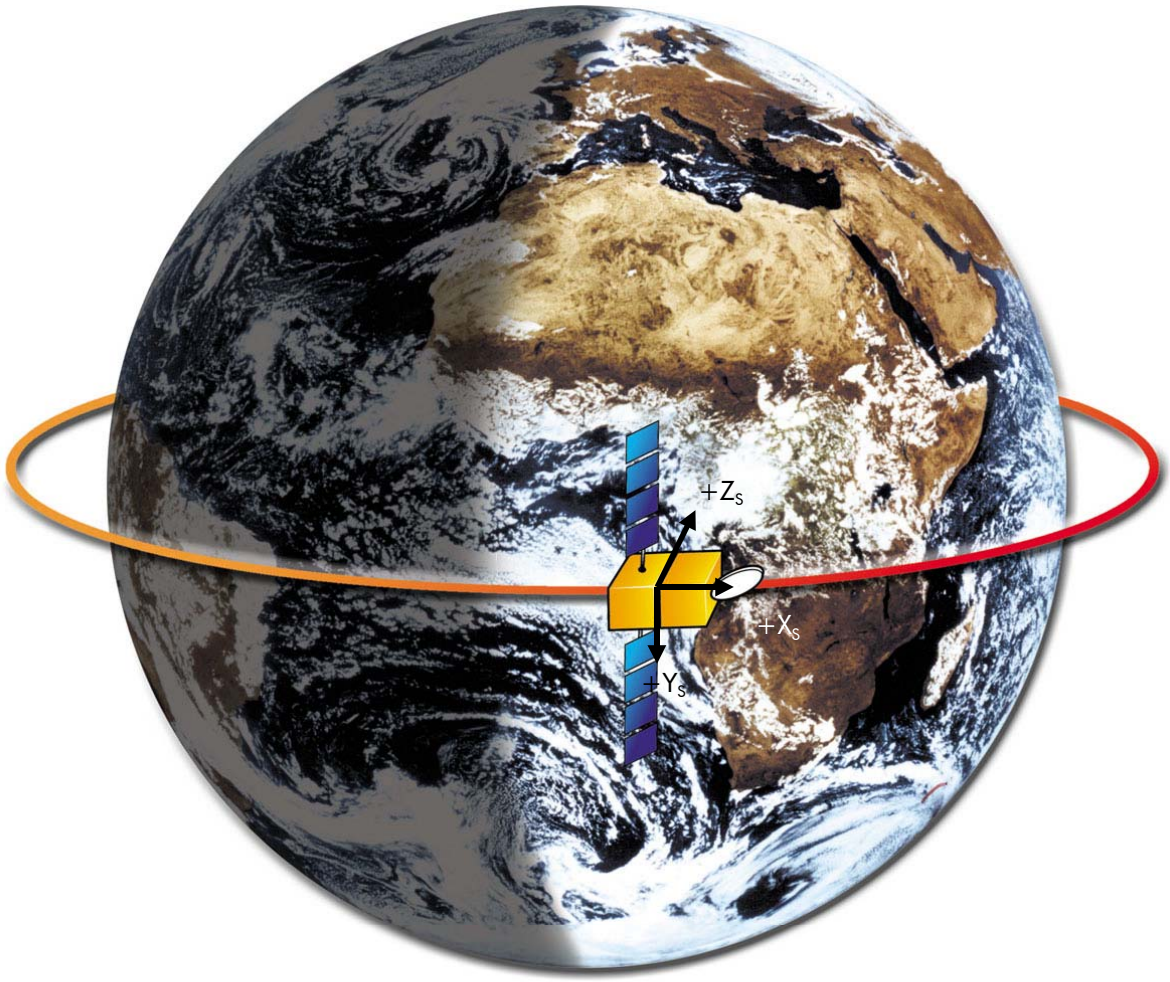

Figure 2.3-4 : low inclination orbits

2.3.1.3 Earth pointing / free yaw / all orbits

For Earth pointing, the +Zs satellite axis remains pointed towards the Nadir axis. In the free yaw case, the satellite attitude follows a yaw steering to orient the solar arrays towards the sun (cf. Figure 2.3-5); the aim is to avoid thermal and power losses.

The satellite rotates around the Earth direction to maintain the Sun in the (Xs, Zs) plane, with the Sun in the -Xs hemisphere (configuration close to the Safe Holdmode). Then the solar arrays are continuously oriented towards the Sun, following a near sinusoidal movement along the orbital period.

When the Sun angle versus the orbital plane is less than 20 deg (typical value), the yaw steering movement is stopped and the satellite follows a three axis pointing profile with +Xs or -Xs oriented towards the orbital speed.

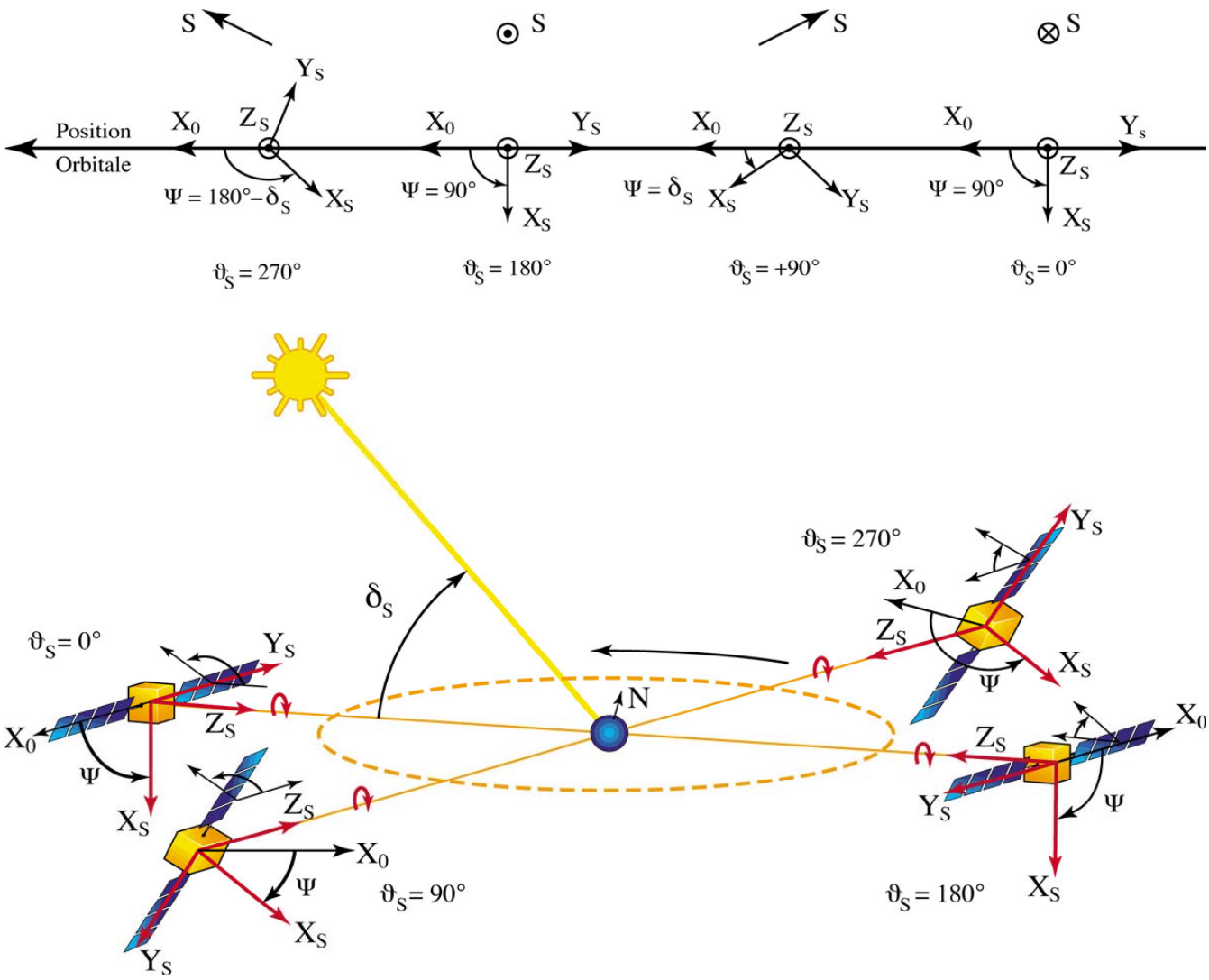

Figure 2.3-5 : Yaw steering

The yaw angle theoretical evolution versus Sun/orbital plane is shown on Figure 2.3-6.

Figure 2.3-7 and Figure 2.3-8 show the yaw angle and the solar array position given by the implemented approximated laws.

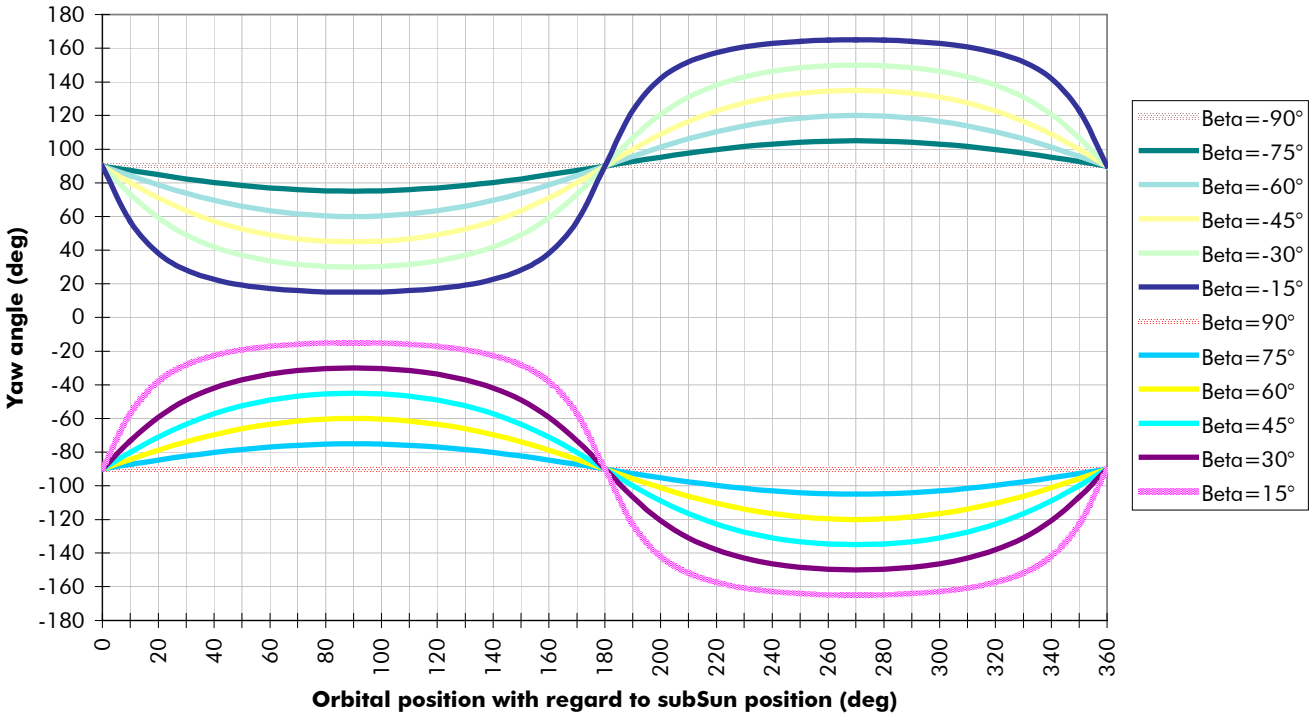

Figure 2.3-6: Theoretical evolution of the yaw angle along the orbit

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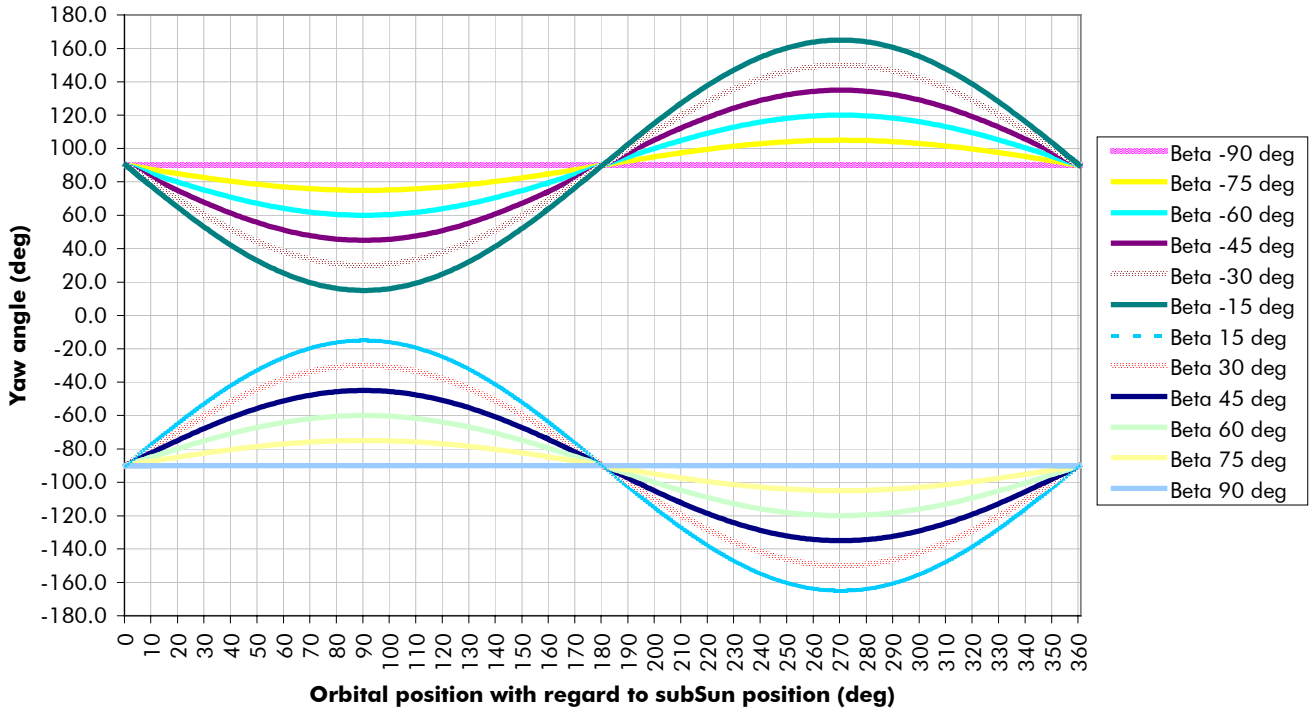

