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Venus Express: Consolidated Report On Mission Analysis

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by

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05 April	3		Chapter 13	Now presents the delta-V budget for every launch day.	
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1. PURPOSE

The Venus Express Consolidated Report on Mission Analysis, CRMA, summarizes the current status of the Venus Express mission analysis tasks performed by ESOC/OPS-GA up to now, and presents relevant mission data for the design of Venus Express.

Data is presented for:

- Parametric analysis
- Mission baseline description
- Launch with Soyuz+Fregat and Flight Programs
- Launch Window
- Interplanetary trajectory and associated data
- Navigation
- Venus Orbit Insertion
- Venus operational orbit and pericenter control
- Delta-V budget

All trajectories are propagated using high precision numerical integration, including all relevant perturbing forces, and using the JPL planetary ephemerides.

The analysis of the Venus Orbit Insertion and of the Operational Orbit Pericentre Control is presented for a strategy that maximizes the pericentre controlled lifetime. The proposed operational strategy is based on a maximization of the propellant mass at the end of the required minimum 1000 days pericentre controlled lifetime (R 11). The characteristics of the Venus injection phase (eclipses, Sun and Earth aspect angles, transmission black-out…) are very similar for both cases.

2. REFERENCES

2.1 Applicable Documents

A 1. ESA, Venus Express Mission Requirements Document, VEX-EST-RS-0022, Issue 2, Rev. 0, 31 July 2003

2.2 Reference Documents

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R 10. EADS Astrium, Venus Express Mass Property Budget Report, Issue 4, Rev. 0, 24.02.2004.

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3. MISSION REQUIREMENTS AND CONSTRAINTS

3.1 General Mission Description

The Venus Express mission was proposed in response to a "Call for Ideas for the Re-Use of the Mars Express Platform" issued by ESA in March 2001. After a Mission Assessment Study, R 3, the mission was selected for further development. Work at Pre-phase B level extended up to mid 2002. The final selection as an ESA mission was made in November 2002. The mission is performed as an ESA flexible mission with Astrium leading the industrial team.

The scientific objectives of Venus Express include sub-surface and surface studies, atmospheric physics up to the plasma environment, and interaction with the solar wind. These objectives will need a multipurpose remote sensing and *in situ* payload. The payload is based on the instruments available from the Mars Express and from the Rosetta missions. The scientific payload considered as a baseline for the current industrial study is composed of the instruments described in Table 1:.

Instrument Name	Description
ASPERA	Analyser of Space Plasmas and Energetic Neutral Atoms
MAG	Magnetometer
PFS	High-resolution Atmospheric IR Fourier Spectrometer
SPICAV	UV and IR spectrometer for solar/stellar occultation and nadir observation
VeRa + USO	Venus radio science experiment + Ultra Stable Oscillator
VIRTIS	UV-Visible-near IR imaging spectrometer
VMC	Venus monitoring camera

Table 1: Venus Express Model Scientific Payload

The mission is based on a spacecraft derived from the Mars Express platform. The spacecraft will carry instruments adapted from instruments developed for Rosetta or for Mars Express. The propulsion system is based on a Main Engine of 400N, with attitude and orbit control thrusters of 10N.

Using a Soyuz+Fregat launch vehicle, Venus Express will be launched from Baikonur in the period October-November 2005. After an interplanetary trajectory of about 5 months, it will reach Venus and, after a sequence of manoeuvres, it will be injected into its final operational orbit.

The orbit around Venus has been selected as a polar orbit with an initial pericenter height of 250 km, and an orbital period near to 24 hours (apocenter height of about 66600 km). The high inclination has been selected to allow a complete coverage in latitude, and is a good compromise between the high resolution observations of the northern hemisphere, in particular the region Ishtar Terra (at 70.4º N, 27.5º E), and the global coverage during high altitude parts of the orbit. At the Beginning of Mission, the pericenter latitude is near 70º. The height of apocenter is selected as a function of the available propellant; it is preferred to have a value as small as possible. The Data and Telemetry recovery from the single ground station Cebreros in Spain is maximised by selecting an orbital period near 24 h.

All trajectory data presented in this report is bases on numerical integration using all relevant perturbing forces, and using the JPL planetary ephemerides.

3.2 Venus Facts

Venus is the second planet from the Sun and the sixth largest. Its orbit lies in a near circular orbit, it has the smaller eccentricity of all planets, with an inclination with respect to the ecliptic of 3.4°. The orbital period is about 224.7 days. Venus rotation about its axis is very slowly with a sidereal rotation period of 243.7 days in a retrograde motion. The equatorial system of reference for Venus is defined in R 5, and the standard reference atmosphere in R 7.

	Venus	Ratio Venus/Earth
Equatorial radius (km)	6051.8	0.949
GM ($x10^6$ km^3/s^2)	0.3249	0.815
J_2 ($x10^6$)	4.458	0.004
Semi-major axis (10^6 km)	108.21	0.723
Synodic period (days)	583.92	
Sidereal orbit period (days)	224.7	0.615
Mean Orbital Vel. (km/s)	35.02	1.176
Obliquity to orbit (deg)	177.36	
Orbit inclination (deg)	3.39	--
Length of day (hrs)	2802.0	116.75
Sidereal rotation (hrs)	-5832.5	243.686

Table 2: Some Venus – Earth bulk and orbital parameters

3.3 Selected Mission Requirements

The following requirements from A 1 are important for the mission analysis of Venus Express:

MISS 005 The spacecraft shall be designed to obtain the operational orbit around Venus nominally with chemical propulsion in order to start science operations in the shortest possible time.

MISS 010 The spacecraft shall be capable of performing a Venus Orbit Insertion manoeuvre using a back-up mode with reaction control thrusters in case of main engine failure.

MISS 015 The spacecraft shall be capable to perform an injection error correction manoeuvre at any time within 7 days after launch (estimated 15 m/s at 99% probability)

MISS 020 The spacecraft shall be capable to perform midcourse corrections at any time before the Venus capture manoeuvre (estimated at < 5 m/s over 4 manoeuvres).

MISS 025 The spacecraft shall provide the resources to operate the spacecraft and the relevant instruments during two Venusian days (~500 Earth days) after the Venus operational orbit has been achieved.

MISS 070 Spacecraft consumables shall be sized to account for the case of extended mission duration of an additional 2 Venusian days (~500 Earth days) after the nominal mission.

MISS 075 The spacecraft shall be capable of maintaining the operational orbit pericenter altitude between 250 km and 400 km.

MISS 085 The final operational orbit around Venus shall be …. Inclination = 90 degrees.

LAUN 001 The spacecraft shall be fully compatible with a launch by Soyuz-Fregat and the operations and safety requirements …..

LAUN 003 The spacecraft nominal launch mass including launch vehicle adaptor shall be 1270 kg

LAUN 015 If at the time of launch the launcher capacity is exceeding the current estimates, it shall be possible to increase the propellant load with a maximum of 20% of the nominal propellant load or to the propellant tank capacity (whichever is less) without re-qualification of the spacecraft or launch vehicle adaptor.

SCIE 050 The spacecraft design and any payload to be carried shall comply with the requirements of COSPAR regulation concerning planetary protection, R 4, as a category I mission.

3.4 Ground Stations

The Ground Stations performing telemetry, telecommand and tracking operations will be the ESA 35 m Cebreros station that will be used throughout all mission phases, and the ESA 35 m New Norcia station as a backup. The 15 m Kourou station will complement the support during the Near-Earth mission phase and during the Venus Orbit Insertion phase. The NASA Deep Space Network, DSN, 34 m network will provide support for back up and for critical mission phases as needed, and the 70 m DSN network for emergency cases.

3.5 Soyuz-Fregat Launcher

A Soyuz – Fregat launch vehicle will launch Venus Express from Baikonur. The launcher has three stages and a Fregat upper stage that support multiple firing of the main engine with long coast arcs (several hours). Some improvements have been made starting from the original version that was used to launch Cluster II; the increase in the performances of the first and second stage engines is already available and it will be used by Mars Express, while Fregat Evolution, third stage engine performances improvement, digital avionics, ST Fairing are improvements that could be available by 2003-2004.

Once all the improvements are available, the performance of Soyuz-Fregat, as presented by STARSEM, R 1, R 2, is given in Figure 3-1:.

Since Venus Express is using the same launcher, and the same spacecraft bus as Mars Express, the wet mass of the spacecraft at launch cannot be very different from the one of Mars Express, since, otherwise, the spacecraft will need to be re-qualified for a higher wet mass with the consequent increase in the cost of the mission. Therefore, the maximum wet mass of Venus Express, including adaptor, has been fixed to 1270 kg. STARSEM has still to confirm the performance of Soyuz-Fregat for this mission.

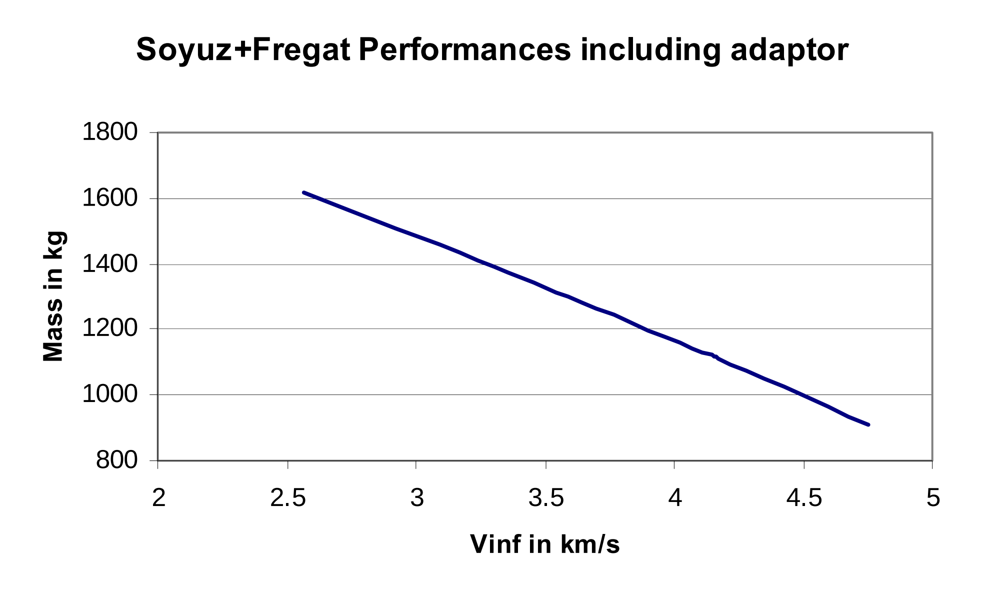

Figure 3-1: Performance of Soyuz-Fregat after modifications scheduled for 2004.

3.5.1 Sequence of Events

The Soyuz-Fregat trajectory is defined by the target escape orbit and the launch vehicle constraints leading to the following sequence of events:

- End of vertical ascent at T_0 + 8 s
- First/Second Stage separation at T_0 + 1 min 58 s
- Fairing jettisoning at T_0 + 4 min 14 s
- Second/Third Stage separation at T_0 + 4 min 47 s
- Aft section separation at T_0 + 4 min 58 s
- Nose Module separation at T_0 + 8 min 49 s
- First Fregat burn at T_0 + 8 min 54 s
- First Fregat cut-off at T_0 + 10 min 9 s
- Second Fregat burn depends on the target orbit, but will be more than 1 h after T_0

The launch management is made from Golitsino Flight Control Center, using the ground stations at Baikonur, Kolpashevo, Barnaul, Ienisseisk, Ulan Ude, Svobodny, Ussuriisk, Khimki, Tsheikovo and Krasnoye.

The Fregat Information Center at NPO – Lavochkin will make the interface with STARSEM and ESA.

The latest information provided by STARSEM on the performances of Soyuz–Fregat indicates that a spacecraft + LVA with a mass of up to 1270 kg can be launched into the required trajectory to Venus for launch days up to Nov. 24, 2005.

3.5.2 Eclipses

There are no eclipses after injection into the Earth escape orbit for launch in the period 26/10 to 24/11/2005.

3.5.3 Ground Station Coverage

The ground stations that will support the Launch and Early Orbit Phase of Venus Express are: Cebreros, New Norcia, and Kourou. Figure 3-2 presents the ground track of the spacecraft orbit just after the separation and for about one day for four representative days of the launch window. The ground track curves itself towards the West over the Pacific passing North to Australia.

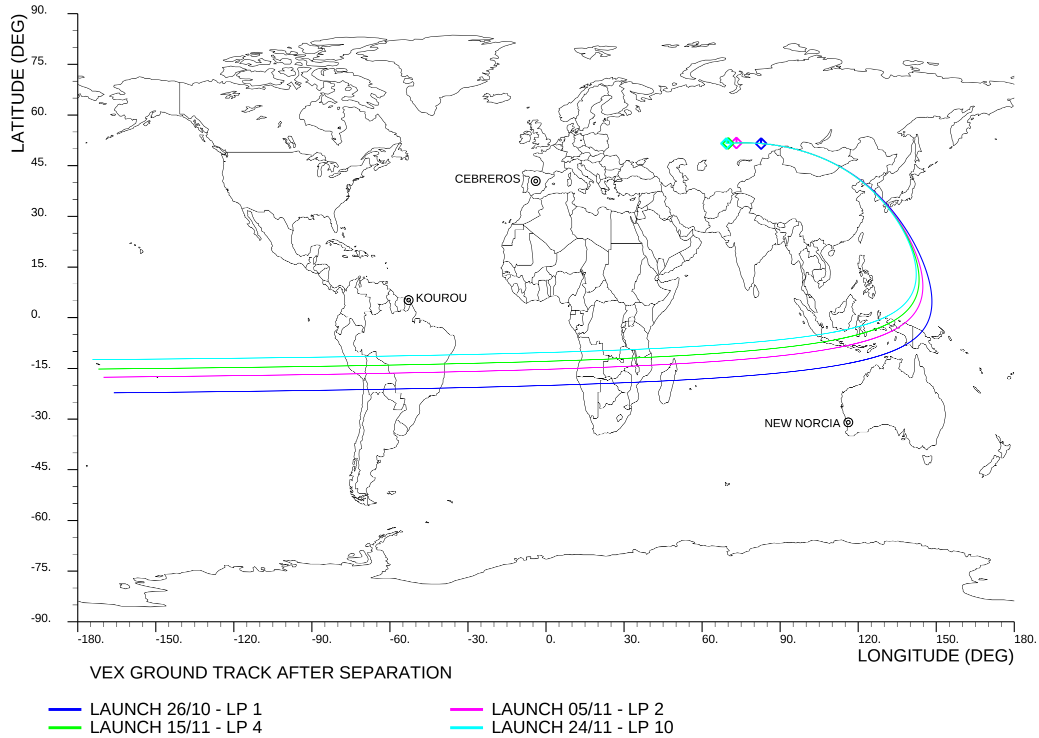

Figure 3-2: Ground track after injection for different launch days.