Figure 2.3-7: PROTEUS evolution of the yaw angle along the orbit

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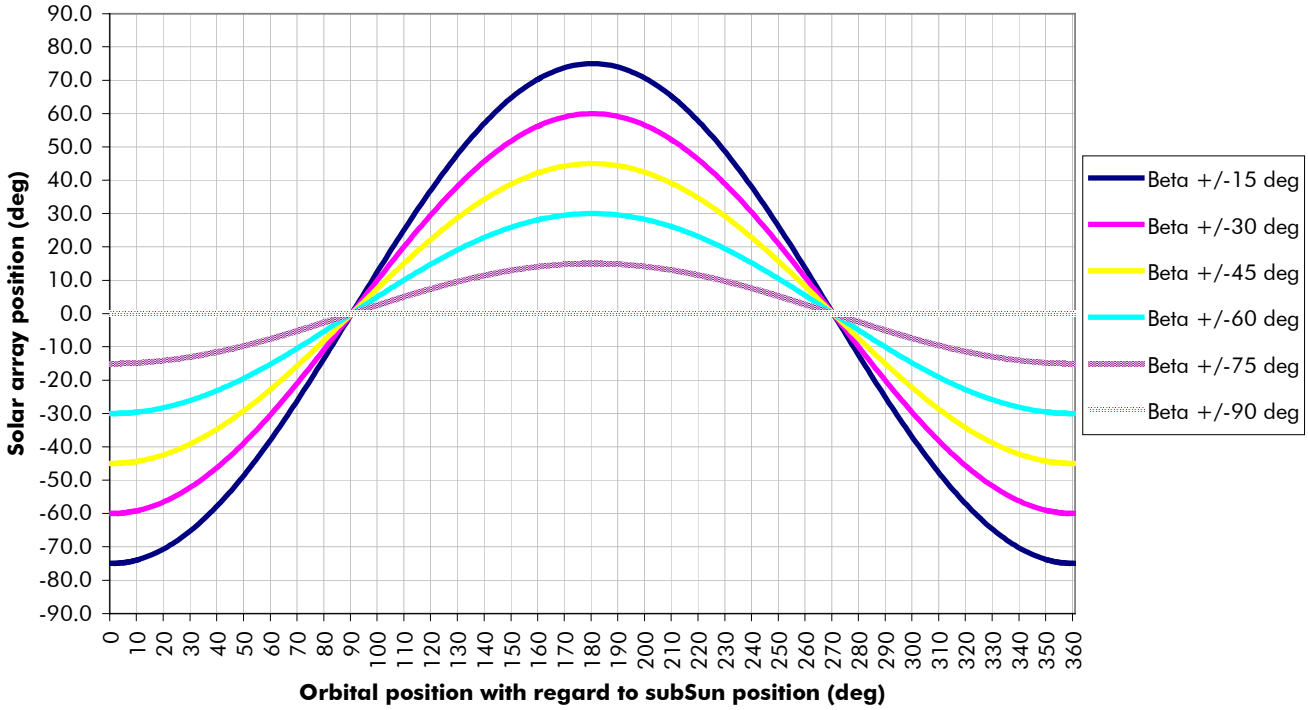

Figure 2.3-8: PROTEUS solar array position

2.3.1.4 Inertial pointing

In order to avoid Earth shadowing, the polar inertial orbit is recommended.

For inertial missions, the payload will have its field of view boresight towards +X satellite axis.

It is supposed that these missions do not impose attitude around X axis (inertial, but 2 axis pointing) and the Sun could remain in the -X satellite hemisphere. Then, the attitude around the X axis will be chosen such that the Sun is near (X,Z) plane to minimize thermal constraints and optimize the power budget:

the Sun could be placed perpendicular to the Solar Array by a rotation of these Solar Arrays around these axis.

the Sun could remain in the +Z hemisphere to minimize solar aspect on the battery radiator, performing a 180° slew around +X when the Sun crosses the (X,Y) plane (solar aspect up to 10 to 20° maximum on the -Z face tolerated).

Star trackers orientation will be optimized (near payload boresight) to avoid Earth, Sun, Moon, planets, stars holes perturbations.

These limitations can be reviewed on a case by case analysis due to the difficulty to define a generic inertial pointing mission and associated constraints.

2.3.1.5 Sun pointing

Sun pointing can be considered as a particular case of inertial pointing, with the Sun along the +Xs or -Xs direction. In order to avoid Earth shadowing (Sun eclipse), a Sun synchronous orbit with a 6 am or 6 pm node and with a high altitude is recommended. The pointing direction (Xs axis) is oriented between 0 and 32 deg from the perpendicular to the orbit.

2.3.2 POINTING COMMAND

The pointing command is defined by time tagged commands. These commands are defined by:

the time T_0 from which the time tagged command shall be applied

the attitude quaternion at T_0 (the command shall be continuous with the preceding command)

the quaternion evolution rate from T_0 . This evolution rate is described by:

a flag indicating if the evolution is versus an Inertial Frame or the Local Orbital Frame.

a flag indicating if the evolution is described by a polynome defined versus T_0 or a Fourier serie defined versus the orbital position ωt.

the degree of the polynomial or the Fourier Serie (maximum value 4).

the polynomial or Fourier serie coefficients along the three axes.

the Solar array commanded evolution. This command is defined as the sum of a linear function and a Fourier serie of degre 1:

$$a_0+a_1t+b_1\sin \omega t +c_1 \cos \omega t$$

These commands are applied until a new time tagged command replaces them (infinite duration possible).

For instance, a geocentric mission has a command with all coefficients at 0, so this command is valid for the whole mission duration.

2.3.3 POINTING AND RESTITUTION PERFORMANCES

The pointing requirement is mission dependent. It consists in two main items :

Alignment difference (bias, thermoelastic…) between payload boresight and STA interface plane (mission and payload dependent)

Pointing performance at STA interface plane level with respect to reference frame (inertial or local orbital frame)

The satellite system provides this pointing performance at the STA interface plane with an accuracy of 0,05 deg (3σ) around each axis.

In order to achieve this performance, the payload shall fulfil the two requirements given in section 3.1.4.3.

The platform pointing stability in routine is mission dependent.

Without perturbation due to the payload, the typical values are indicated in Table 2.3-2.

Frequency band	pointing stability (3σ)
0.1 to 1 Hz	7.10^{-4} deg/s
1 to 5 Hz	3.10^{-4} deg
5 to 20 Hz	10^{-2} deg/s
20 to 80 Hz	2.10^{-4} deg
>80 Hz	3.10^{-2} deg/s

Table 2.3-2 : Typical satellite pointing stability

2.4 ORBIT DETERMINATION AND CONTROL

2.4.1 ORBIT DETERMINATION PERFORMANCES

The measurements to determine the orbit are performed with the on board GPS. There are three methods to determine the orbit:

real time on board measurements: the orbit restitution accuracy is estimated to 120 m (3σ),

post-processing of the telemetry data: in this case, the orbit restitution accuracy is better than 30 m (3σ)

(this value must be confirmed for the case of very low orbit with an altitude < 800 km and depends on solar activity)

the orbit prediction depending on the initial error issued from the orbit restitution and the error issued from the orbit perturbation. In the low orbit case, the atmospheric drag effect is estimated with difficulties, affecting the orbit prediction accuracy. The orbit prediction accuracy is along the satellite track and strongly depends on the orbit altitude and on the solar activity.

2.4.2 ORBIT CONTROL

The PROTEUS platform is equipped with an hydrazine propulsion system which allows

a complementary injection after the launch phase,

to acquire the orbit with accuracy and to correct for launch errors,

to maintain the orbit.

The orbital manoeuvres resulting from these three operation types must correspond to an overall velocity increment ΔV equal to :

$$\Delta V = 130 \text{ m/s (for the 450 kg satellite class).}$$

The detail for other satellites is given in chapter 2.5.7.

The possible minimal magnitude of a manoeuvre is estimated to 0,5 mm/s, and the maximal one is equal to 5 m/s. The manoeuvres resolution is better than 0,5 to 1 mm/s depending on the number of thrusters used (2 to 4 which corresponds to the OCM2 and OCM4 modes). The accuracy to perform the manoeuvres is better than 5% after the in flight calibration.

The delay between two orbital corrections is typically one orbit duration for inclination corrections and 0.5 orbit duration for semi major axis corrections. The delay between two manoeuvres depends on the visibility characteristics; during the first visibility, a telemetry allows to know with accuracy the orbit just after the first manoeuvre and during the second visibility, a telecommand is sent to perform the next manoeuvre.

The satellite slew rate depends on the inertia of the payload and of the wheels torque capacity. In the nominal case, the available torque is 0.1 Nm (worst case) on each axis for a maximum duration of nearly 1 minute.

For information, Platform inertias and Center of Gravity (CoG) position are given in satellite co_ordinate system with solar arrays folded and unfolded (cf. Table 2.4-1). These platform characteristics do not take into account STA and launch vehicle adapter characteristics.

	Folded configuration	Unfolded configuration
Platform Inertias		
Ix (m^2*kg)	55	425
Iy (m^2*kg)	45	45
Iz (m^2*kg)	45	415
Platform CoG position		
X (mm)	480	525
Y (mm)	0	0
Z (mm)	-10	-10

Table 2.4-1 : Platform Inertias in CoG Satellite Reference Frame and CoG position in Satellite Reference Frame

The mission interruption duration depends on the flight satellite configuration :

For a standard flight configuration with the +Xs satellite axis aligned with the velocity direction in the Nadir or Earth pointing case, orbital manoeuvres shall not imply any flight configuration change, so the payload should stay pointed to the same direction. In this case, orbital manoeuvres should not imply any mission interruption. But during these manoeuvres, the pointing performance could be damaged.

For a vertical flight with the Zs satellite axis aligned with the velocity direction, the satellite shall be rotated by 90° before performing orbital manoeuvres. It will imply mission interruption; the duration will depend on satellite inertia and the time to perform the manoeuvres.

All these main performances are summarised in Table 2.4-2.

	Characteristic values
overall velocity increment ΔV	130 m/s (for the 450 kg satellite class) (detail given in chapter 2.5.7)
manoeuvres magnitude : minimal maximal	 1 mms 2.5 m/s
Manoeuvres resolution	0.5 - 1 mm/s
Manoeuvres accuracy	5%
Delay between two orbital corrections : for inclination corrections for semi- major axis corrections	 1 orbit duration 0.5 orbit duration
Available torque on each satellite axis	0.1 Nm for about 1 minute

Table 2.4-2 : Orbit control performances

2.5 FUNDAMENTAL NOTIONS FOR MISSION ANALYSIS

This chapter is a general introduction to the spatial mechanics for low Earth orbits. These first rules allows the User to perform his mission analysis, they are not specific to PROTEUS based missions.

The user must choose the orbit kind fulfilling the technical and scientific needs of his mission. At first, the criteria necessary to select the orbit are listed; the aim is to decide on the main orbital parameters without impacting the platform performances, without restricting the specifications at the payload level and to achieve the mission objectives.