The visibility periods of Venus Express after injection into the Earth escape orbit are shown in Figure 3-3 for a launch on 26/10/2005, and in Figure 3-5 for a launch on 24/11/2005.

For a launch on 26/10/2005, New Norcia get contact with the spacecraft, at elevation of more than 10º, about 25 minutes after injection, while Cebreros get AOS after 8 h 41 min, and

Kourou 10 h 34 min. For a launch on 24/11/2005 the values are New Norcia 29 min., Cebreros 7 h 24 min., Kourou 9 h 56 min.

For the sequence New Norcia, Cebreros, and Kourou, there is continuous coverage with overlap up to the time of first LOS from Kourou. Thereafter there is a visibility gap of 102 min for a launch on 26/10/2005 and of 118 min for a launch on 24/11/2005.

For a launch on 26/10/2005 Figure 3-4 shows the elevation from the ground stations. The maximum elevation from New Norcia is above 70 deg for the first pass and above 80 deg afterwards, while the maximum elevation from Kourou is above 60 deg and from Cebreros above 25 deg. For a launch on 24/11/2005, Figure 3-6, the higher departure declination results in a lower maximum elevation from New Norcia, 60 deg for the first pass and 70 deg afterwards, and higher maximum elevations from Kourou and Cebreros above 70 deg and 35 deg, respectively.

The sequence of the visibility passes from the three ground stations and the visibility gaps during the early orbit is presented in Table 3. The data begin exactly at injection, which is during a visibility gap (NOV = No visibility) before the acquisition from New Norcia. The table shows the beginning, the end and the duration of all the events. Similar tables for launches on days 05/11/2005, 15/11/2005 and 24/11/2005 are presented in the Appendix B – Section 16.

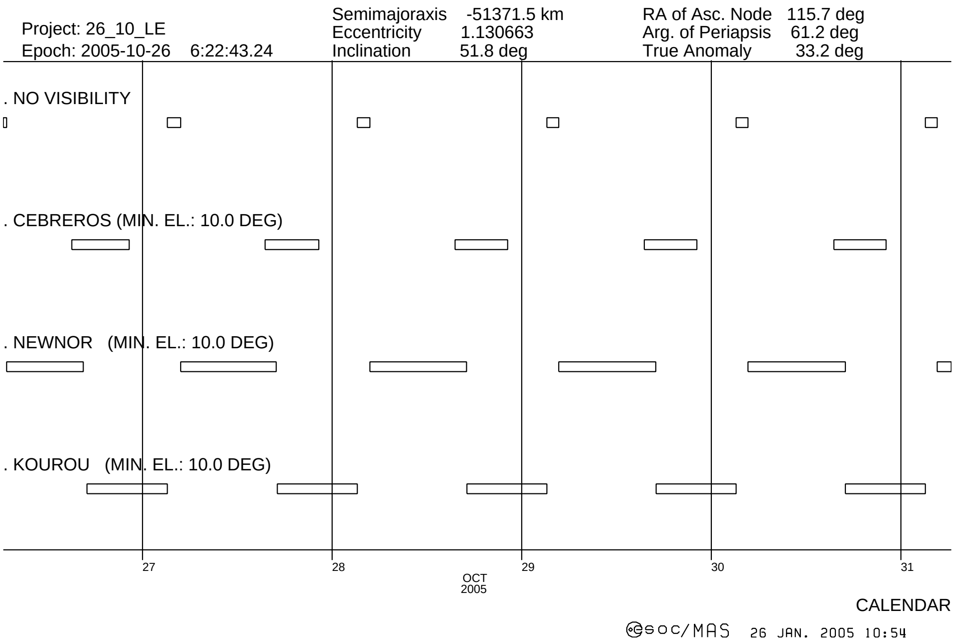

Figure 3-3: Visibility periods for the first 5 days after launch on 26/10/2005.

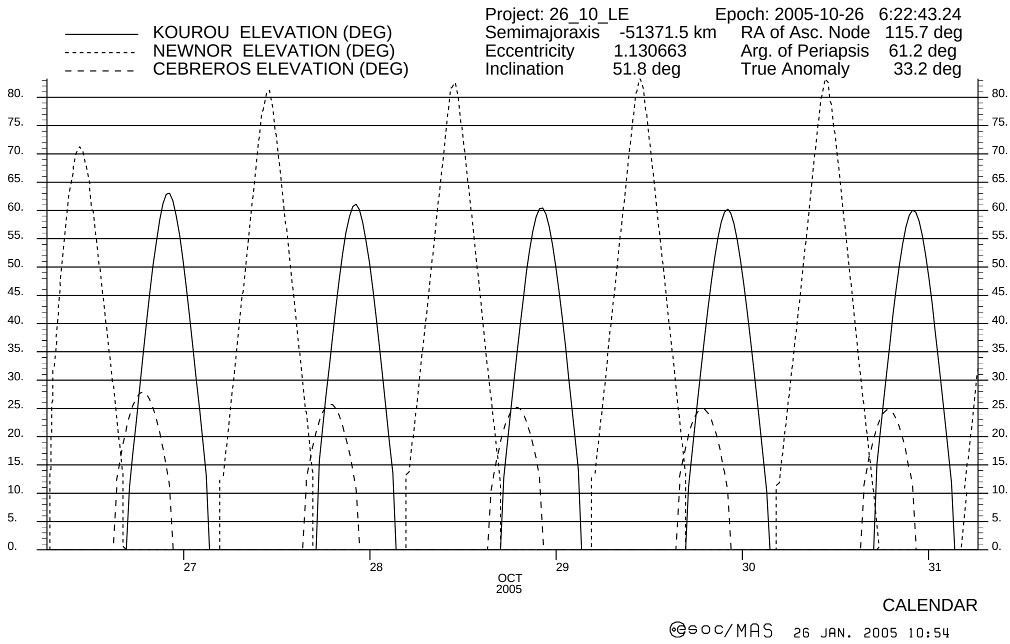

Figure 3-4: Elevation from ground stations for the first 5 days after launch on 26/10/2005.

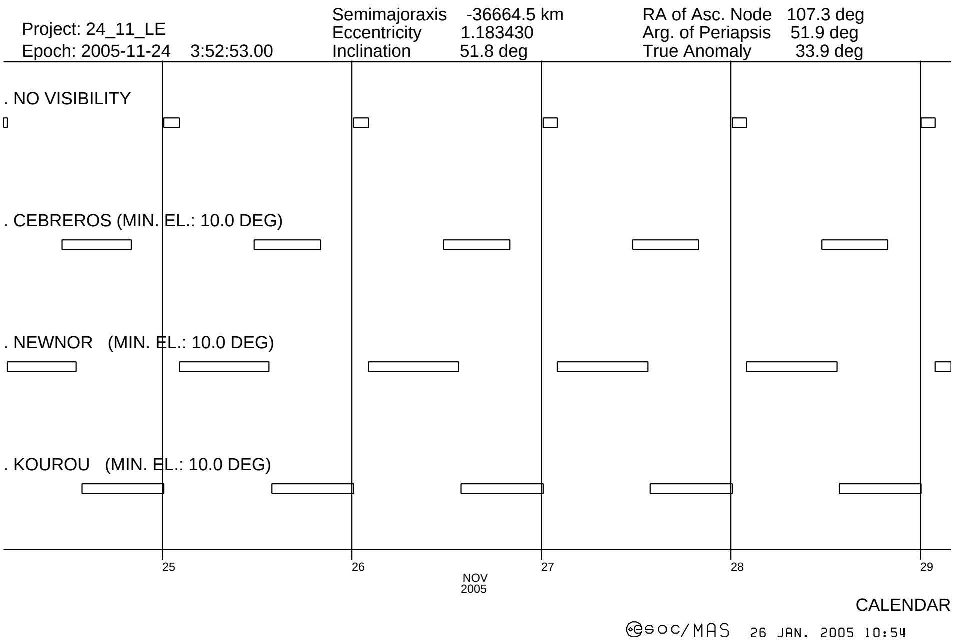

Figure 3-5: Visibility periods for the first 5 days after launch on 24/11/2005

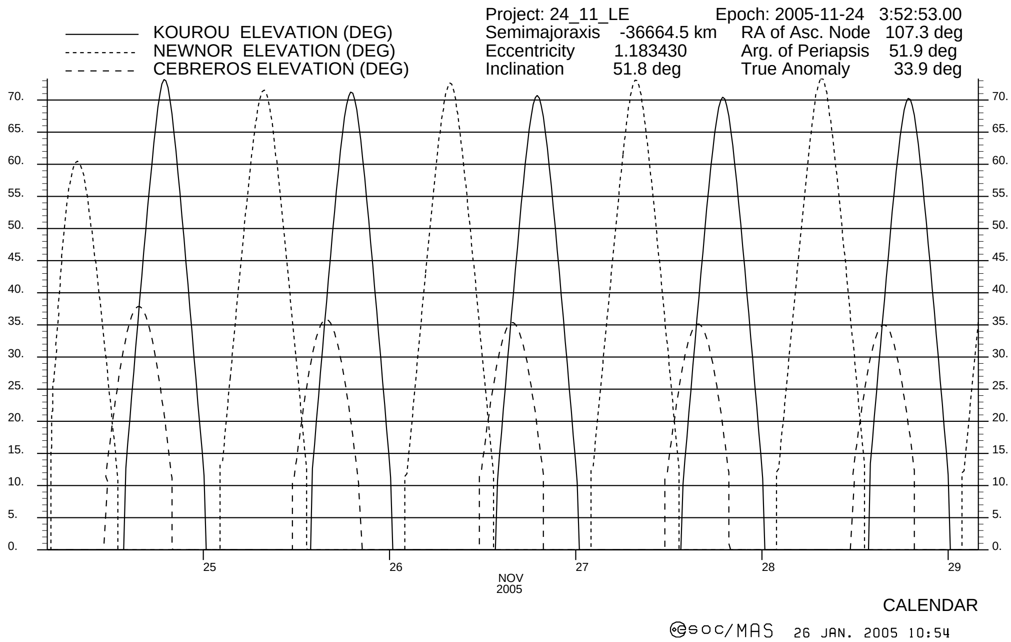

Figure 3-6: Elevation from ground stations for the first 5 days after launch on 24/11/2005.

DATE	HOUR (UTC)	TIME (HOURS)	EVENT			
05-10-26	06:21:39.0	0	BEGIN			
05-10-26	06:47:05.1	0.42	EXIT	NOV	DURATION	00:25:26
05-10-26	06:47:05.1	0.42	AOS	NEWNOR		
05-10-26	15:02:39.7	8.68	AOS	CEBRER		
05-10-26	16:28:24.7	10.11	LOS	NEWNOR	DURATION	09:41:19
05-10-26	16:55:43.7	10.57	AOS	KOUROU		
05-10-26	22:18:27.3	15.95	LOS	CEBRER	DURATION	07:15:47
05-10-27	03:06:52.3	20.75	LOS	KOUROU	DURATION	10:11:08
05-10-27	03:06:52.3	20.75	ENTRY	NOV		
05-10-27	04:49:12.3	22.46	EXIT	NOV	DURATION	01:42:19
05-10-27	04:49:12.3	22.46	AOS	NEWNOR		
05-10-27	15:28:47.0	33.12	AOS	CEBRER		
05-10-27	16:55:33.2	34.57	LOS	NEWNOR	DURATION	12:06:20
05-10-27	17:02:48.6	34.69	AOS	KOUROU		
05-10-27	22:16:26.1	39.91	LOS	CEBRER	DURATION	06:47:39
05-10-28	03:10:57.1	44.82	LOS	KOUROU	DURATION	10:08:08
05-10-28	03:10:57.1	44.82	ENTRY	NOV		
05-10-28	04:44:55.8	46.39	EXIT	NOV	DURATION	01:33:58
05-10-28	04:44:55.8	46.39	AOS	NEWNOR		
05-10-28	15:31:44.1	57.17	AOS	CEBRER		
05-10-28	16:58:47.2	58.62	LOS	NEWNOR	DURATION	12:13:51
05-10-28	17:01:49.9	58.67	AOS	KOUROU		
05-10-28	22:13:08.6	63.86	LOS	CEBRER	DURATION	06:41:24
05-10-29	03:09:59.9	68.81	LOS	KOUROU	DURATION	10:08:10
05-10-29	03:09:59.9	68.81	ENTRY	NOV		
05-10-29	04:41:02.2	70.32	EXIT	NOV	DURATION	01:31:02
05-10-29	04:41:02.2	70.32	AOS	NEWNOR		
05-10-29	15:30:59.5	81.16	AOS	CEBRER		
05-10-29	16:58:18.3	82.61	LOS	NEWNOR	DURATION	12:17:16
05-10-29	16:59:22.0	82.63	AOS	KOUROU		
05-10-29	22:09:30.6	87.8	LOS	CEBRER	DURATION	06:38:31
05-10-30	03:07:41.4	92.77	LOS	KOUROU	DURATION	10:08:19
05-10-30	03:07:41.4	92.77	ENTRY	NOV		
05-10-30	04:37:16.5	94.26	EXIT	NOV	DURATION	01:29:35
05-10-30	04:37:16.5	94.26	AOS	NEWNOR		
05-10-30	15:29:01.4	105.12	AOS	CEBRER		
05-10-30	16:56:20.6	106.58	AOS	KOUROU		
05-10-30	16:56:23.6	106.58	LOS	NEWNOR	DURATION	12:19:07
05-10-30	22:05:51.7	111.74	LOS	CEBRER	DURATION	06:36:50
05-10-31	03:04:47.2	116.72	LOS	KOUROU	DURATION	10:08:26
05-10-31	03:04:47.2	116.72	ENTRY	NOV		
05-10-31	04:33:28.4	118.2	EXIT	NOV	DURATION	01:28:41
05-10-31	04:33:28.4	118.2	AOS	NEWNOR		
05-10-31	06:21:39.0	120	END			

Table 3: Sequence of visibility events for the first 5 days after launch on 26/10/2005

Figure 3-7 presents the evolution of the distance rate of the spacecraft w.r.t. the Earth for different launch days. A small jump can be appreciated 2 days after separation that corresponds to the deterministic manoeuvre of the launcher program correction. There is a maximum of about 6 Km/s close to the separation, Figure 3-8, and then it decreases tending to the velocity at infinity after a couple of days.