Some useful notions for the orbit choice are reminded :

the main orbit kinds are listed with an abacus which allows to determine the orbit plane position for every case; this may imply an orbit inclination depending on the altitude,

the orbit period and the eclipse duration depending on the altitude are given in the keplerian orbit case,

the main forces which can disturb the satellite motion and their impact are described, for instance the daily inclination drift and the altitude drift versus time.

the visibility duration depending on the altitude, the elevation, the different stations used.

These notions are applied for circular orbits at an altitude between 400 and 1000 km and with an inclination between 0 and 100 deg because these orbits are considered as the most usual ones for PROTEUS missions. Highly elliptical or other particular orbits are conceivable but need specific studies.

The second part of this chapter deals with the orbital manoeuvres. It allows to determine the cost in ΔV (and therefore in propellant mass) so that the satellite is able to reach and to maintain the chosen operational orbit even when launch errors and orbital perturbations are considered. Then, the User can estimate if PROTEUS is able to fulfil the mission according to the chosen orbit.

2.5.1 CRITERIA FOR ORBIT DESIGN

The orbit choice criteria are classified into two groups: the first one concerns items having a direct impact on the platform performances, the second list only concerns the payload. In the latter, the most usual items are reported assuming each payload has very specific needs.

Main criteria which have an impact on the platform performances:

the pointing type, the payload accommodation and the orbit are coupled (see paragraph 2.3),

over 1000 km in altitude, the radiation effects become important and affect the mission life duration,

under 600 km in altitude, the atmospheric drag implies a propellant consumption increase to maintain the orbit; mono-oxygen reacts with satellite external materials like Kapton or solar cell connections, meaning that mission duration is affected,

the ground visibility duration with one Earth terminal leads to select a high inclination orbit, or an equatorial orbit and a high altitude,

the launch vehicle cannot reach all inclinations because of the launch pad latitude and/or azimuth restrictions. As a satellite can not procure a high orbit inclination modification on its own, the orbit inclination choice does not only depend on the payload needs; the launch pad location and the launch vehicle performances also have to be considered. For instance, an orbit inclination lower than 28.5 deg cannot be reached with a small launch vehicle from its usual launch pad, without an important dogleg (change of orbital plane by the launch vehicle needs an important propellant consumption)

the link budget optimisation leads to minimise the altitude for radar or telecommunication missions.

Main criteria for orbit design not impacting the platform performances:

the resolution for an optical mission leads to choose a low altitude for the orbit,

the accessibility of the mission leads to increase the altitude and constrains some particular altitudes depending on the mission needs,

the altitude repetitivity from one orbit to another one leads to select a frozen eccentricity.

2.5.2 DIFFERENT ORBIT TYPES

Hereafter are described some particular orbits:

phased orbits for which the satellite flies over the same ground track with a periodical cycle,

sun synchronous orbits for which the orbital plane keeps a constant angle with the Earth-Sun direction,

frozen orbits for which the perigee argument and the eccentricity are constant.

2.5.2.1 Phased orbits

A phased orbit ensures the periodicity of the satellite ground track. That means the satellite performs a daily revolution number corresponding to a rational fraction p = n+m/q with n = number of full revolutions per day, m = sub cycle duration (in days), q = cycle duration (in days). In this case, the ground track will have a period of q days. Chart 2.5.1 gives the inclination depending on the circular phased orbit altitude. The orbital perturbation taken into account is the J_2 effect for Earth gravitational potential, due to the Earth oblateness.

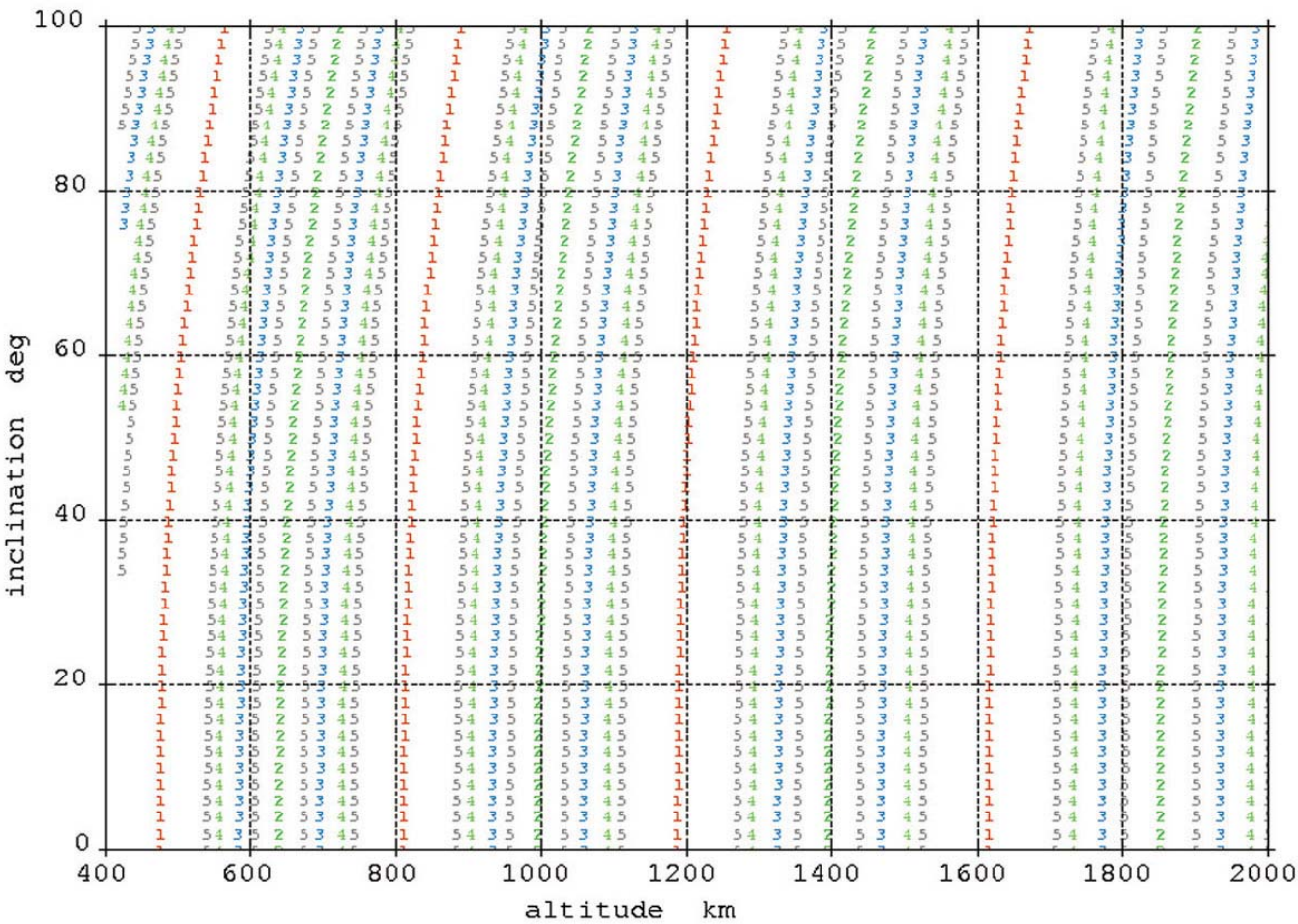

Figure 2.5-1: Phased circular orbits 1 to 5 days - the inclination depending on the altitude

2.5.2.2 Sun synchronous orbits

For Sun synchronous orbits, the orbital ascending node drift is equal to the Sun mean apparent rate

$\Omega_1 = n_S = 0.985626$ deg/day. Figure 2.5-2 shows the inclination for circular Sun synchronous orbits in the typical altitude range for PROTEUS based missions.

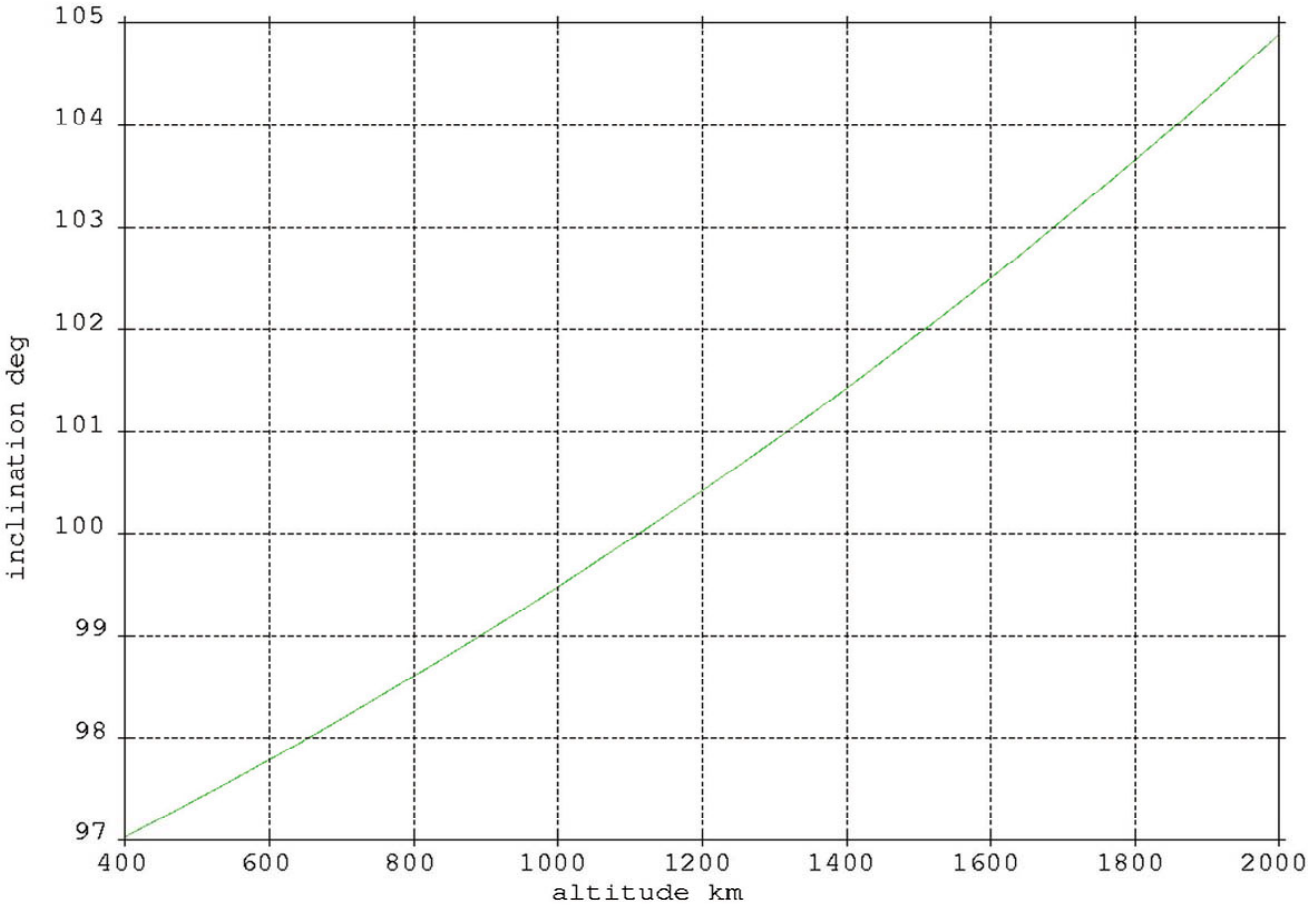

Figure 2.5-2: Sun synchronous circular orbit inclination versus altitude

2.5.2.3 Frozen orbits

Frozen orbits are chosen in certain missions to obtain a repetitive altitude at any given latitude. This is used for radar applications, where the range needs to be determined with accuracy. In order to maintain a constant altitude, the secular variations of average parameters such as the eccentricity e and the argument of perigee ω which are under Earth potential perturbation effects must be cancelled.

The following equation set is solved: $de/dt = f_1(e, \omega, u) = 0$ and $d\omega/dt = f_2(e, \omega, u) = 0$, where u corresponds to the orbital parameters such as the semi major axis *a* and the inclination *i*, f_1 and f_2 being temporal functions linked to the Earth potential.

Assumption: the expression used for the Earth potential to estimate the solutions includes the secular and long period terms (the zonal terms, up to degree 50), short period perturbations being considered to be negligible.

There are two optimum values of the frozen perigee $\omega_G = 90°$ and $\omega_G = 270°$. Figure 2.5-3 shows the frozen eccentricity depending on the couple of parameters (a, i).

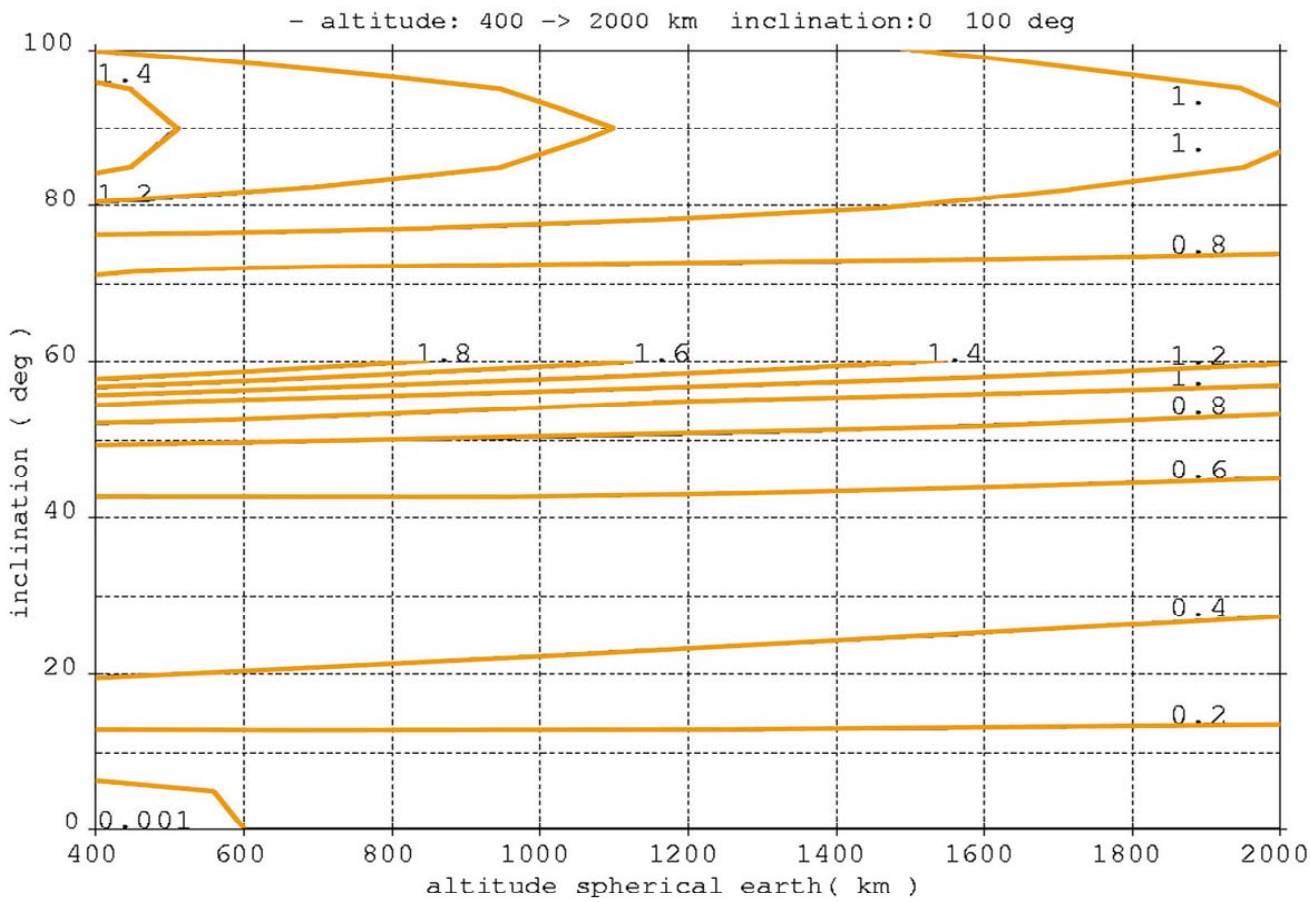

Figure 2.5-3 Frozen eccentricity for w = 90°

The eccentricities given on the Figure 2.5-3 are to be multiplied by 10^{-3}.

2.5.3 ORBIT PERIOD AND ECLIPSE DURATION

The objective is to give a rough idea of the orbit period and eclipse duration. These values are estimated under the keplerian orbit assumption. This model means that the only force applied to the satellite is the central force expressed in $1/r^2$ and caused by the Earth gravity. The keplerian period depends on the altitude for circular orbits as shown on Figure 2.5-4. The maximum eclipse duration depends on the altitude, and appears on Figure 2.5-5.

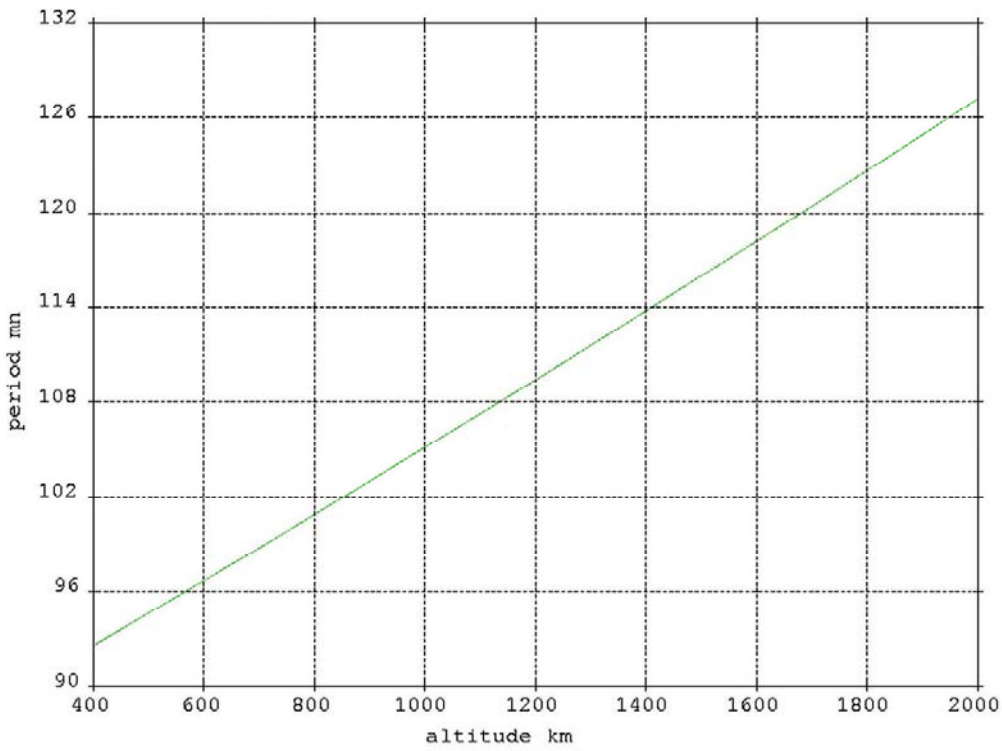

Figure 2.5-4: keplerian orbital period

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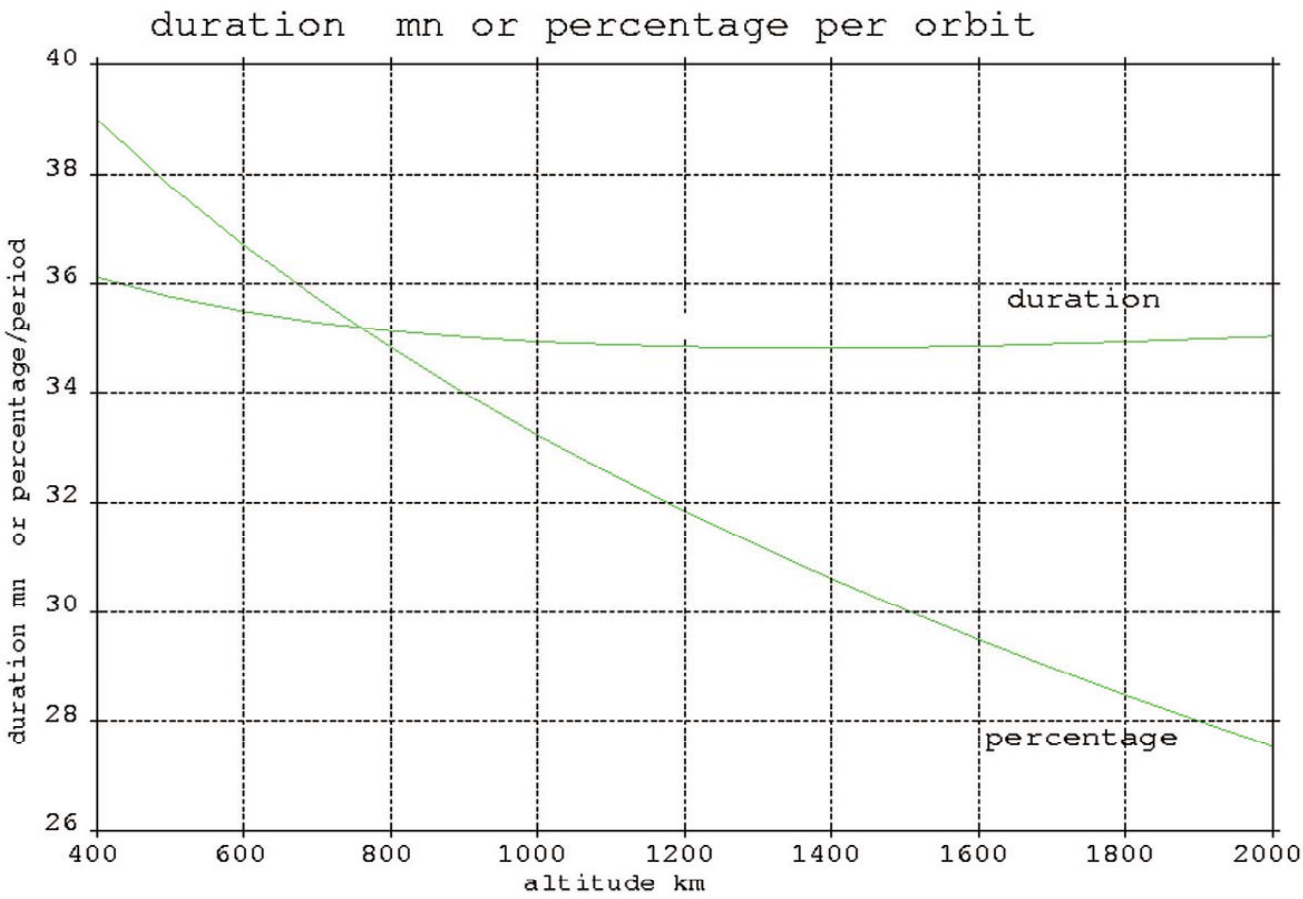

Figure 2.5-5: Eclipse duration and percentage of the orbital period depending on the altitude

2.5.4 ACCESSIBILITY

This property assesses the capacity to access the satellite from a given location on Earth. In the case of phased orbits for instance, it is very useful to evaluate the duration of the cycle for full access of the satellite to the equator over its orbital cycle. Figure 2.5-6 shows, as a function of altitude and for various orbital cycle durations, the half field of view of the instrument (or antenna) required for a full equatorial coverage. A calculation of this kind is necessary for each possible orbit, in order to estimate the performance of the mission.