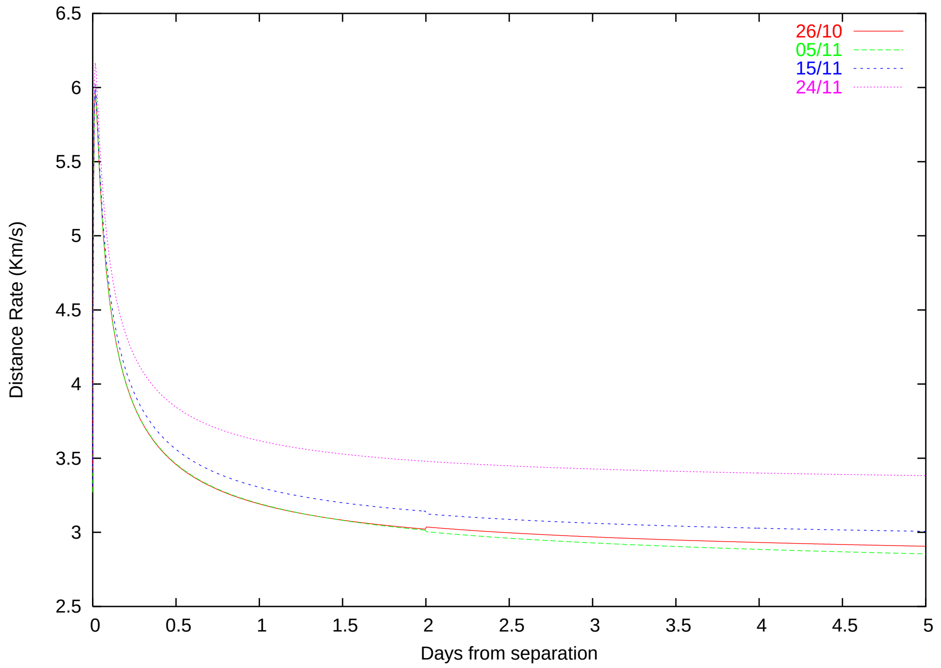

Figure 3-7: Distance rate for the first 5 days after launch

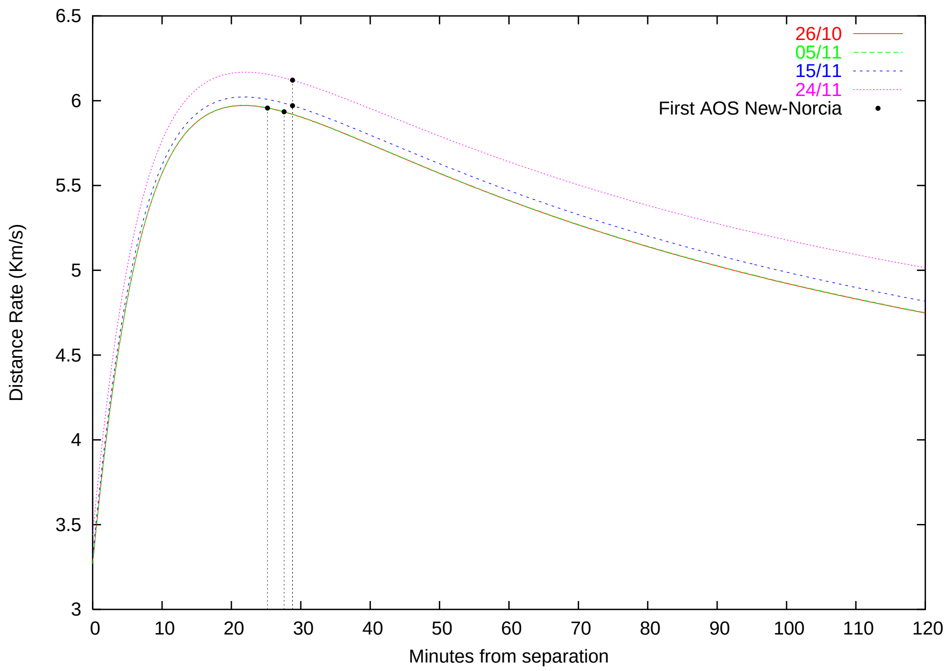

Figure 3-8: Detail of the distance rate for the first 120 minutes after separation

Figure 3-9 presents the evolution of the Earth-S/C-Sun angle during the first 5 days after launch for different launch days. This angle has an optimum close after separation and then tends to stabilize to a value between 85 deg for launch on 26/10/2005 and 127 deg for launch on 24/11/2005. Figure 3-10 presents the evolution of the angle for the first 120 min after separation showing the precise geometric configuration when the first AOS from New Norcia ground station should occur.

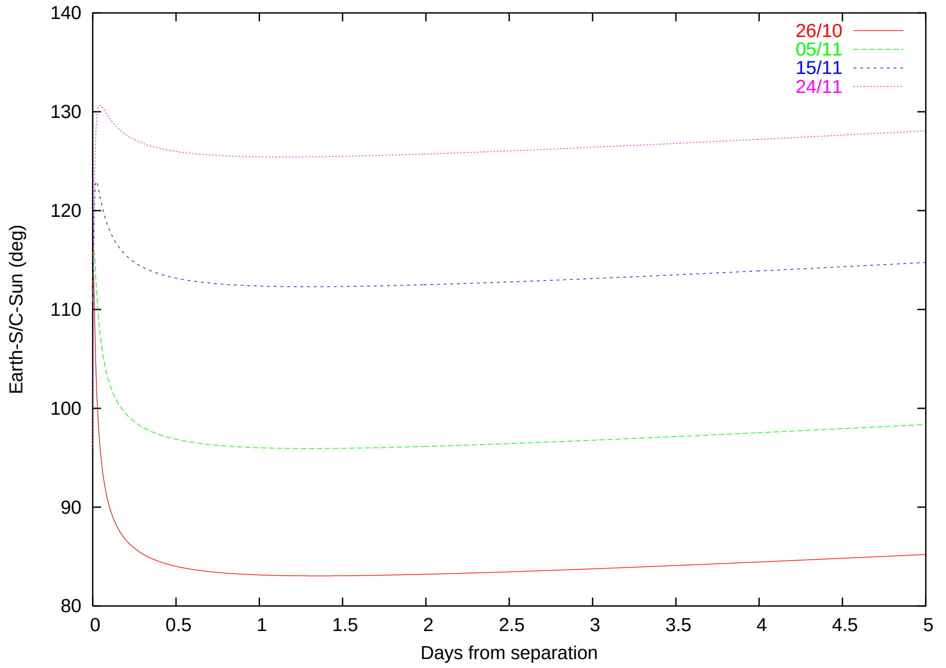

Figure 3-9: Angle Earth-S/C-Sun for the first 5 days after launch

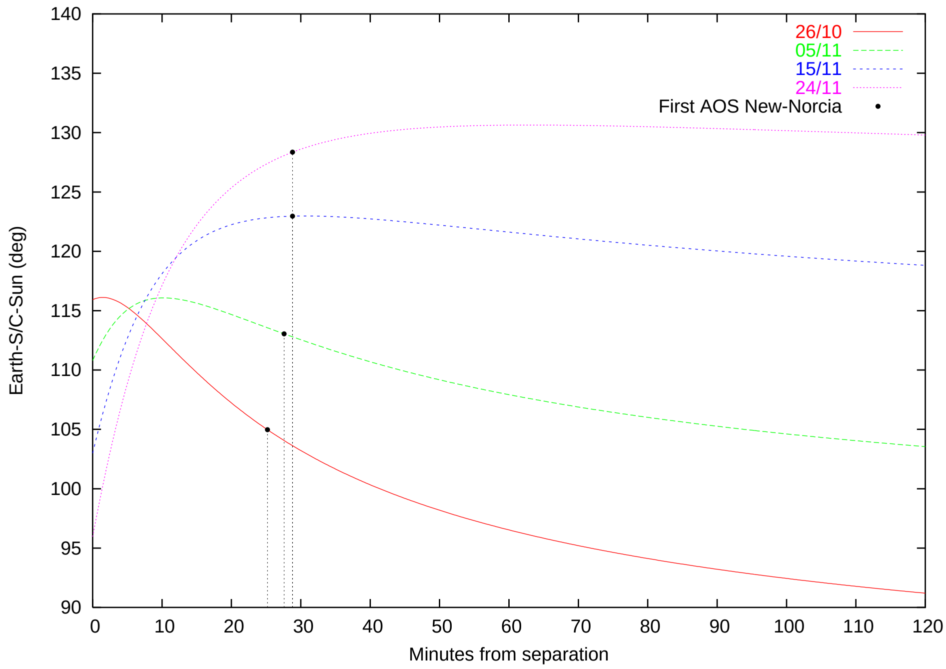

Figure 3-10: Detail of the Earth-S/C-Sun angle for the first 120 minutes after separation

3.5.4 Launcher Injection Errors

The upper stage Fregat will deliver the spacecraft into the escape orbit with some inaccuracy. This inaccuracy is described in terms of the covariance matrix of the position and velocity dispersion errors at a given point of the departure hyperbola. Starsem has provided this covariance matrix for the launch of VEX in reference R 6. The covariance matrix is referred to the pericentre and provided in the reference frame given by the directions:

Along – Track (A-T) : direction of the inertial velocity vector.

Cross – Track (C-T) : direction perpendicular to the orbital plane, that is, the direction along the momentum vector.

Radial (R) : direction of the position vector from the center of the Earth.

1-σ	Position Vector			Velocity Vector		
	A-T (Km)	C-T (Km)	R (Km)	A-T (m/s)	C-T (m/s)	R (m/s)
A-T(Km)	138.4	-6.7	-2.8	-5.5	44.6	-158.6
C-T (Km)		2.10	0.	0.40	-8.40	7.70
R (Km)			0.20	0.	-0.10	3.10
A-T (m/s)				0.70	-2.70	6.50
C-T (m/s)					72.2	-52.2
R (m/s)						196.8

Table 4: Launcher dispersion covariance matrix

The following table shows the standard deviations that correspond to the above covariance matrix together with the errors of the hyperbolic departure velocity osculating at the injection point.

Position Vector		
A-T (Km)	C-T (Km)	R (Km)
11.76	1.45	0.45
Velocity Vector		
A-T (m/s)	C-T (m/s)	R (m/s)
0.84	8.50	14.03
Hyperbolic Velocity		
V_∞ (m/s)	Declination (deg)	Right Ascension (deg)
3.7	0.052	0.072

Table 5: 1-σ Injection Errors

4. TRANSFER TO VENUS IN 2005

For a launch in 2005, Figure 4-1 presents the required departure excess velocity from the Earth to reach Venus. The minimum value is less than 2.8 km/s for a launch beginning of November, and arrival end of April 2006. Since the synodic period of Venus is about 1.6 years, good possibilities of transfer Earth to Venus do exist with that frequency. In fact, a launch in 2007 is more favourable in the sense that less energy is required for the transfer and injection into Venus orbit, and, therefore, more propellant would be available for reducing the apocenter of the operational orbit. However, for the selected orbit inclination of 90º, the latitude of pericenter of the operational orbit will be either near the South Pole or at low northern inclination (< 30º). The launch in 2007 will be analysed in the future to assess the compatibility of the spacecraft design with the requirements of a possible launch delay.

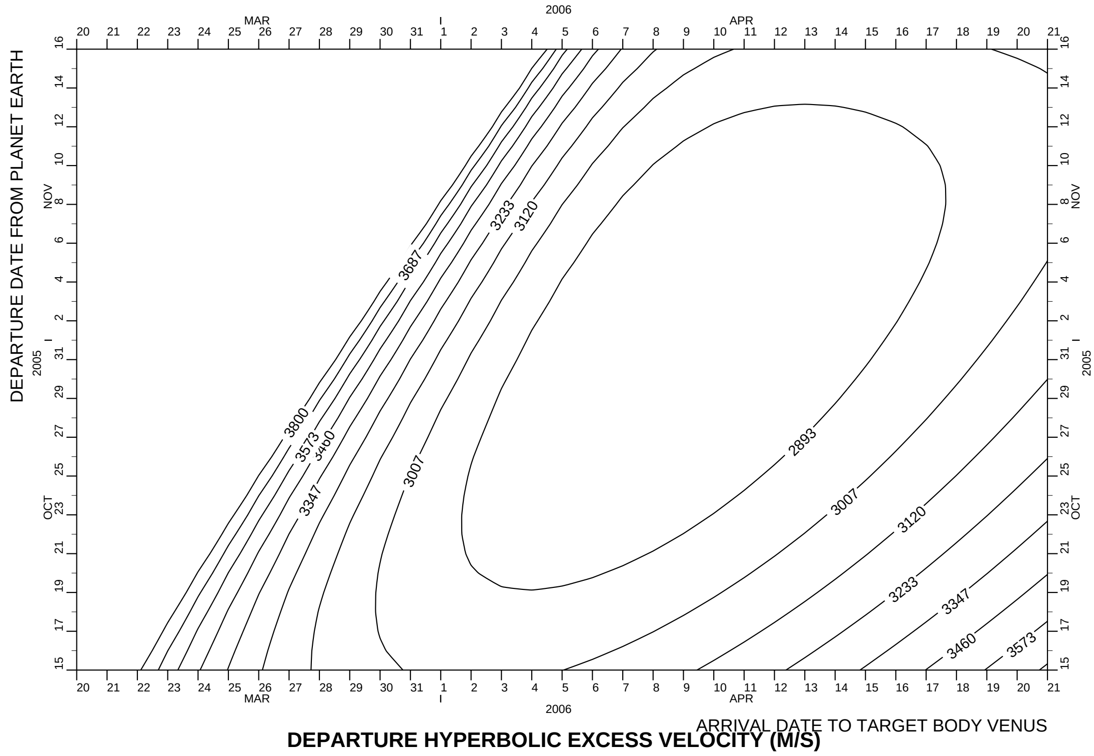

Figure 4-1: Departure hyperbolic excess velocity at Earth required for a transfer Earth to Venus. Departure/Arrival dates are near optimal for launch in 2005

At this time, due to programmatic reasons, only a mission with launch in 2005 is considered.

Figure 4-2 presents the arrival velocity at Venus for the same launch and arrival period as Figure 4-1.

The transfer strategy assumes a direct injection of the spacecraft by Soyuz-Fregat into Earth escape trajectory, followed by a direct capture into orbit around Venus. The operational orbit around Venus must have an inclination of 90°, and a pericenter height near 250 km. To minimize the high gravity loss of the capture manoeuvre, the apocenter height of the capture orbit is selected rather big, corresponding to an orbital period of about 5.5 days.

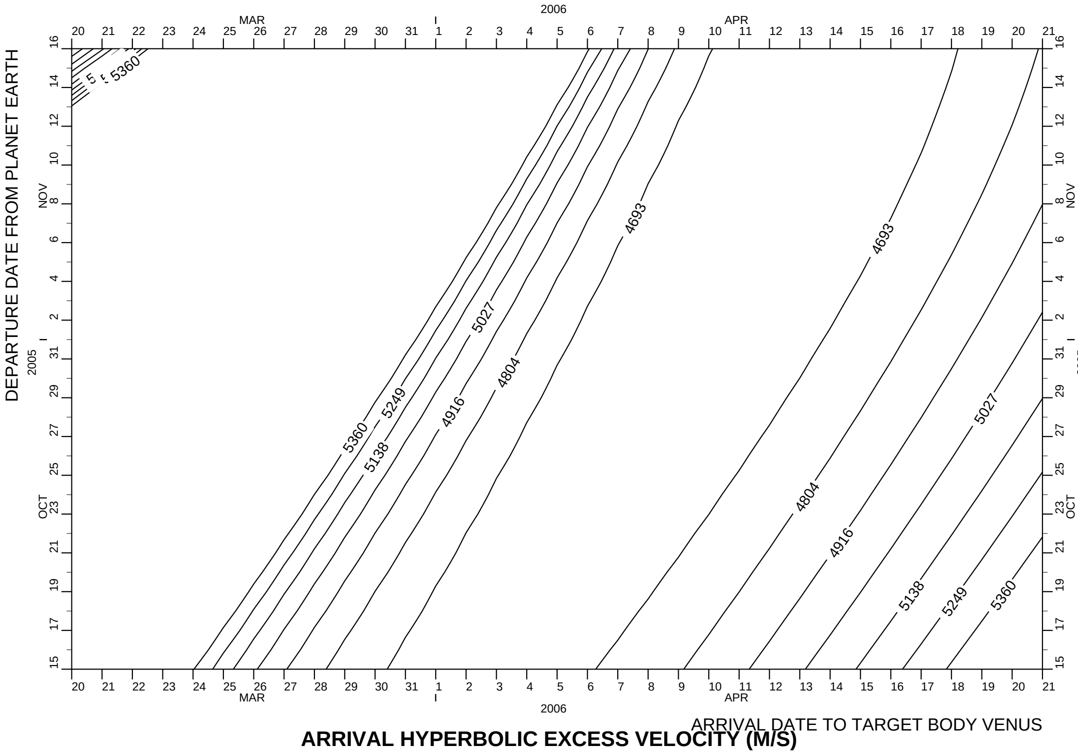

Figure 4-2: Arrival hyperbolic excess velocity at Venus after a transfer Earth to Venus for a launch in 2005. Departure/Arrival dates are near optimal for launch in 2005

An optimisation of the transfer supposing an impulsive manoeuvre for Venus Insertion Manoeuvre at pericenter results in the data given in Table 6. The Earth escape velocity at infinity varies between 2.774 km/s and 3.302 km/s, with a minimum for launch near 02/11/2005. The Venus arrival velocity at infinity varies much less from 4.60 km/s to 4.62 km/s. The arrival date at Venus varies between April 6, and April 14, 2006.

The mass and the radius of Venus are very similar to the one of the Earth, Table 2. For the same altitude, the orbital speed around Venus, or the velocity at pericenter of a hyperbolic approach trajectory are very much like in the case of the Earth. Therefore, a big manoeuvre is needed to insert the spacecraft from the hyperbolic approach orbit into the operational orbit.

The impulsive manoeuvre needed to inject into the operational orbit is greater than 1100 m/s. This manoeuvre is greater than in the case of Mars Express. Because of the limited thrust level available, 414 N, it will be very inefficient to perform it in a single burn. A single burn will imply very long thrusting time with gravity losses of almost 20%. Figure 4-3 graphically present the information of Table 6.

Earth Departure				Venus Arrival	
Date	Lift Off	V_∞ Km/s	δ_∞ deg	Date	V_∞ Km/s
2005/10/26	4:41:16	2.802	-26.353	2006/04/06	4.621
2005/10/27	4:36:59	2.795	-25.975	2006/04/06	4.619
2005/10/28	4:32:40	2.788	-25.612	2006/04/07	4.617
2005/10/29	4:28:19	2.783	-25.252	2006/04/07	4.615
2005/10/30	4:23:56	2.779	-24.876	2006/04/08	4.613
2005/10/31	4:19:31	2.776	-24.473	2006/04/08	4.612
2005/11/01	4:15:05	2.775	-24.100	2006/04/08	4.610
2005/11/02	4:10:37	2.774	-23.699	2006/04/09	4.608
2005/11/03	4:06:08	2.775	-23.310	2006/04/09	4.607
2005/11/04	4:01:42	2.777	-22.889	2006/04/09	4.605
2005/11/05	3:57:28	2.781	-22.450	2006/04/10	4.604
2005/11/06	3:54:15	2.790	-21.844	2006/04/10	4.603
2005/11/07	3:48:02	2.777	-20.530	2006/04/10	4.603
2005/11/08	3:38:35	2.780	-21.395	2006/04/11	4.600
2005/11/09	3:34:17	2.795	-21.033	2006/04/11	4.599
2005/11/10	3:29:22	2.807	-20.650	2006/04/11	4.598
2005/11/11	3:24:11	2.822	-20.255	2006/04/11	4.599
2005/11/12	3:18:51	2.840	-19.864	2006/04/12	4.598
2005/11/13	3:13:26	2.861	-19.480	2006/04/12	4.599
2005/11/14	3:08:00	2.884	-19.107	2006/04/12	4.599
2005/11/15	3:02:33	2.911	-18.718	2006/04/13	4.600
2005/11/16	2:57:09	2.941	-18.329	2006/04/13	4.601
2005/11/17	2:51:48	2.975	-17.952	2006/04/13	4.603
2005/11/18	2:46:32	3.013	-17.559	2006/04/13	4.604
2005/11/19	2:41:20	3.053	-17.185	2006/04/13	4.607
2005/11/20	2:36:15	3.097	-16.799	2006/04/14	4.610
2005/11/21	2:31:15	3.144	-16.408	2006/04/14	4.612
2005/11/22	2:26:22	3.193	-16.015	2006/04/14	4.616
2005/11/23	2:21:34	3.247	-15.616	2006/04/14	4.619
2005/11/24	2:16:53	3.302	-15.223	2006/04/14	4.623

Table 6: Time and hyperbolic velocity of departure and arrival in case of optimum transfer

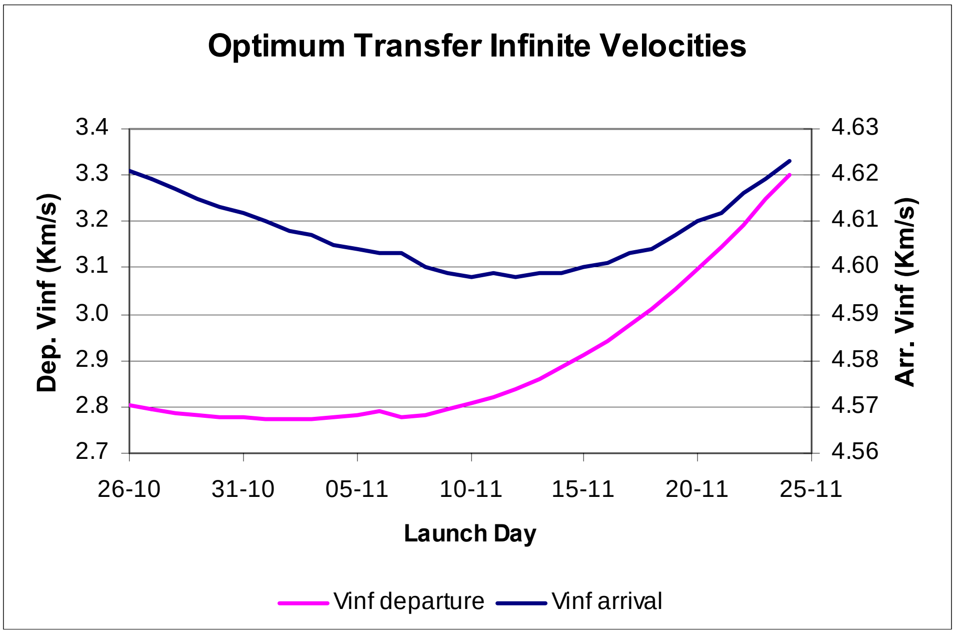

Figure 4-3: Optimum transfer hyperbolic velocities for launch from 26/10/2005 to 24/11/2005

The baseline capture strategy has suffered several changes from the last issue of this document. The initial strategy was planned as follows:

> A capture manoeuvre with the main engine injects the spacecraft from the incoming hyperbola with a nominal pericenter height of 500 km to an elliptical capture orbit with a low pericenter height and a period of 5.5 days (apocenter radius of 266000 km). During the thrust arc, which extends from almost -90º to +90º of true anomaly, the spacecraft follows an attitude steering mode that keeps the thrust aligned with and opposite to the velocity vector and the minimum height over the Venus surface is above 250 km. The capture orbit is selected as a compromise between the reduction of gravity losses and pericenter height increase due to the gravity effect of the Sun. The argument of pericenter is reduced by about 1° in this thrusting phase.
>
> During the next orbit the effect of the Sun gravitational attraction increases the height of pericenter to the required 250 km. Just before the next pericenter pass, a second thrusting arc with the main engine is initiated in an attitude steering mode as in the previous thrusting arc in order to inject the spacecraft into the operational orbit of 1-day period.

This strategy is a trade-off between the manoeuvre efficiency and the time needed for Venus Orbit Insertion. In order to maximise the effect of the two manoeuvres, the attitude of the spacecraft is controlled such that the thrust is always in the opposite direction to the velocity. The Attitude Control of the Venus Express spacecraft has this capability, which has already been proved by the Mars Express mission.

However, the strategy has been modified because of a limitation on the use of the main engine. The Venus Express tanks draining mechanisms from the beginning of the Venus insertion manoeuvre until the end of the apocenter-lowering manoeuvre is analysed in R 9. The analysis shows that, when the propellant in the tanks is below 60.1 Kg, the main engine can not be used. This forces to stop the apocenter lowering manoeuvre with the main engine before the operational orbit has been reached. Therefore, the remaining apocentre lowering has to be achieved by manoeuvres with the reaction control thrusters. 4 of these thrusters can function together in modulated mode to provide a thrust of 11.2 N and up to 4 consecutive manoeuvres can be necessary to reach the 1-day orbit.

An analysis about the apocenter altitude of the intermediate capture orbit has been carried out and presented in R 8. The recommendation is to increase slightly the period of the capture orbit to 6.2 days.

The reference R 11 proposes an alternative capture strategy in order to obtain an operational orbit with the same argument of pericentre for the whole launch window. The argument of the pericentre is reduced specially for the last days of the launch window. This can be obtained by implementing asymmetrical manoeuvres with the main engine and by introducing a tilt angle of the thrust w.r.t. the anti-velocity vector. The period of the capture orbit is also incremented up to 10 days. The advantage consists of an operational orbit with a smaller need of pericentre control manoeuvres during the extended mission lifetime of 1000 days. The fuel margin is increased and is made positive even for the very last launch window days.