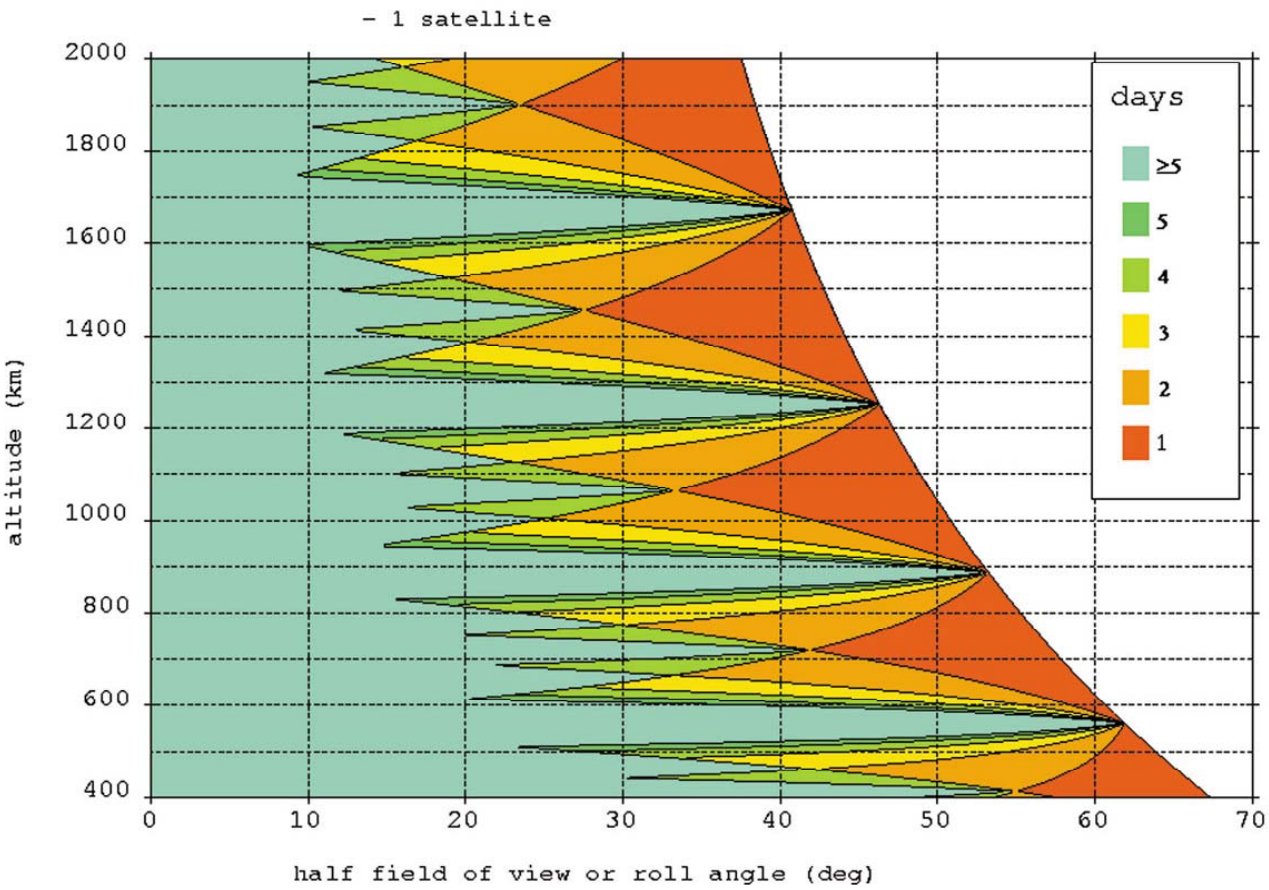

Figure 2.5-6: Equatorial accessibility of a phased satellite

2.5.5 VISIBILITY DURATION

The PROTEUS based satellite - ground link is ensured by a S-band TM/TC link which characteristics are detailed in chapter 9. Except for vertical flight (some restrictions to be analysed on a case by case basis), PROTEUS allows a TM/TC link budget with a margin greater than 3 dB for all altitudes from 400 to 1800 km and for all elevations over 5 deg (with no mask for ground antenna). For a circular orbit, the visibility duration of PROTEUS only depends on the orbit altitude, minimum elevation, and station localisation versus orbital track.

The control station visibility typical duration is estimated to 10 minutes in the case of low orbits characterised by a period of around 100 minutes.

The User can estimate the visibility duration for his mission with the following graphs. Depending on the ground station location, the visibility duration may be reduced due to some geometrical masks for elevation between 5 deg and 10 deg. Before the choice of the ground station location, the visibility duration budget shall be done with an assumption of a minimum elevation of 10°.

Figure 2.5-7 gives the station visibility duration depending on the maximum elevation when the satellite enters the visibility area (defined by a minimum 5 elevation, here), for low orbits (altitudes between 400 km and 1800 km). Figure 2.5-8 gives the same think for a minimum 10° elevation.

A computation of the duration of the accesses of the satellite to the associated ground station is necessary for each specific mission, in order to make sure that the link budget is adapted to the downloading needs of the mission, given the capabilities of the TM/TC function.

For information, Figure 2.5-9, 2.5-11, 2.5-13 and Figure 2.5-15 give for Kiruna (Sweden), Aussaguel (France), Kourou (French Guiana) and Hartbeesthock (HBK South Africa) stations the mean daily visibility duration function of altitude and inclination with a minimal elevation equal to 5°. Figure 2.5-10, 2.5-12, 2.5-14 and 2.5-16 give the same think for a minimal elevation equal to 10°.

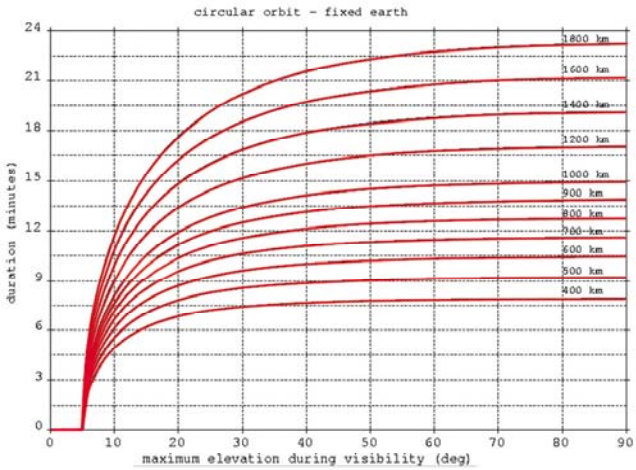

Figure 2.5-7: Station visibility duration (minimum elevation 5°)

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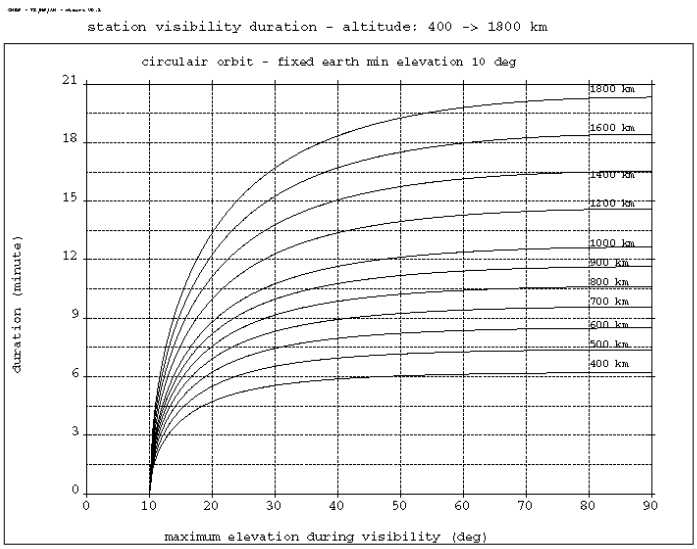

Figure 2.5-8: Station vibility duration (minimum elevation 10°)

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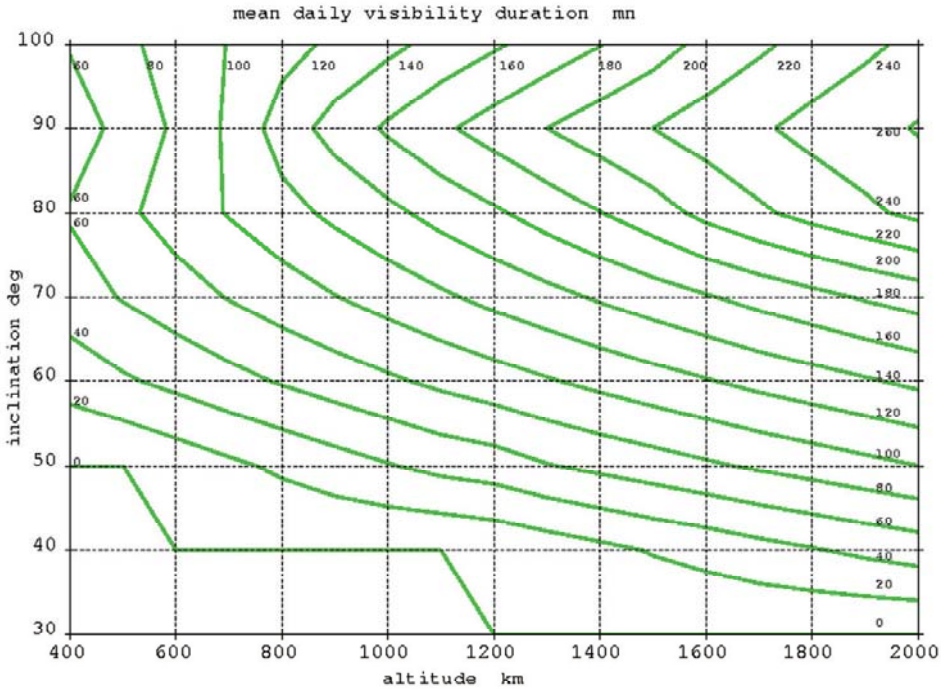

Figure 2.5-9 : Kiruna (21.1 E, 67.9 N) (minimum elevation 5°)

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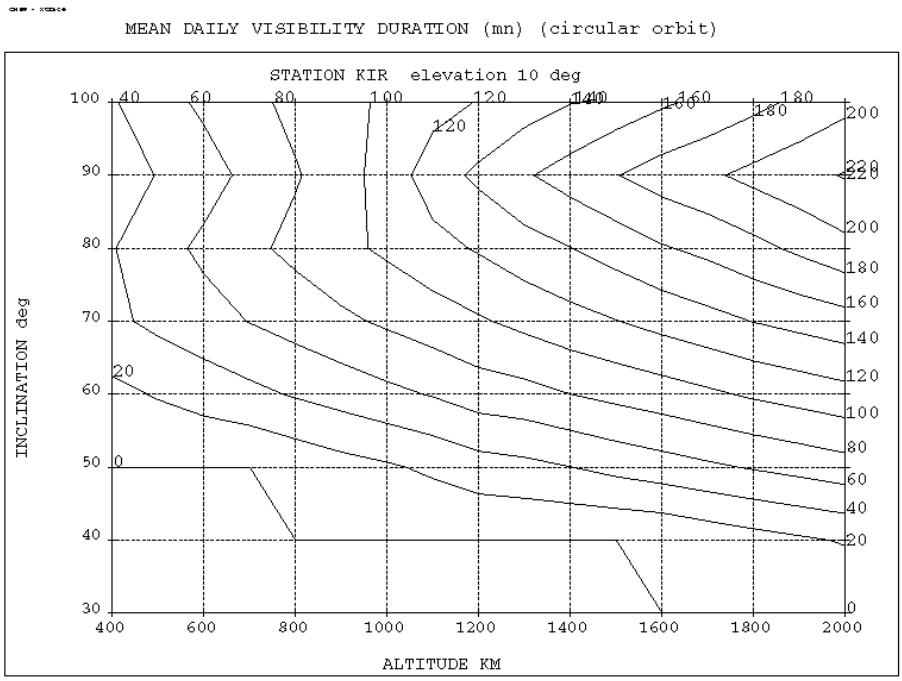

Figure 2.5-10: Kiruna (21.1 E, 67.9 N) (minimum elevation 10°)

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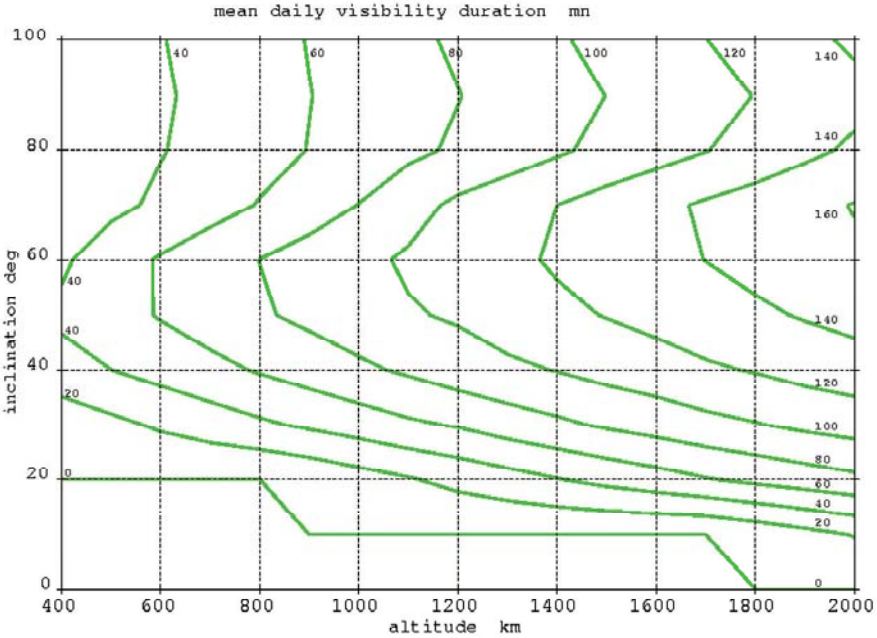

Figure 2.5-11 : Aussaguel (1.5 E, 43.4 N) (minimum elevation 5°)

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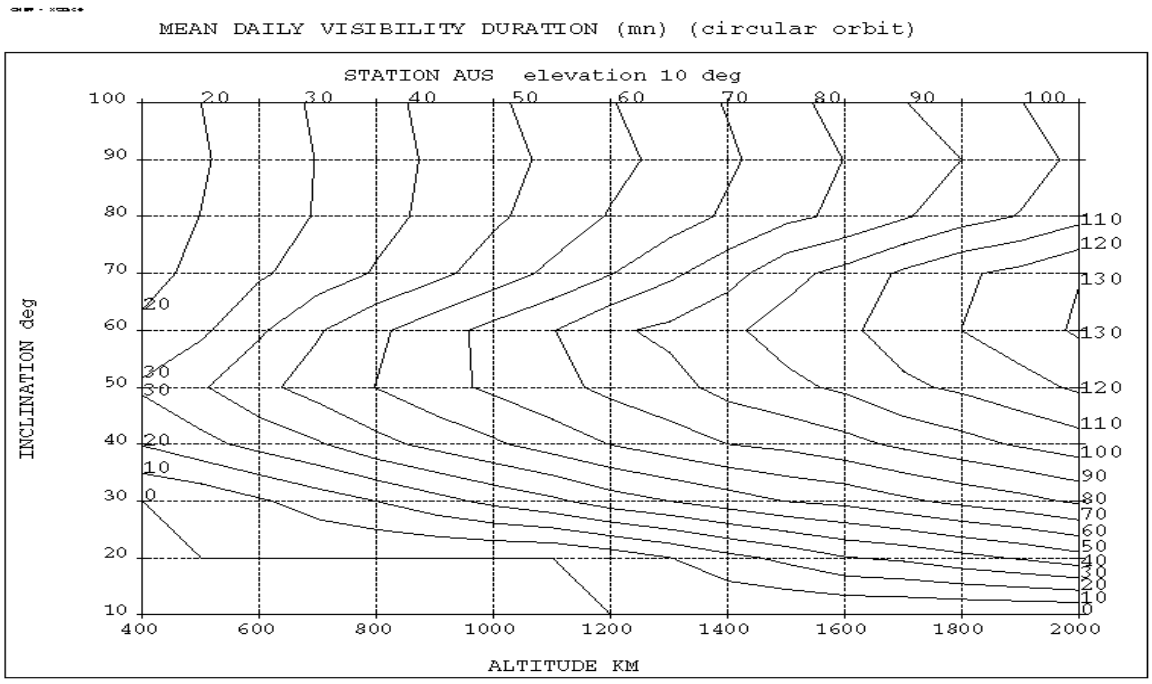

Figure 2.5-12 : Aussaguel (1.5 E, 43.4 N) (minimum elevation 10°)

..