However, an intermediate approach has been presented in R 8 that does not require an operational orbit with the same initial argument of pericenter. The manoeuvres with the main engine are opposite to the velocity and a reduction of the argument of the pericenter is obtained with the asymmety of these manoeuvres. The delta-V required for pericenter control is diminished specially at the end of the launch window. Even with a capture orbit of 6.2 days, the fuel margin is positive for the whole launch window, therefore the extended mission lifetime of

1000 days can be guaranteed. Furthermore, the total duration that the spacecraft can be controlled in orbit around Venus has been optimized for every launch day.

Another effect to be taken into account is the correction manoeuvre at Earth departure required to reduce the number of launch programs, see section 12.1. This manoeuvre can vary from almost 0 to about 26 m/s, therefore the spacecraft will reach Venus with a different mass depending on the launch day. The following sections present the results of the Venus Orbit Insertion and Apocentre Lowering considering the launch program effect, an estimation of the stochastic delta-V required for navigation up to the Venus arrival (20 m/s), the limitation on the use of the main engine and the optimum capture strategy in order to maximise the operational lifetime.

The current baseline capture strategy is as follows:

Capture manoeuvre with the main engine at the pericenter of the arrival hyperbola to insert the spacecraft into a 6.2 days capture orbit. The thrust is directed against the velocity and it is asymmetrical with respect to the pericentre in order to modify the argument of the pericenter. At the apocenter of the capture orbit a small correction manoeuvre tunes the pericenter altitude that has been increased due to the Sun gravity pull.

An apocentre lowering manoeuvre with the main engine is implemented at the next pericenter up to its limitat. Again the thrust is against the velocity and the thrust arc is asymmetrical, beginning close to the pericentre and helping to reduce the initial argument of pericenter.

Consecutive apocentre lowering manoeuvres with the RCT to reach the operational period of 1 day. These manoeuvres are symmetric as the thrust is limited and they can produce a very small effect on the argument of the pericenter. Waiting orbits can be used between two apocentre lowerings in order to determine the orbit and plan the next manoeuvre or to phase the pericentre pass with the visibility from the ground station.

The gravity losses are important for the two manoeuvres implemented with the main engine, with losses of about 13%. The apocentre lowering with the RCT is split in several small manoeuvres to limit the effect of the gravity losses to about 8%. Of course, the minimum altitude of 250 km above Venus surface is respected during all the thrust arcs.

Considering a limited number of launch programs, the actual departure and arrival conditions for the mission differ slightly from those in the Table 6. The following table shows the exact departure and arrival conditions for every launch day using the launch conditions agreed with STARSEM to use a reduced set of 10 launch programs, section 12.1

Figure 4-5 shows the departure and arrival hyperbolic velocities for the transfers considering the effect of the launch programs.

For a fixed inclination of the arrival hyperbola at Venus of 90 deg, the argument of pericenter of the arrival orbit is fixed. For the optimum transfer between the Earth and Venus the argument of pericenter varies linearly over the launch window from 100 deg for launch on 26/10/2005 to 114 deg for launch on 24/11/2005. This also means that the initial latitude of the pericenter varies about 14 deg depending on the launch day. Figure 4-6 presents the variation of the argument of pericenter at Venus arrival over the whole launch window period when the strategy of 10 launch programs is used. There is also an increase from 101 deg to 114 deg.

Figure 4-7 presents the variation of the right ascension of the ascending node at Venus arrival over the whole launch window period. The right ascension varies from 104 deg to 111.5 deg.

Earth Departure				Venus Arrival	
Date	Lift Off	V_∞ Km/s	δ_∞ deg	Date	V_∞ Km/s
2005/10/26	04:43:38.7	2.7855	-25.614	2006/04/06	4.6215
2005/10/27	04:37:42.4	2.7855	-25.614	2006/04/07	4.6192
2005/10/28	04:31:46.4	2.7855	-25.614	2006/04/07	4.6171
2005/10/29	04:25:36.0	2.7855	-25.613	2006/04/07	4.6153
2005/10/30	04:19:25.9	2.7855	-25.613	2006/04/07	4.6139
2005/10/31	04:13:10.7	2.7855	-25.613	2006/04/07	4.6128
2005/11/01	04:06:50.1	2.7855	-25.613	2006/04/08	4.6121
2005/11/02	04:00:23.6	2.7855	-25.613	2006/04/08	4.6119
2005/11/03	03:53:50.4	2.7855	-25.612	2006/04/08	4.6120
2005/11/04	03:47:09.4	2.7855	-25.612	2006/04/08	4.6127
2005/11/05	04:03:21.1	2.7904	-21.052	2006/04/10	4.6059
2005/11/06	03:57:04.2	2.7904	-21.051	2006/04/10	4.6036
2005/11/07	03:44:32.1	2.7904	-21.051	2006/04/10	4.6033
2005/11/08	03:39:30.3	2.7904	-21.051	2006/04/11	4.5999
2005/11/09	03:33:34.5	2.7904	-21.050	2006/04/11	4.5990
2005/11/10	03:26:40.7	2.7904	-21.050	2006/04/11	4.5986
2005/11/11	03:19:19.0	2.7904	-21.050	2006/04/11	4.5987
2005/11/12	03:19:32.8	2.8560	-19.502	2006/04/12	4.5983
2005/11/13	03:12:43.3	2.8560	-19.502	2006/04/12	4.5984
2005/11/14	03:04:40.2	2.8560	-19.502	2006/04/12	4.5990
2005/11/15	03:02:33.2	2.9248	-18.474	2006/04/13	4.5998
2005/11/16	02:55:14.4	2.9248	-18.474	2006/04/13	4.6010
2005/11/17	02:51:57.2	2.9945	-17.688	2006/04/13	4.6026
2005/11/18	02:44:46.0	2.9945	-17.688	2006/04/13	4.6044
2005/11/19	02:41:23.8	3.0708	-17.013	2006/04/14	4.6067
2005/11/20	02:34:20.2	3.0708	-17.013	2006/04/14	4.6093
2005/11/21	02:30:20.9	3.1388	-16.423	2006/04/14	4.6122
2005/11/22	02:25:27.3	3.1886	-16.031	2006/04/14	4.6155
2005/11/23	02:20:39.8	3.2415	-15.637	2006/04/14	4.6191
2005/11/24	02:15:58.5	3.2972	-15.240	2006/04/14	4.6231

Figure 4-4: Time and infinite velocity of departure and arrival of transfer with 10 launch programs

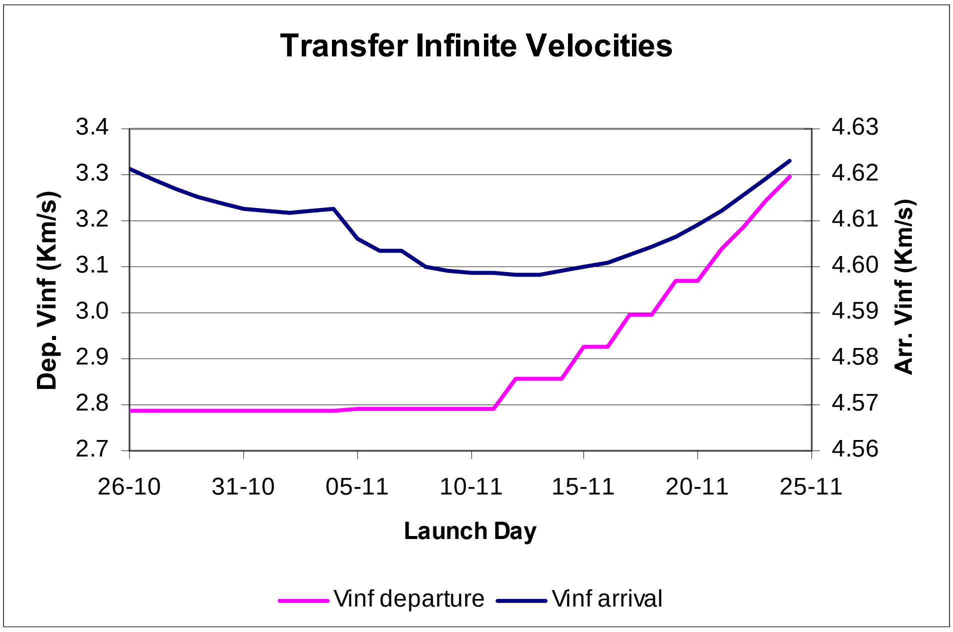

Figure 4-5: Hyperbolic velocities for launch from 26/10/2005 to 24/11/2005 using 10 launch programs

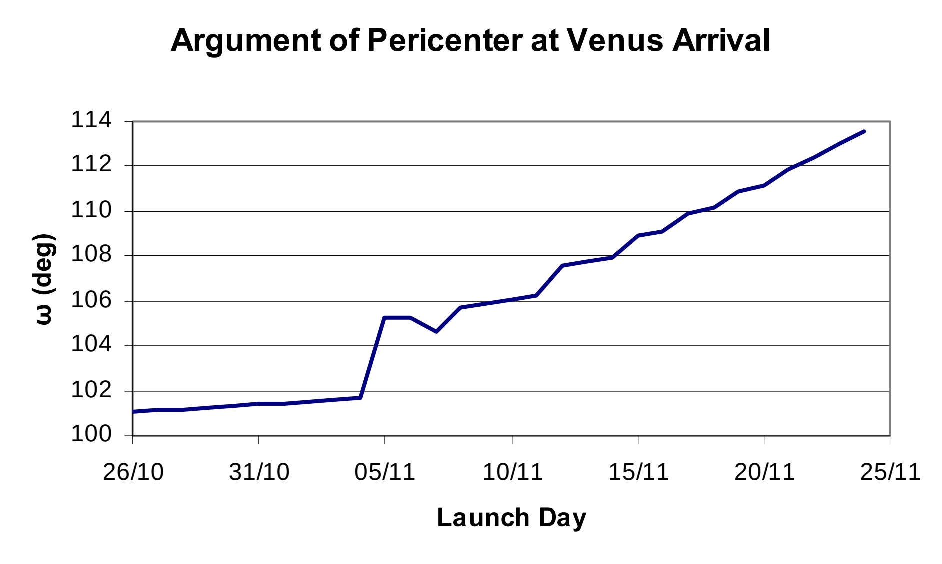

Figure 4-6: Argument of the pericenter of the arrival hyperbola at Venus over the launch period 26/10/2005 to 24/11/2005

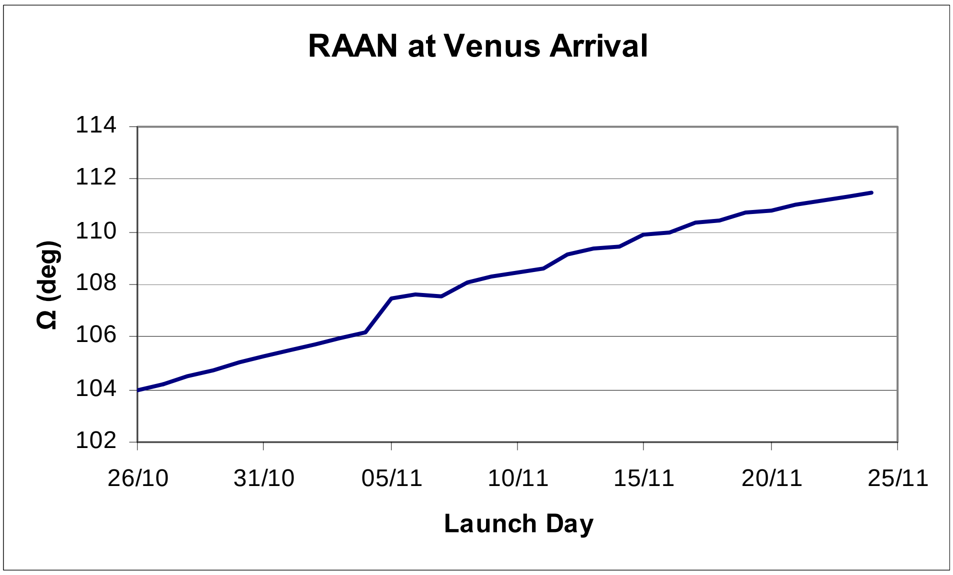

Figure 4-7: Right ascension of ascending node of the arrival hyperbola at Venus over the launch period 26/10/2005 to 24/11/2005

4.1 *Finite Thrust Venus Orbit Insertion into Capture Orbit*

The results of the analysis of the Venus Orbit Insertion into the capture orbit using finite thrust are presented in Table 7 and in Figure 4-8. The spacecraft mass at Venus arrival is computed assuming an initial wet mass of the spacecraft at launch of 1231.7 Kg (adaptor mass 38.3 Kg) and subtracting the ΔV of the launch program correction, 20 m/s for the launcher dispersion correction and interplanetary navigation and 4 m/s for ME and thrusters check-out. The specific impulse of the main engine is assumed 313 s. The equivalent Delta-V of the capture burn is in the range of 1276 to 1325 m/s equivalent to a propellant mass consumption from about 411 to 427 kg.