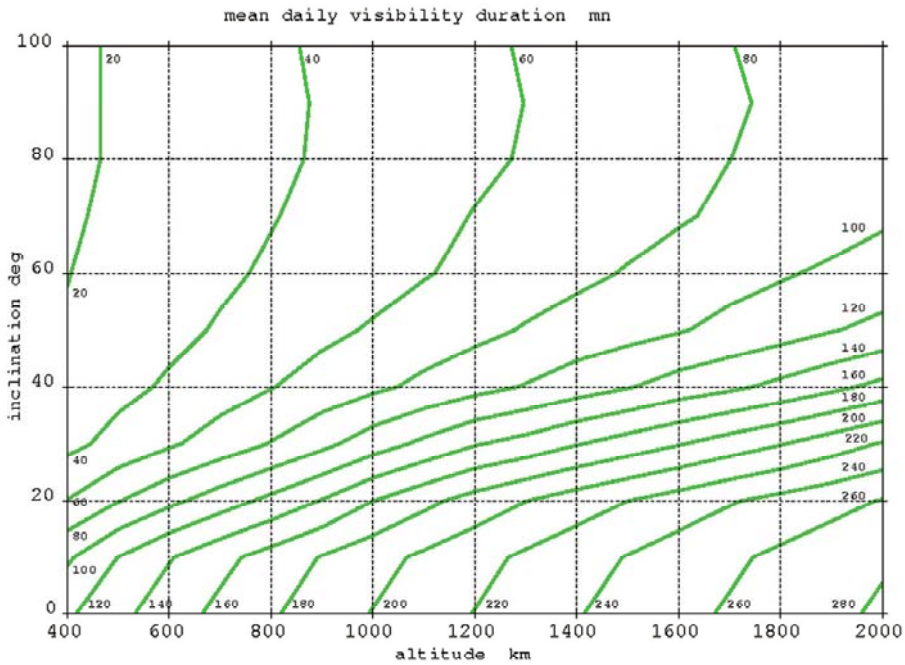

Figure 2.5-13 : Kourou (52.6 W, 5.1 N) (minimum elevation 5°)

..

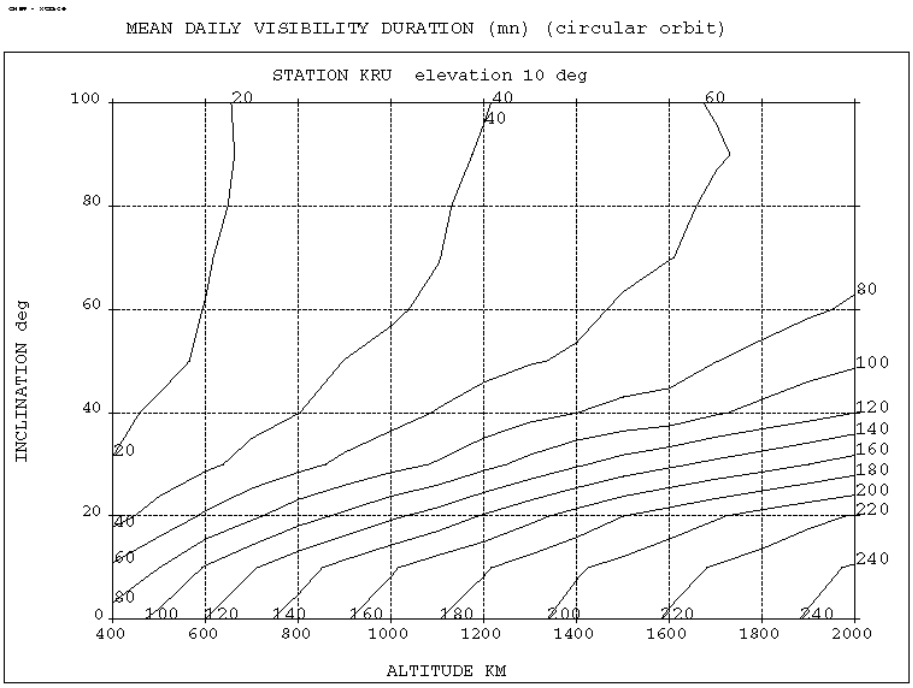

Figure 2.5-14 : Kourou (52.6 W, 5.1 N) (minimum elevation 10°)

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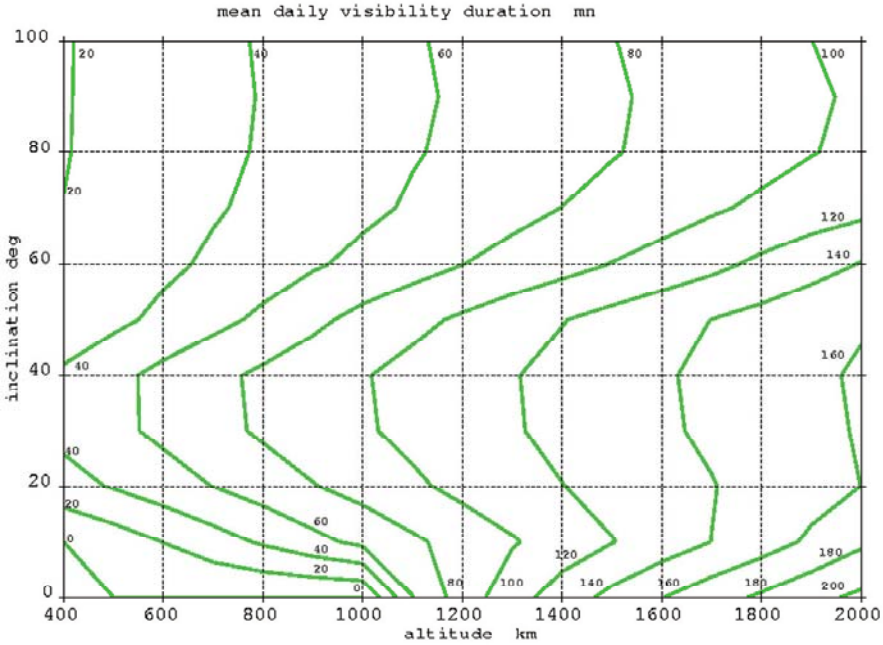

Figure 2.5-15 : Hartbeesthoek (27.7 E, 25.9 S) (minimum elevation 5°)

..

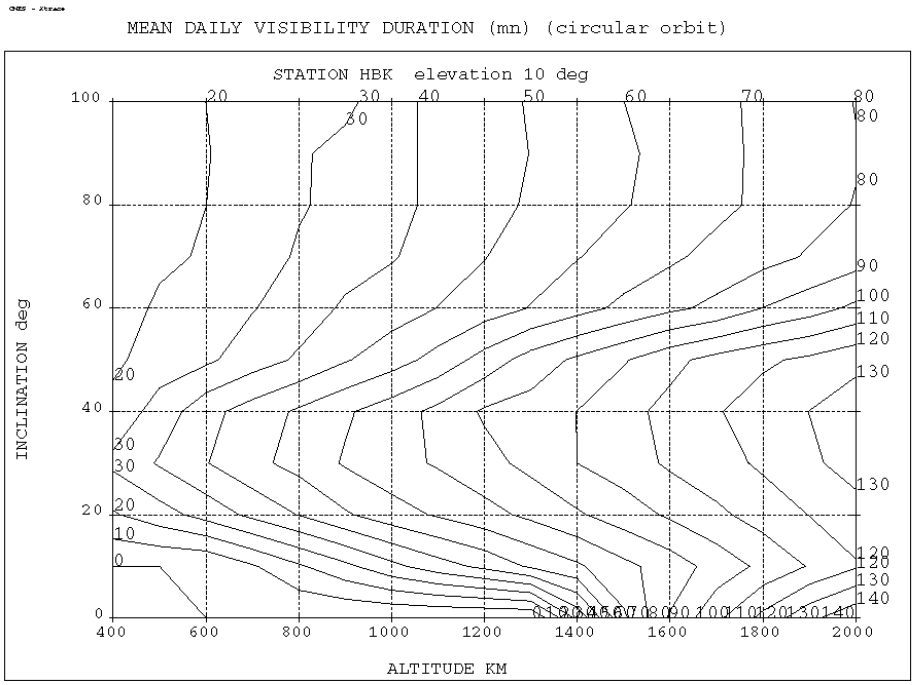

Figure 2.5-16 : Hartbeesthoek (27.7 E, 25.9 S) (minimum elevation 10°)

2.5.6 ORBITAL PERTURBATIONS

In the keplerian orbit case, the satellite only experiences a central force behaving in $1/r^2$. In reality, the satellite movement is perturbed by other forces:

gravitational disturbances: oblateness of the Earth, Sun and Moon gravitational attraction,

non gravitational disturbances: solar radiation pressure, atmospheric drag.

2.5.6.1 Earth potential

Earth potential is neither spherical nor homogeneous. It can be described as the sum of spherical harmonics. $C_{l,m}$ and $S_{l,m}$ are the harmonic coefficients with the degree l and the order m. The spherical harmonics can be classified into two groups:

the zonal harmonics ($m = 0$, $J_n = -C_{no}$) correspond to the irregularities in latitude

the tesseral harmonics ($m \neq 0$, $m \neq l$) corresponds to the irregularities in longitude

The first zonal harmonic (J_2) is about 10^{-3} in comparison with the main term in μ/r whereas higher terms have a magnitude lower or equal to 10^{-6}. In order to have better results, the model for Earth potential often takes into account the J_2 harmonic.

According to this assumption, the secular variations of orbital elements of the orbital node Ω and the perigee argument ω spell in the following way:

the orbital node drift in deg/day:

$$\Omega_1 = -\frac{3nJ_2R_t^2\cos i}{2(1-e^2)^2a^2}$$

$$\Omega_1 = \frac{-2.064 \text{ x } 10^{14}\cos \text{i}}{\left(1-\text{e}^2\right)^2 \ \text{a}^{7/2}}$$

the perigee argument drift in deg/day:

$$\omega_1 = -\frac{3nJ_2R_t^2(1-5\cos^2 i)}{4(1-e^2)^2a^2}$$

$$\omega_1 = -\frac{1.032 \text{ x } 10^{14}\left(1-5\cos^2 i\right)}{\left(1-e^2\right)^2 \ \text{a}^{7/2}}$$

with n = Keplerian average angular velocity, a= semi major axis in km, i = inclination in deg.

Figure 2.5-17 shows the orbital node drift depending on the (inclination, altitude) couple for a circular orbit.

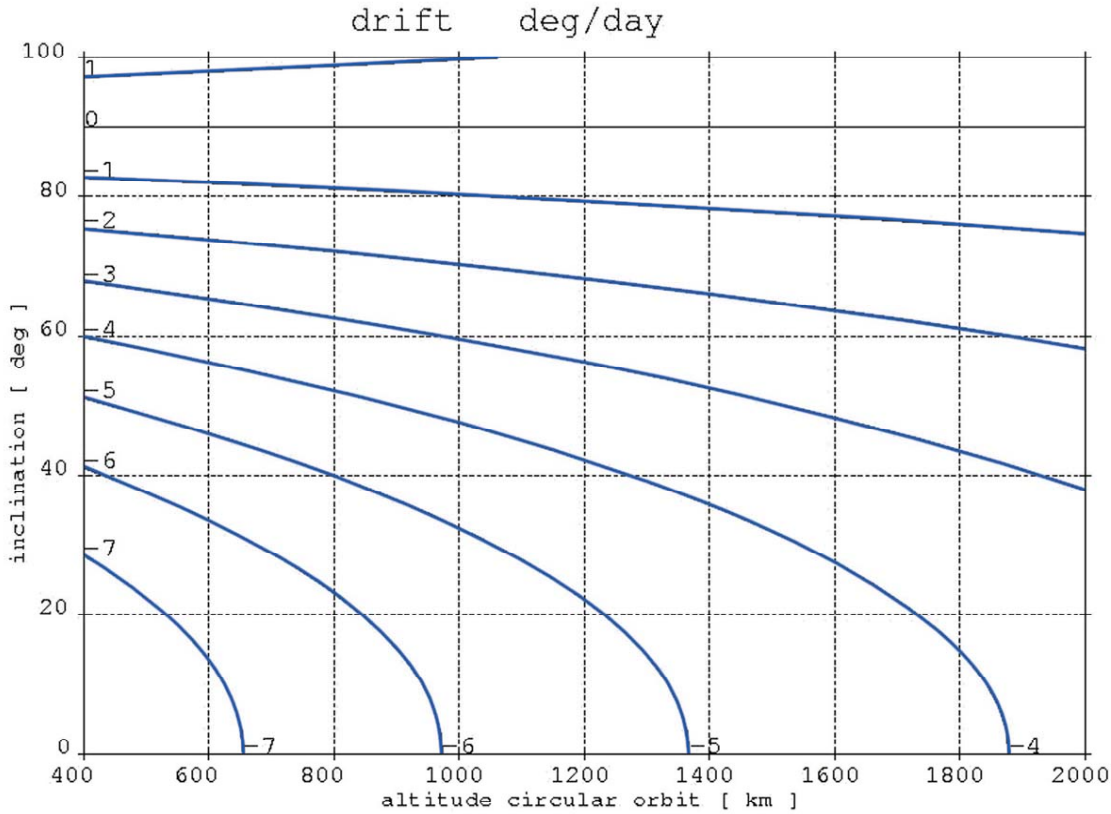

Figure 2.5-17: Orbital node secular drift, for a circular orbit

2.5.6.2 Moon and Sun gravity potential influence

As the Moon is attracted by the Earth and the Earth by the Sun, a satellite in space is attracted by more than one celestial body. There are usually long period effects (1 year) on the inclination i, the orbital node Ω and the position on orbit α and on the secular terms on Ω. In the Sun synchronous orbit case, there is a secular drift on i and Ω.

The secular drift due to the effects of the Sun and Moon on i and Ω strongly depends on the local hour of the ascending node: this property can be used to reduce this drift value.

2.5.6.3 Atmospheric drag

Whereas the previous perturbations have a gravitational origin, the atmospheric drag creates areal forces.

The aerodynamic force F, caused by the atmospheric drag up to a 1000 km altitude, gives the corresponding acceleration:

$$\gamma = -\frac{1}{2}\rho V^2 C_X \left[\frac{S}{m}\right]\vec{u}$$

where ρ is the atmospheric density, V the spacecraft velocity versus the atmosphere, S the drag effective surface, C_X the drag coefficient, m the mass (C_x depends on the altitude; C_x (400 km) = 2.2, C_x (600 km) = 2.5, C_x (800 km) = 2.7, C_x (1000 km) = 2.75).

The drag effect on the orbital elements is mainly a secular decrease on the semi-major axis:

$$\frac{da}{dt}\ (\text{m/day}) = -1.725.10^{12}\ \sqrt{a}\rho\ \ C_X\ \frac{S}{m}$$

with a (m), ρ (kg/m^3), C_X (m^2/kg).

Figure 2.5-18 gives the the solar activity curve for the [01/2003-01/2015] period, taken into account in the da/dt and ΔV calculation (table 2.5-1)

SOLAR ACTIVITY 01/2003-01/2015

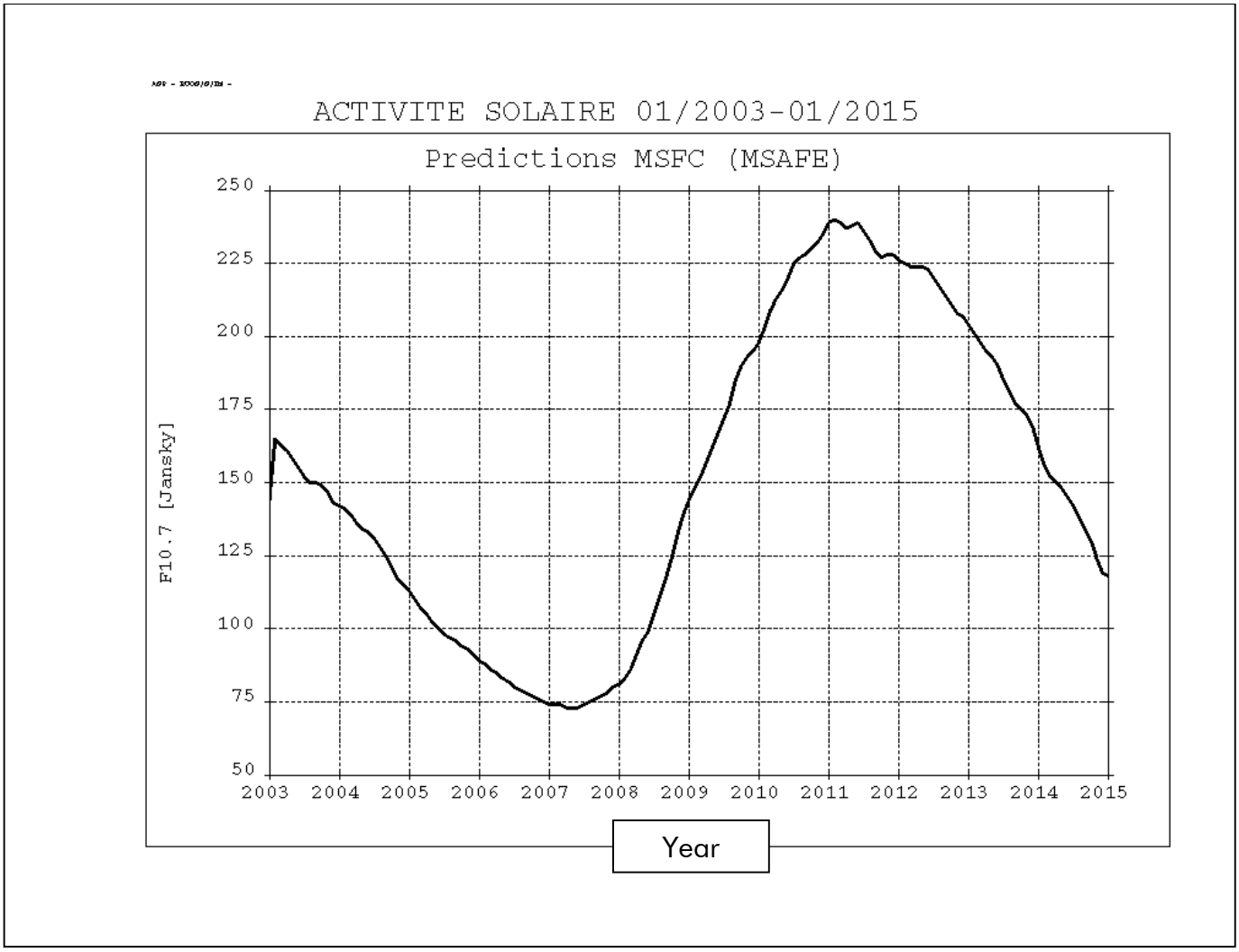

Figure 2.5-18: Solar activity for 01/2003-01/2015 period

Table 2.5-1 gives the annual budget of the estimated altitude drift da/dt in m/year and the corresponding velocity increment ΔV (m/s) needed to compensate for this drift in order to maintain the nominal orbit for a 400-2000 km altitude range and for the years 2003 to 2014. The S/m coefficient depends on the size and mass of the payload, the local time of the ascending node, the orbit inclination, and the season. For the da/dt and DV calculation, the main hypothesis are given here after :The S/m ratio is equal to 10^{-2} m^2/kg

the satellite mass is of 500 kg,

the Cx coefficient is between 2.2 and 2.7, depending on the altitude,

the orbit is Sun Synchronous with an ascending node at 12h00.

the orbit maintenance strategy is based on semi-major axis correction (section 2.5.7.4).

This table is given for information only; the calculation specific to the mission (taking into account the drag effective surface, the satellite mass, the chosen orbit, the orbit maintenance strategy) will be performed case by case.

	2003	2004	2005	2006	2007	2008	2009	2010	2011	2012	2013	2014	
400 km	198963 112,5	135517 76,7	77340 43,8	47854 27,1	44583 25,2	93321 52,8	240417 136	449580 254,3	525718 297,4	438443 248	157,3	278053 82,3	
450 km	89583 50,1	57982 32,4	31422 17,6	17892 10	16658 9,3	37731 21,1	110102 61,6	221187 123,8	263154 147,2	216272 121	72,8	130138 35,5	
500 km	42718 23,6	26292 14,6	13228 7,3	7138 4	6632 3,7	16552 9,2	53326 29,5	114078 63,1	139178 77	111964 62	35,5	64222 15,9	
550 km	21355 11,7	12386 6,8	6028 3,3	3142 1,7	2868 1,6	7618 4,2	27000 14,8	61562 33,7	76158 41,7	60010 32,9	18,2	33320 7,7	1
600 km	12224 6,6	7239 3,9	3139 1,7	1551 0,8	1385 0,8	3878 2,1	15936 8,6	39222 21,2	48602 26,3	38114 20,6	10,8	19851 4,3	8
650 km	6365 3,4	3602 1,9	1568 0,8	784 0,4	747 0,4	2053 1,1	8399 4,5	21950 11,8	27773 14,9	21819 11,7	5,8	10919 2,1	39
700 km	3641 1,9	1836 1	830 0,4	509 0,3	490 0,3	1170 0,6	4565 2,4	12639 6,7	16619 8,8	12733 6,8	3,3	6169 1,1	21
750 km	1945 1	1049 0,6	553 0,3	343 0,2	324 0,2	591 0,3	2631 1,4	7550 4	9952 5,2	7492 3,9	1,8	3451 0,6	118
800 km	1272 0,7	751 0,4	385 0,2	231 0,1	250 0,1	385 0,2	1638 0,8	4970 2,6	6704 3,5	4855 2,5	1,1	2177 0,4	848
900 km	531 0,3	374 0,2	216 0,1	138 0,1	138 0,1	236 0,1	747 0,4	2105 1,1	2616 1,3	1908 1	0,5	885 0,2	393
1000 km	261 0,1	221 0,1	141 0,1	100 0,1	100 0,1	120 0,1	341 0,2	944 0,5	1285 0,6	964 0,5	0,2	442 0,1	221
1100 km	164 0,1	143 0,1	61 < 0,1	61 < 0,1	61 < 0,1	102 0,1	225 0,1	451 0,2	615 0,3	512 0,2	0,1	266 0,1	143 0,1
1200 km	125 0,1	63 <0,1	63 <0,1	42 <0,1	42 <0,1	63 <0,1	125 0,1	272 0,1	397 0,2	272 0,1	0,1	167 0,1	63 <0,1
1500 km	44 <0,1	22 <0,1	22 <0,1	22 <0,1	22 <0,1	22 <0,1	66 <0,1	111 0,1	133 0,1	133 0,1	0,1	66 <0,1	44 <0,1
2000 km	24 < 0,1	24 < 0,1	< 1 < 0,1	< 1 <0,1	< 1 < 0,1	< 1 < 0,1	24 < 0,1	24 < 0,1	49 < 0,1	24 < 0,1	24 < 0,1	24 < 0,1	

Table 2.5-1: Annual da/dt (m/year) and ΔV (m/s) budgets for a normalized $\frac{S}{m}$ = 10^{-2} m²/kg

2.5.6.4 Solar radiation pressure

The solar wind implies an alteration of the satellite momentum.

The acceleration corresponding to the direct pressure is:

$$M = -K.\left(\frac{S}{m}\right).i_p.u_{sun}$$

where m is the spacecraft mass, S the equivalent surface (spacecraft + solar arrays) perpendicular to the solar flux, K the radiation coefficient and equal to 4.56 10^{-6}.N/m^2, i_p the illumination parameter (1 if the spacecraft is illuminated, 0 if not), u_{sun} the unit vector spacecraft-Sun.

The main parameter is the angle β between the orbital plane and the Earth-Sun direction; for a given altitude, it determines the illuminated part x of the orbit (associated period T_β).