Launch	Thrust. Time (s)	TA_0 / TA_f (deg)	Capture orbit					M prop (kg)	ΔV (m/s)
			a (Km)	e	i	AN	PE		
2005/10/26	3079.7	-87.9 / 82.7	133078.2	0.95374	90.00	103.94	100.94	415.2	1286.4
2005/10/27	3085.1	-89.2 / 79.9	133079.7	0.95372	90.00	104.22	101.88	416.0	1284.5
2005/10/28	3091.8	-89.1 / 80.6	133079.0	0.95373	90.00	104.49	101.78	416.9	1284.1
2005/10/29	3086.7	-88.1 / 82.6	133078.1	0.95374	90.00	104.76	101.21	416.2	1283.9
2005/10/30	3077.8	-89.3 / 79.4	133080.8	0.95370	90.00	105.01	102.22	415.0	1281.6
2005/10/31	3073.8	-88.8 / 80.4	133080.7	0.95370	90.00	105.27	101.96	414.4	1281.1
2005/11/01	3071.0	-88.8 / 80.3	133081.1	0.95370	90.00	105.51	102.07	414.1	1280.6
2005/11/02	3069.5	-89.5 / 78.5	133081.8	0.95369	90.00	105.75	102.64	413.9	1280.0
2005/11/03	3070.5	-89.5 / 78.5	133081.7	0.95369	90.00	105.98	102.70	414.0	1280.2
2005/11/04	3073.0	-88.8 / 80.4	133080.8	0.95370	90.00	106.20	102.23	414.3	1281.0
2005/11/05	3076.4	-86.3 / 85.9	133075.4	0.95378	90.00	107.43	104.13	414.8	1281.4
2005/11/06	3079.4	-86.9 / 84.8	133076.9	0.95376	90.00	107.59	104.46	415.2	1279.5
2005/11/07	3063.2	-86.2 / 85.5	133077.1	0.95375	90.00	107.57	103.54	413.0	1279.1
2005/11/08	3067.9	-85.8 / 86.4	133075.4	0.95378	90.00	108.07	104.37	413.6	1278.7
2005/11/09	3082.0	-85.8 / 86.9	133073.7	0.95380	90.00	108.27	104.47	415.5	1279.4
2005/11/10	3070.2	-85.3 / 87.4	133073.4	0.95381	90.00	108.45	104.37	414.0	1279.4
2005/11/11	3049.0	-85.8 / 85.7	133077.9	0.95374	90.00	108.62	105.01	411.1	1276.5
2005/11/12	3061.8	-84.8 / 87.9	133072.9	0.95382	90.00	109.17	105.66	412.8	1279.5
2005/11/13	3093.2	-83.6 / 90.8	133063.6	0.95395	90.00	109.33	104.95	417.1	1285.3
2005/11/14	3063.8	-83.7 / 89.7	133068.3	0.95388	90.00	109.47	105.38	413.1	1282.5
2005/11/15	3096.1	-80.0 / 95.4	133045.6	0.95422	90.00	109.88	104.29	417.5	1296.6
2005/11/16	3087.4	-82.6 / 92.0	133059.9	0.95401	90.00	110.00	105.81	416.3	1288.6
2005/11/17	3084.7	-81.0 / 93.9	133052.6	0.95411	90.00	110.34	105.87	415.9	1293.6
2005/11/18	3111.3	-79.6 / 96.2	133040.7	0.95429	90.00	110.44	105.24	419.5	1301.6
2005/11/19	3110.9	-78.5 / 97.3	133034.5	0.95438	90.00	110.72	105.53	419.4	1306.3
2005/11/20	3121.0	-77.8 / 98.2	133028.7	0.95447	90.00	110.80	105.40	420.8	1310.9
2005/11/21	3141.4	-79.4 / 97.2	133033.9	0.95439	90.00	111.04	106.66	423.6	1309.3
2005/11/22	3157.2	-77.9 / 99.0	133022.5	0.95456	90.00	111.20	106.49	425.7	1317.5
2005/11/23	3146.1	-79.8 / 96.9	133034.8	0.95438	90.00	111.36	107.95	424.2	1311.7
2005/11/24	3171.3	-77.3 / 99.9	133015.8	0.95465	90.00	111.50	107.31	427.6	1324.9

Table 7: Capture Orbit characteristics. Capture performed with the main engine

The table presents also the initial true anomaly of the thrust arc TA_0 and the final true anomaly TA_f. For the last days of the launch window, it can be seen how the ignition of the main engine is delayed up to true anomalies of about –77 deg and how the asymmetry of the thrust arc is much bigger than at the beginning of the launch window.

The angular elements of the capture orbit will not change very much in the manoeuvre near the pericenter except for the argument of pericenter due to the asymmetry of the thrust. For launches between 26/10/2005 and 04/11/2005, which corresponds to the first launch program, the argument of the pericenter will experiment a small increase during the thrust. For launches from 05/11/2005 onwards, the argument of pericenter will experiment a reduction during the thrust that can be up to 6 deg. The argument of the pericenter of the capture orbit will vary by about 7 deg depending on the day of launch.

Figure 4-8 presents the required Delta-V for the Venus Orbit Insertion manoeuvre over the whole launch period. The increase at the end of the launch window corresponds not only to the variation of the arrival hyperbolic velocity, but also to the fact that the capture manoeuvre intends to reduce the argument of pericenter as much as possible by using an asymmetrical thrust arc that requires a longer burn time to reach the nominal apocenter altitude.

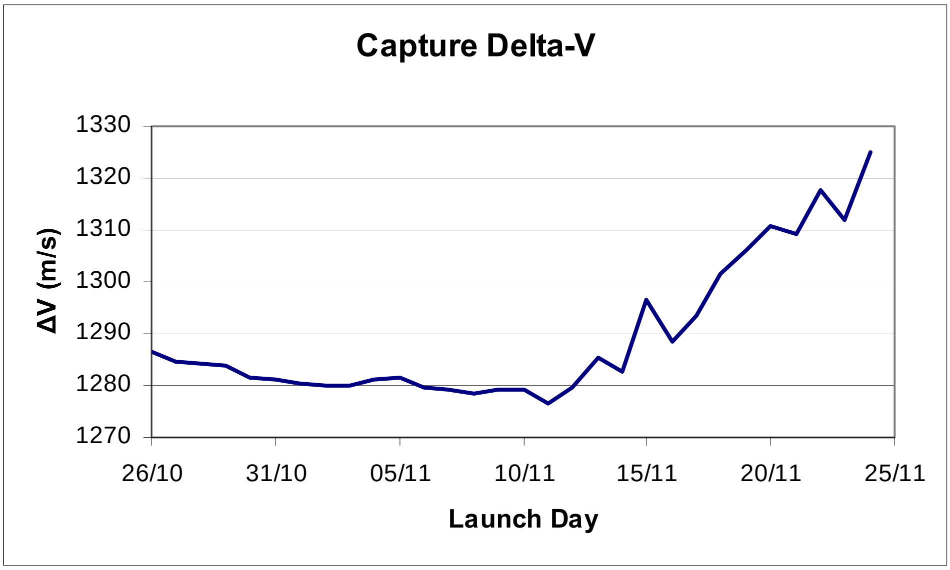

Figure 4-8: Finite thrust equivalent Delta-V used for Venus Orbit Insertion into the Capture Orbit

During main engine thrust, the spacecraft is close to Venus and in an attitude prescribed to maximise the efficiency of the manoeuvre. It is important, therefore, to know the eclipse conditions, and the so-called beta angle (angle between the orbital plane and the direction to the Sun). Figure 4-9 and Figure 4-10 present the variation of these parameters for launch days in the period 26/10/2005 to 24/11/2005.

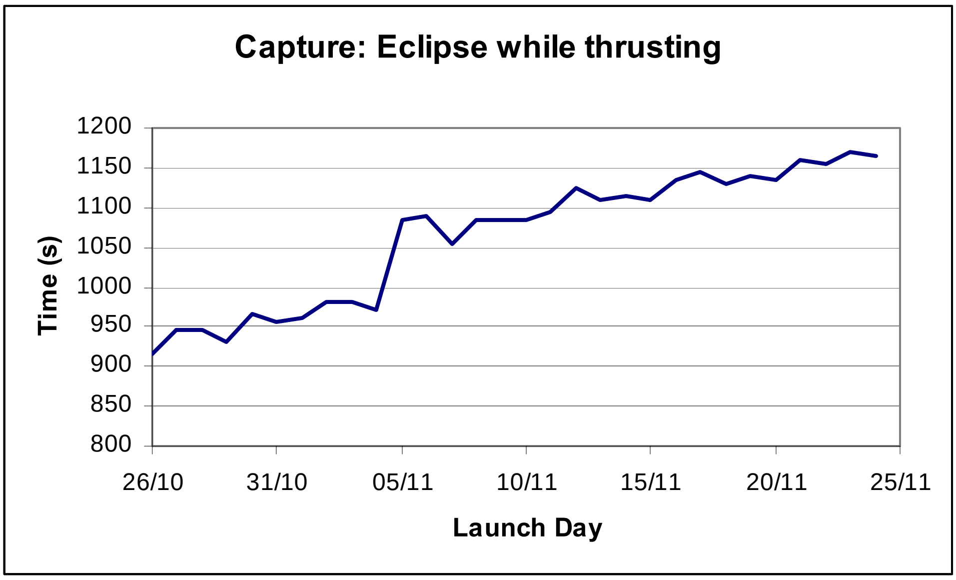

Figure 4-9: Eclipse condition during capture main engine thrust. Injection performed with the main engine

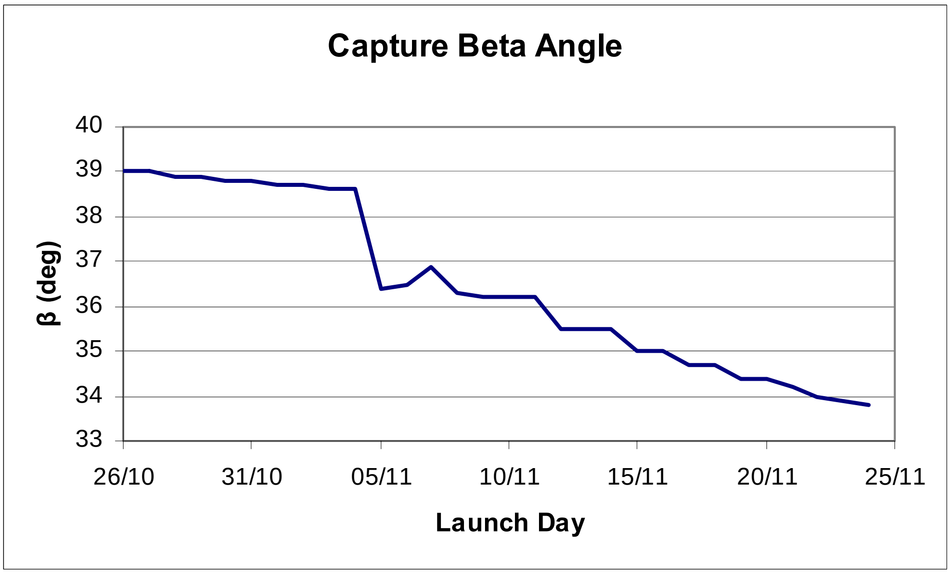

Figure 4-10: Beta angle variations during the Venus Orbit Insertion into the Capture Orbit.

During the main engine thrust there is also an occultation of the line of sight from the Earth to the spacecraft behind Venus. Communications are not possible during this period of time, which starts about the same time as the eclipse and has a smaller duration as shown in Figure 4-11.

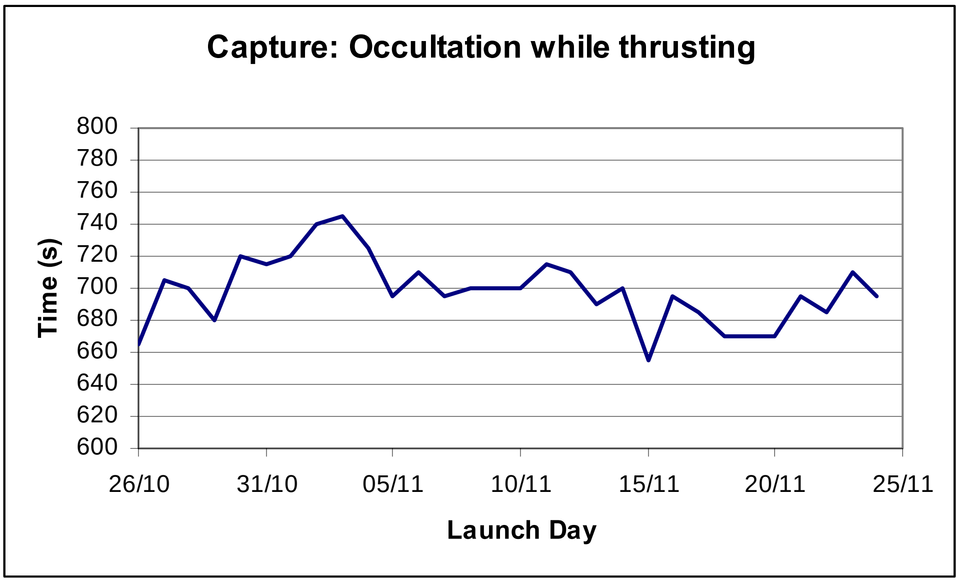

Figure 4-11: Occultation conditions during capture main engine thrust. Injection performed with the main engine.

4.2 *First Apocenter Lowering with the Main Engine*

After the spacecraft is injected into the capture orbit, the gravity pull of the Sun will increase the height of pericenter so that at the next pericenter pass the height is 250 km as desired. If it is needed, a small trim manoeuvre at apocenter of the capture orbit will be executed to adjust the pericenter height to the required value, Figure 4-12. This manoeuvre is less than 1.5 m/s.

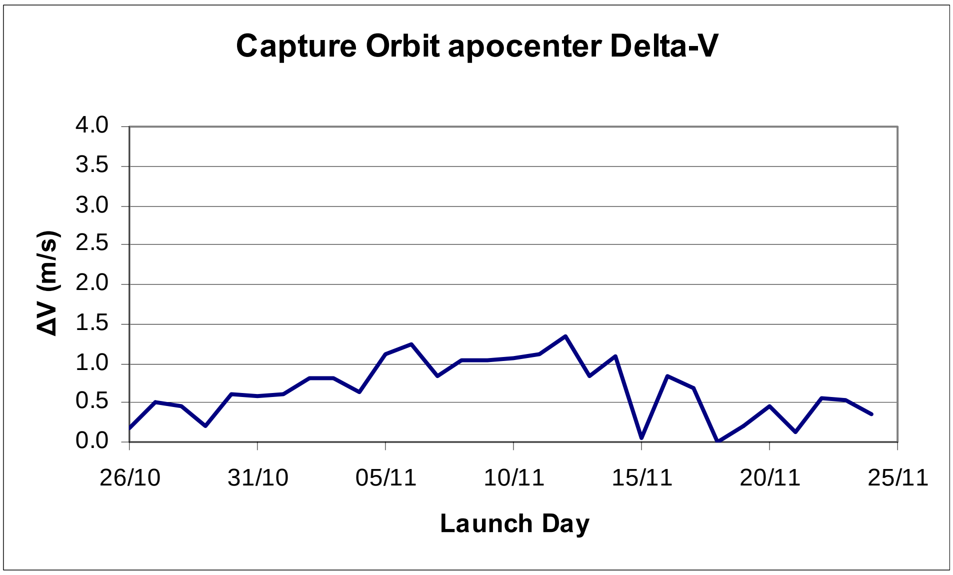

Figure 4-12: Trim manoeuvre at apocenter of capture orbit to correct height of pericenter

Table 8 contains the detail of the variation of the intermediate orbit obtained after the first apocenter lowering manoeuvre is performed with the main engine. The manoeuvre uses a propellant mass between 59 and 73 kg depending on the launch day, equivalent to 238-291 m/s.

At the end of the thrusting phase, the spacecraft enters into eclipse. The duration of the eclipse while the main engine thrust can be up to 500 seconds and its variation with the launch day is presented in Figure 4-13. The total duration of the eclipse can reach 1730 seconds and its variation is presented in Figure 4-14. The solar beta angle is presented in Figure 4-15.

Due to the asymmetry of the manoeuvre, the occultation of the spacecraft only occur while thrusting and for some launch days and can be up to 270 seconds (2005/11/24), Figure 4-16

Launch	Thrust. Time (s)	TA_0 / TA_f (deg)	Intermediate orbit					M prop (kg)	ΔV (m/s)
			a (Km)	e	i	AN	PE		
2005/10/26	496.2	-28.3 / 14.6	42065.2	0.85019	90.14	103.48	101.14	66.9	268.9
2005/10/27	516.4	-25.5 / 19.6	40963.9	0.84616	90.16	103.72	101.84	69.6	279.4
2005/10/28	531.9	-23.2 / 23.4	40188.4	0.84319	90.15	104.00	101.54	71.7	287.4
2005/10/29	524.1	-26.6 / 19.0	40593.3	0.84471	90.14	104.29	101.22	70.7	283.4
2005/10/30	519.1	-22.0 / 23.6	40837.4	0.84563	90.16	104.51	101.93	70.0	280.8
2005/10/31	514.1	-24.2 / 20.8	41088.1	0.84663	90.16	104.77	101.84	69.3	278.2
2005/11/01	511.6	-21.1 / 23.9	41213.5	0.84708	90.16	105.01	101.74	69.0	276.9
2005/11/02	511.3	-25.6 / 19.1	41227.2	0.84714	90.17	105.23	102.62	68.9	276.7
2005/11/03	512.4	-25.3 / 19.4	41174.1	0.84695	90.17	105.45	102.66	69.1	277.3
2005/11/04	513.1	-20.3 / 24.9	41143.5	0.84683	90.16	105.69	101.85	69.2	277.7
2005/11/05	516.5	-18.9 / 26.6	40976.1	0.84621	90.19	106.88	103.65	69.6	279.4
2005/11/06	532.8	-20.7 / 26.2	40156.9	0.84307	90.20	107.02	104.04	71.8	287.8
2005/11/07	504.5	-6.0 / 37.8	41899.6	0.84960	90.18	107.03	102.30	68.0	273.2
2005/11/08	515.5	-13.6 / 31.7	41124.3	0.84663	90.19	107.50	103.55	69.5	278.9
2005/11/09	538.4	-11.8 / 35.2	40047.1	0.84264	90.19	107.70	103.43	72.6	290.7
2005/11/10	516.0	-15.1 / 30.3	41057.7	0.84651	90.19	107.89	103.64	69.6	279.1
2005/11/11	490.8	2.0 / 43.4	42950.0	0.85318	90.21	108.03	103.40	66.2	266.1
2005/11/12	498.5	-10.2 / 33.4	42084.9	0.85014	90.22	108.57	104.70	67.2	270.1
2005/11/13	527.2	-10.3 / 35.7	40619.6	0.84486	90.20	108.76	103.87	71.1	284.9
2005/11/14	485.6	-7.7 / 34.7	42835.1	0.85281	90.21	108.88	104.33	65.5	263.4
2005/11/15	470.8	-3.7 / 37.1	43788.8	0.85609	90.18	109.34	103.08	63.5	255.7
2005/11/16	497.5	-5.1 / 37.9	42296.8	0.85088	90.22	109.41	104.57	67.1	269.6
2005/11/17	464.7	-2.0 / 38.1	44217.4	0.85748	90.22	109.74	104.61	62.7	252.5
2005/11/18	471.9	-0.5 / 39.9	43862.2	0.85633	90.20	109.87	103.87	63.6	256.2
2005/11/19	445.0	5.1 / 42.6	45700.4	0.86211	90.21	110.15	104.00	60.0	242.2
2005/11/20	437.8	5.7 / 42.6	46190.4	0.86357	90.20	110.23	103.88	59.0	238.5
2005/11/21	486.0	1.8 / 42.9	43191.0	0.85409	90.23	110.42	105.09	65.5	263.6
2005/11/22	469.1	11.9 / 49.6	44703.1	0.85903	90.23	110.59	104.50	63.2	254.8
2005/11/23	480.2	3.7 / 44.1	43595.0	0.85545	90.26	110.69	106.31	64.7	260.6
2005/11/24	455.3	4.5 / 42.8	45049.7	0.86012	90.24	110.86	105.76	61.4	247.6

Table 8: Orbit characteristics after first apocenter lowering with the main engine

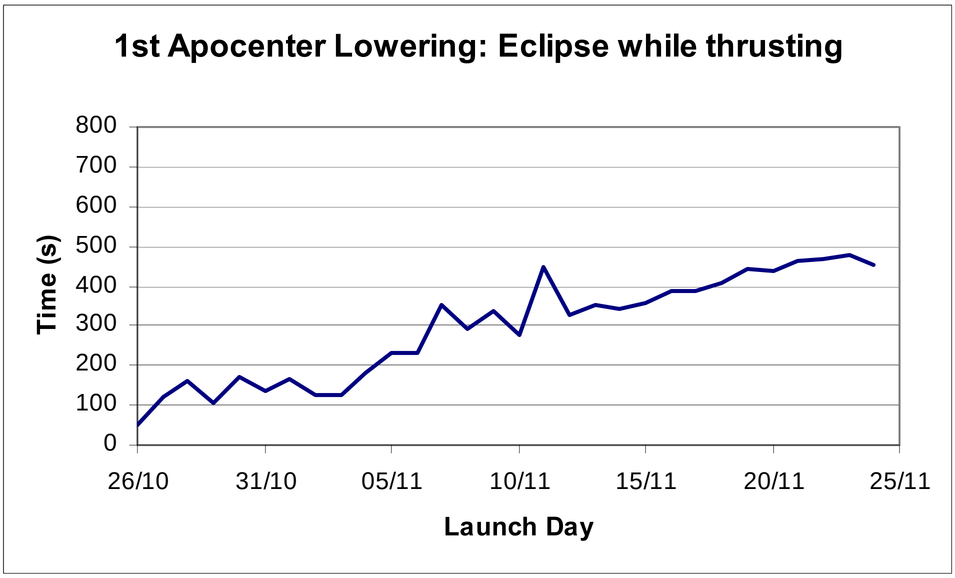

Figure 4-13: Eclipse condition during 1st apocenter lowering main engine thrust.

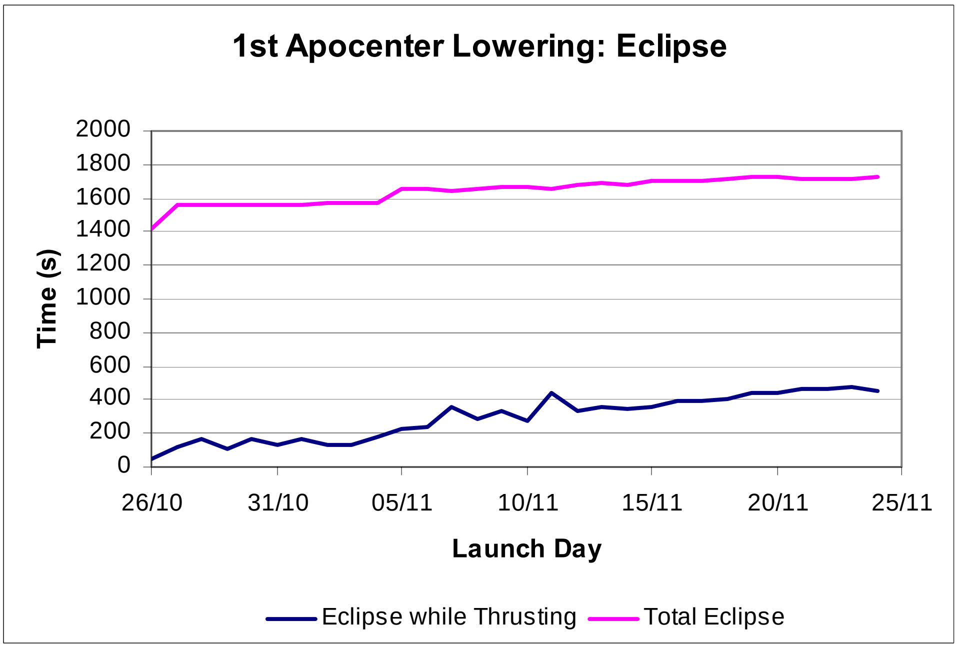

Figure 4-14: Total eclipse duration during 1st apocenter lowering main engine thrust.

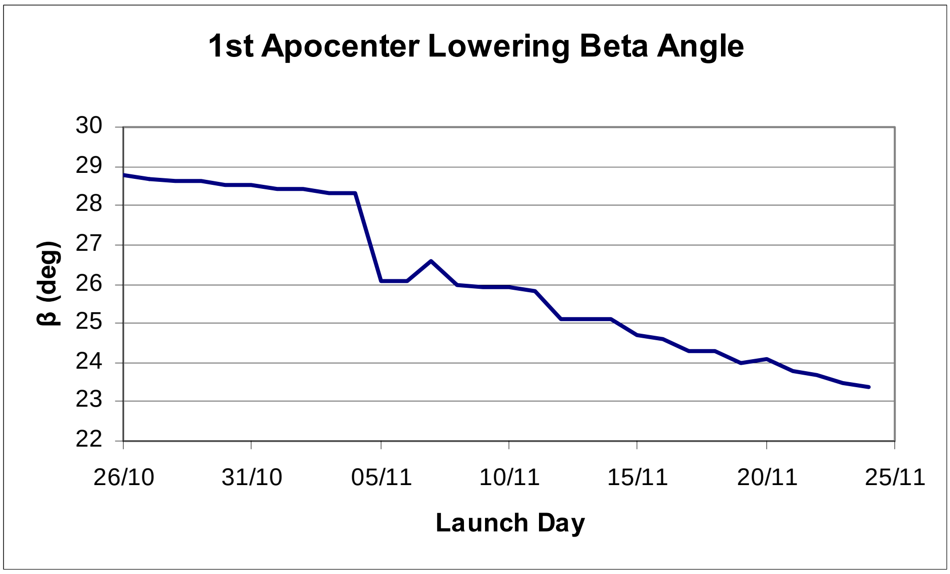

Figure 4-15: Beta angle during 1st apocenter lowering main engine thrust.

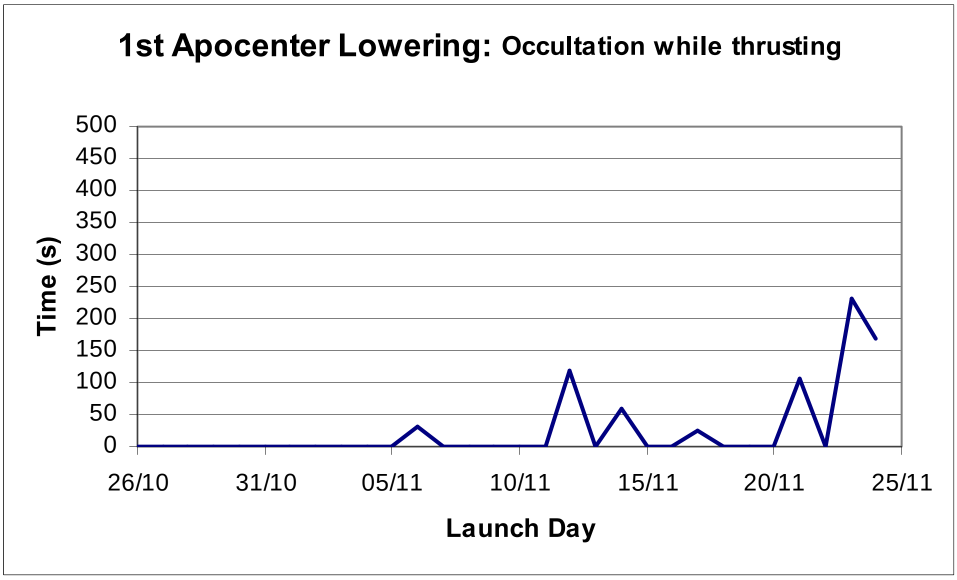

Figure 4-16: Occultation condition during 1st apocenter lowering main engine thrust.