The usual effects on the orbital elements are long term perturbations on the eccentricity vector coordinates e_x and e_y, on the inclination i, on the position on orbit α.

Typical values of this perturbation for the main orbital elements are:

$$\frac{di}{dt} = -5.2\ \ 10^{-4} \text{ degrees/year} \qquad ; \qquad \frac{d\,pso}{dt} = 3.10^{-3} \text{ degrees/year}$$

$$\frac{de_y}{dt} = 10^{-4}\text{/year}$$

For Sun synchronous orbits, these terms become secular terms depending on the local hour of the ascending node and on x.

2.5.6.5 Synthesis

In comparison with the central term of Earth potential, the Earth oblateness term (J_2) is the most important orbit perturbation. The main effects are secular drifts of the orbital parameters ω, Ω (typically a few degrees/day) and α.

For long range studies, the non periodic variations of elements (secular terms) are predominant.

The atmospheric drag causes a secular drift on the semi major axis which becomes very important under 600 km.

For most missions, other effects can be considered as negligible in a first approach.

2.5.7 ORBITAL MANOEUVRES

The satellite is able to modify the orbit characteristics with its propulsion system. It can perform orbital manoeuvres for several reasons:

in order to correct certain orbital parameters after injection by the launch vehicle (possible errors at injection or sometimes because the launch vehicle cannot deliver directly the satellite on its operational orbit),

because of perturbations which are the result of a non ideal keplerian movement (see paragraph 2.5.4),

for orbit transfer or *rendez-vous* manoeuvres.

2.5.7.1 PROTEUS capabilities

The PROTEUS Satellite manoeuvres use a propelling system with hydrazine (maximum capacity of the tank: 28 kg).

In order to estimate the mass of propellant used for an instantaneous thrust necessary for orbital manoeuvres, the following formula is used:

$$\Delta m = m_0 \left(1 - e^{\frac{-\Delta V}{g\,Isp}} \right) (1)$$

with m_o is the initial mass of the satellite, I_{sp} is the specific impulse of the engines (in seconds).

Figure 2.5-19 shows the satellite capability over its life time in term of total velocity increment that allows a hydrazine mass of 28 kg. The curve gives the evolution of the ΔV versus payload mass. The assumptions for this chart are the following:

a specific impulse I_{sp}=220 s,

a satellite initial mass m_0 = 330 + m_{ePL}, assuming that m_{ePL} is the equipped payload mass and that 330 includes 28 kg of Hydrazine (see Table 3.1-1).

Figure 2.5-19: Maximum DV as a function of the equipped payload mass

A specific and detailed fuel budget will be calculated for each mission taking into account launch vehicle dispersions, the manoeuvres needs (established by mission analysis) and the performance criteria of the propulsion and its components (from Design and Acceptance testing).

2.5.7.2 Cost in ΔV to change orbital parameters

Orbital manoeuvres mainly consist in correcting the orbital parameters like the inclination and the semi major axis. This allows for control of the orbital plane drift, satellite position and phasing parameters.

The cost in ΔV corresponding to an inclination or semi major axis correction in the case of low circular orbits can be evaluated using the charts provided hereafter.

The orbital parameters variations (Δa, Δe_x, Δe_y, Δi, $\Delta\Omega$, $\Delta\alpha$) can be expressed as functions of the tangential, radial and normal coordinates of the velocity increment ΔV:

$$\Delta a = 2a\frac{\Delta V_T}{V} \quad (2)$$

$$\Delta e_x = 2\cos\alpha\frac{\Delta V_T}{V} + \sin\alpha\frac{\Delta V_R}{V}$$

$$\Delta e_y = 2\sin\alpha\frac{\Delta V_T}{V} - \cos\alpha\frac{\Delta V_R}{V}$$

$$\Delta i = \cos\alpha\frac{\Delta V_N}{V}$$

$$\Delta\Omega = \frac{\sin\alpha}{\sin i}\frac{\Delta V_N}{V}$$

$$\Delta\alpha = -2\frac{\Delta V_R}{V} - \frac{\sin\alpha}{\tan i}\frac{\Delta V_N}{V}$$

Figure 2.5-20 gives the velocity increment for altitudes between 400 and 1600 km needed for semi major axis corrections (the formula (2) is applied for the calculation). For instance, if the need is to correct the semi major axis of 40 km at an altitude equal to 700 km, this manoeuvre corresponds to a change in velocity ΔV = 21 m/s. It can be deduced from Figure 2.5-20 that the PROTEUS satellite can perform this manoeuvre with a payload mass up to 400 kg. The associated consumed propellant mass can be estimated with formula (1).

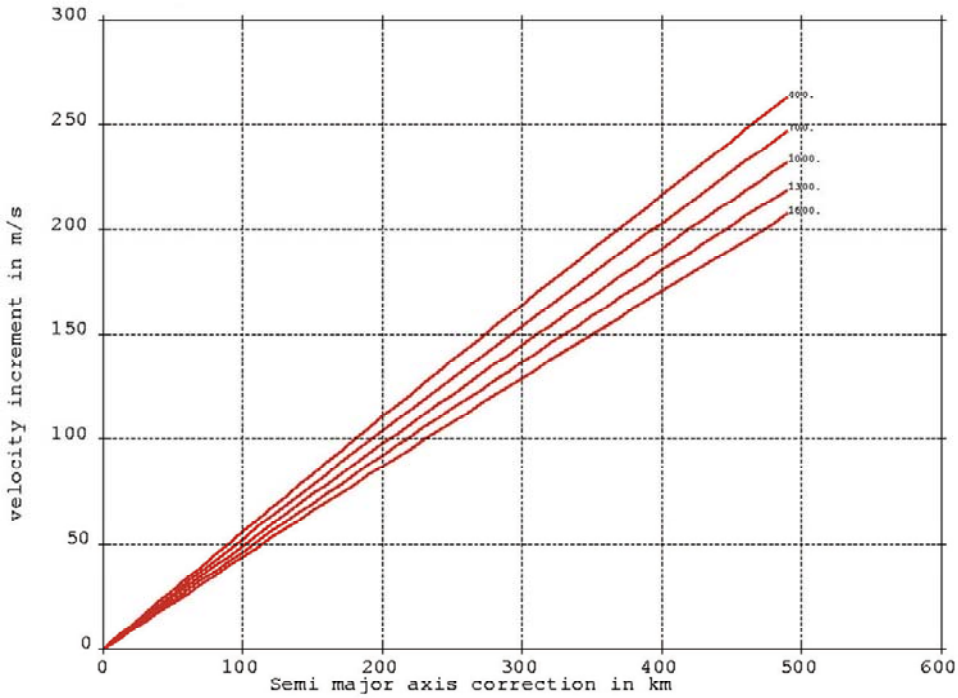

Figure 2.5-20: Semi major axis correction cost

Figure 2.5-21 gives the velocity increment for altitudes between 400 and 1600 km, for an inclination correction from 0 to 0.9 deg. Using the same process as above, one can estimate PROTEUS capacity in terms of inclination correction.

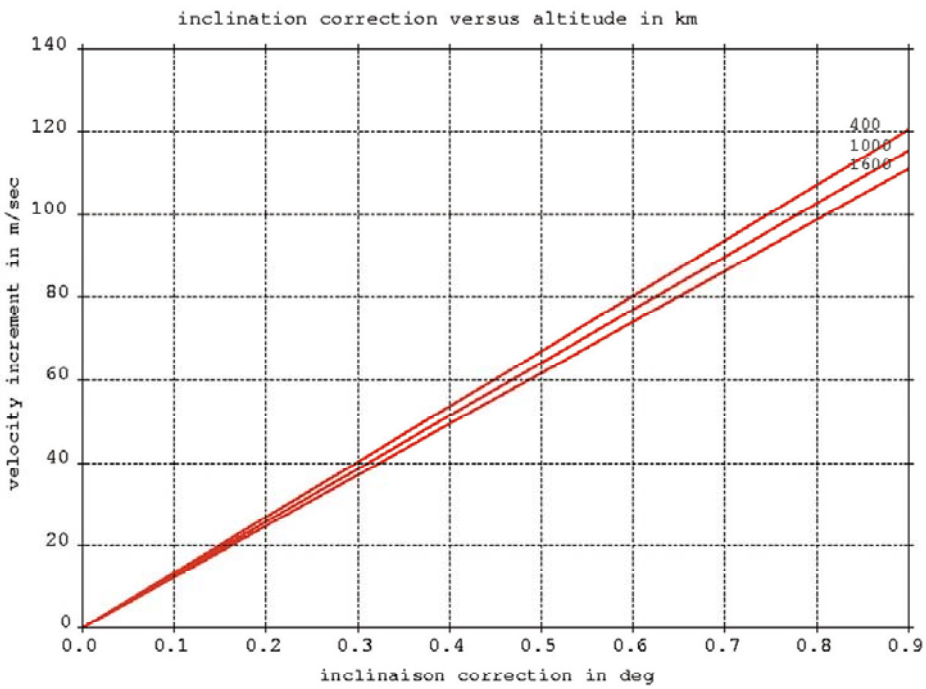

Figure 2.5-21:Inclination correction cost

2.5.7.3 Orbit positioning

In most of the cases, after launch, a certain period is dedicated to the satellite in order for it to reach the target orbit necessary for mission completion. The initial orbit, depending on the capacity and on the accuracy of the launch vehicle, can be close to the operational orbit, or the satellite can be put on a parking orbit which can be quite different from the one of the mission itself. In both cases, the satellite must perform manoeuvres to be positioned on its nominal orbit. These positioning manoeuvres can modify every orbital parameter. In order to limit the propellant consumption of the satellite, the ΔV manoeuvres are performed tangential to the velocity (semi major axis corrections).

Launch errors

The satellite can not reach the target orbit with the sole launch vehicle; errors occur during flight and at the satellite/launch vehicle separation. These errors are usually estimated by a covariance matrix at injection. In order to correct the orbit parameters, two approaches can be applied:

- the satellite is close to the target orbit and the significant launch vehicle errors are corrected,
- the satellite is on a drift or parking orbit and the strategy consists in correcting the injection errors during the optimisation of the global orbit positioning process to decrease the manoeuvres number and the cost in ΔV.

Strategy

The strategy for orbit positioning often foresees a *rendez-vous* to achieve the target orbit with accuracy. In a first step, the strategy consists in introducing a sequence of impulse manoeuvres and Hohmann transfers between intermediate orbits in order that the satellite reaches the target orbit.

As soon as the User chooses the sequence, the kind and the number of orbital manoeuvres, he can estimate the cost in ΔV, and so the propellant consumption of the satellite and the mission feasibility with a PROTEUS system can then be deduced (see paragraphs 2.5.6.1 and 2.5.6.2).

2.5.7.4 Orbit maintenance

For most of the missions a station keeping strategy is necessary to take into account every constraint. For instance, the phased orbits need to have a well defined semi major axis. In this case, because of orbital perturbation effects, the semi major axis value must be regularly corrected to maintain the satellite in a given altitude range. The allowed variations of orbital parameters depend on the mission and their limits can be deduced from a specific analysis.

The usual strategy consists in correcting the semi major axis. The inclination parameter is less often altered. The number and the amplitude of manoeuvres depend on the strategy applied to position the satellite in a given window. The station keeping manoeuvres can affect every orbital parameters (a, e_x, e_y, i, Ω, α). The main perturbation concerning orbit maintenance is due to the atmospheric drag. As soon as the altitude and the duration of the mission are known, one can estimate the cost in ΔV needed to compensate for atmospheric drag (see Table 2.5-1) and the corresponding inclination correction can be evaluated with Figure 2.5-1.

2.5.7.5 Synthesis

The main orbital manoeuvres during orbit positioning and maintenance are used to compensate for launch errors dispersion, phasing if necessary, and atmospheric drag. The most usual manoeuvres consist in correcting the semi major axis and the inclination. Each mission needs a different strategy depending on the mission constraints, the satellite AOCS properties, and the station visibility. For instance, a mission can require to perform every orbital manoeuvre in eclipse or over the sea while the payload is turned off. These constraints, once included into the optimisation process, can modify the manoeuvre schedule. The manoeuvre frequency mainly depends on the satellite altitude and on the window allowed for the orbital parameters.

2.5.8 ORBIT DEBRIS GENERATION ANALYSIS

Analysis shall be led and documented for each Proteus mission to assess orbital debris generation potential and debris mitigation options.

This analysis is required in particular to demonstrate compliance with the requirements of NASA Directive NPD 8710.3 and [RD12]

The analysis shall include the following:

- potential for orbital debris generation in both nominal operation and malfunction conditions, including malfunctions during launch phase.

- potential for orbital debris generation due to on-orbit impact with existing space debris (natural or human generated) or other orbiting space systems.

Such orbital debris generation analysis was performed for the JASON 1 mission [RD13], and can be used as reference for subsequent analysis reports.

The Payload Supplier shall provide input for the corresponding Satellite System Analysis.

END OF CHAPTER