After the first apocenter lowering, the angular orbital elements of the intermediate orbit are slightly different than the ones of the capture orbit. The perturbation of the Sun during the eccentric capture orbit produces an increase of the inclination of about 0.2 deg and a decrease of the right ascension of the ascending node of about 0.5 deg. On the argument of pericenter the effect of the asymmetrical manoeuvre with the main engine is acting as well. For launches from 05/11/2005, the argument of pericenter will experiment a further reduction of up to 2 deg. The argument of pericenter and consequently the latitude of the pericenter will depend on the launch day and it will vary by about 5 deg during the considered launch period 26/10/2005 to

24/11/2005. The argument of pericenter reached after the first apocenter lowering will not change during the rest of the apocenter lowering manoeuvres implemented with the thrusters, therefore it corresponds also to the initial argument of pericenter of the operational orbit. Figure 4-17 presents the variation over the launch period of the argument of pericenter for the three following orbits: the arrival hyperbola at Venus, the capture orbit and the initial operational orbit. From the 15 deg of variation over the launch window at arrival to Venus the capture and first apocenter lowering manoeuvres reduce this range of variation to only 5 deg.

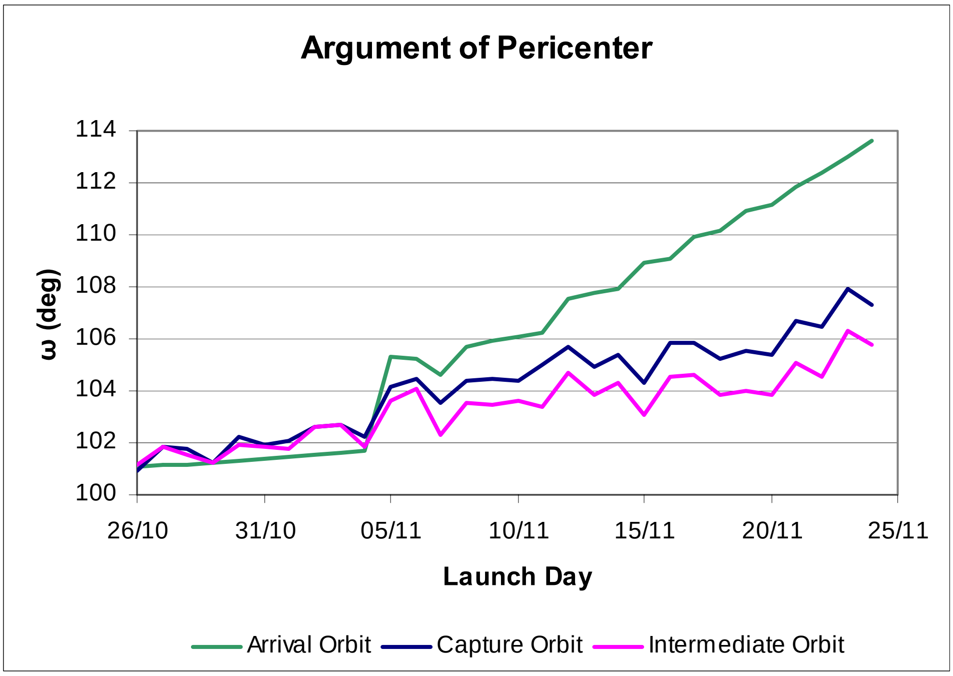

Figure 4-17: Changes experimented by the Argument of Pericenter from arrival up to the operational orbit over the launch period 26/10/2005 to 24/11/2005.

4.3 *Apocenter Lowering Manoeuvres with the thrusters*

Once the spacecraft has performed the capture and the first lowering manoeuvres with the main engine, it is inserted into an intermediate orbit of period from 1.02 days to 1.27 days. The variation of this period over the launch window is presented in Figure 4-18.

The rest of the apocentre lowering until reaching the of 1-day period operational orbit will be provided by the reaction control thrusters at the pericenter passes of the next orbits. It is supposed that the thrusters have to be used in modulated mode at 20 % in order to produce a thrust. The equivalent thrust level is then 11.2 N and the specific impulse 280 s. Being conservative, the direction of the thrust is assumed fixed in an inertial reference and directed against the hypothetical velocity at pericentre. Due to the lower efficiency of the thrusters in comparison with the main engine, the magnitude of the manoeuvres has to be limited in order to maintain the gravity losses below a given value. The limit considered is 20 m/s, which corresponds to manoeuvres extending from –50 deg to 50 deg in true anomaly and having gravity losses of about 8 %.

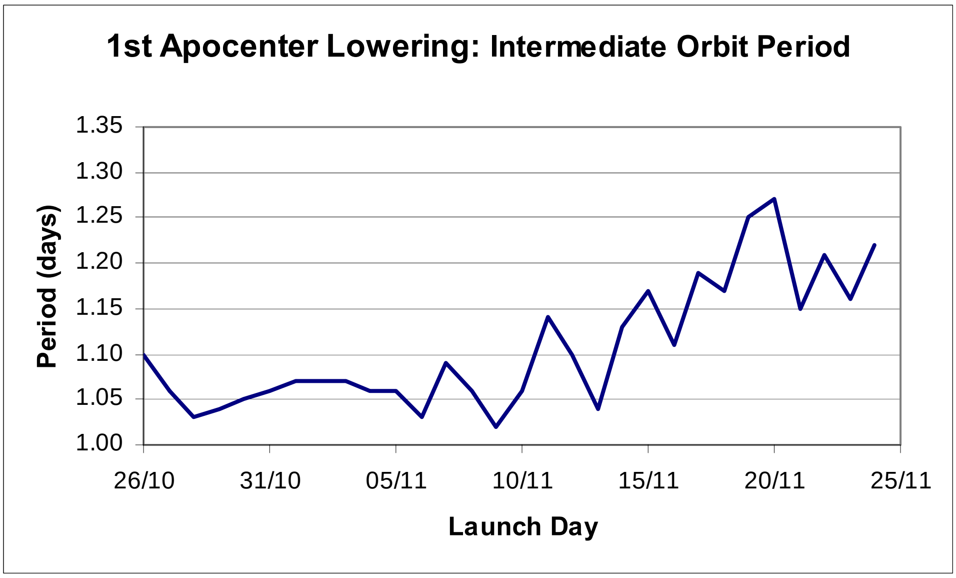

Figure 4-18: Period of the intermediate orbit after the first apocenter lowering with the main engine over the whole launch window

Depending on the period of the intermediate orbit, the number of apocenter lowering manoeuvres with the RCT necessary to reach the operational orbit vary between 1 and 4. The implementation of these manoeuvres allows more flexibility in the phasing of the spacecraft along the orbit, and in the correction of errors. Furthermore, the introduction of waiting orbits between the apocenter lowering manoeuvres will increase this flexibility without any loss on the performance. Table 9 provides an estimation of the delta-V of these manoeuvres and the minimum transfer time from the arrival at Venus up to the operational orbit. The transfer time considers at least 2 days between any of the manoeuvres to determine the orbit and prepare the correction. In the table, the delta-V of every manoeuvre (the label ΔVn corresponds to the n-th manoeuvre) and the duration of every thrust arc are also presented.

During the burns with the thrusters no occultation occurs and eclipse is entered about the half of the thrust arc with a maximum duration while thrusting of 660 s. The solar β angle decreases with the number of manoeuvre and at the end of the launch window, as presented in Figure 4-19.

Launch	#	ΔV1 (m/s)	Thrust Time 1 (sec)	ΔV2 (m/s)	Thrust Time 2 (sec)	ΔV3 (m/s)	Thrust Time 3 (sec)	ΔV4 (m/s)	Thrust Time 4 (sec)	Total ΔV (m/s)	Transfer Time (days)
2005/10/26	2	13.5	879.0	13.4	871.2	-	-	-	-	27.0	11.5
2005/10/27	1	16.2	1052.0	-	-	-	-	-	-	16.2	9.3
2005/10/28	1	7.6	493.6	-	-	-	-	-	-	7.6	9.2
2005/10/29	1	12.0	779.2	-	-	-	-	-	-	12.0	9.2
2005/10/30	1	14.7	956.8	-	-	-	-	-	-	14.7	9.3
2005/10/31	1	17.6	1146.2	-	-	-	-	-	-	17.6	9.3
2005/11/01	1	19.1	1242.7	-	-	-	-	-	-	19.1	9.3
2005/11/02	1	19.3	1253.6	-	-	-	-	-	-	19.3	9.3
2005/11/03	1	18.6	1212.3	-	-	-	-	-	-	18.6	9.3
2005/11/04	1	18.3	1188.7	-	-	-	-	-	-	18.3	9.3
2005/11/05	1	16.3	1060.9	-	-	-	-	-	-	16.3	9.3
2005/11/06	1	7.2	471.3	-	-	-	-	-	-	7.2	9.2
2005/11/07	2	12.7	826.1	12.5	811.6	-	-	-	-	25.2	11.4
2005/11/08	1	18.0	1171.3	-	-	-	-	-	-	18.0	9.3
2005/11/09	1	6.1	395.0	-	-	-	-	-	-	6.1	9.2
2005/11/10	1	17.3	1122.7	-	-	-	-	-	-	17.3	9.3
2005/11/11	2	17.7	1152.9	18.6	1203.9	-	-	-	-	36.4	11.6
2005/11/12	2	13.6	884.1	13.5	876.3	-	-	-	-	27.1	11.5
2005/11/13	1	12.3	798.5	-	-	-	-	-	-	12.3	9.2
2005/11/14	2	17.2	1118.0	17.9	1158.6	-	-	-	-	35.1	11.5
2005/11/15	3	14.4	936.8	14.4	931.9	14.5	933.2	-	-	43.3	13.8
2005/11/16	2	14.6	950.9	14.7	953.7	-	-	-	-	29.3	11.5
2005/11/17	3	15.7	1019.5	15.7	1013.7	16.1	1038.3	-	-	47.5	13.9
2005/11/18	3	14.6	951.0	14.6	946.0	14.8	951.1	-	-	44.0	13.8
2005/11/19	3	19.9	1293.2	19.9	1283.9	22.4	1435.3	-	-	62.2	14.1
2005/11/20	4	15.9	1035.8	15.9	1029.8	15.9	1023.8	16.6	1064.0	64.4	16.4
2005/11/21	2	18.9	1226.3	20.2	1301.3	-	-	-	-	39.0	11.6
2005/11/22	3	17.1	1111.2	17.1	1104.3	18.1	1162.8	-	-	52.3	14.0
2005/11/23	3	13.8	898.7	13.8	894.2	13.7	885.3	-	-	41.4	13.8
2005/11/24	3	18.1	1175.4	18.1	1167.7	19.5	1253.8	-	-	55.7	14.0

Table 9: Sequence of apocenter lowering manoeuvres with the reaction control thrusters to reach the operational orbit

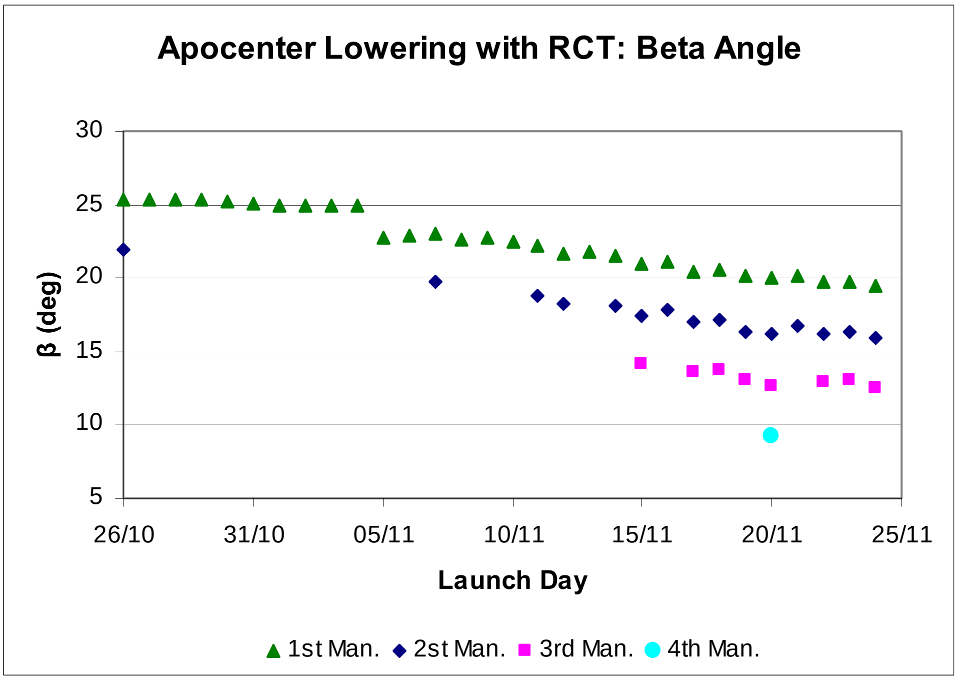

Figure 4-19: Beta angle during the apocenter lowering manoeuvres with the RCT

Figure 4-20 presents the evolution of the altitude of the spacecraft above Venus surface for the days following the Venus orbit insertion and up to the arrival to the operational orbit. The case presented corresponds to a launch on 26/10/2005. The manoeuvres performed at pericentre to reduce the apocentre and the manoeuvres performed at apocentre to correct the altitude of the next pericentre passage are marked in the plot. The actual number of these manoeuvres varies from one launch day to another.

Due to operational reasons, it has been decided to use for every launch day a capture orbit of 10-days of period. This implies no major changes in the strategy to capture and transfer the spacecraft to the operational orbit. However, it can affect the duration of the possible extension of the mission operations above the 1000 days nominal lifetime, see section 6.3.2.

Figure 4-22 presents the evolution of the altitude of the spacecraft for the case of launch on 24/11/2005 and using a capture orbit of a period of 10 days. In this case, the operational orbit is reached with one apocentre lowering manoeuvre with the ME and 3 manoeuvres with the RCT, one apocentre lowering more as in the previous case. Also for operational reasons, the apocentre lowering manoeuvre with the ME of about 270-310 m/s will be actually performed in two steps with the corresponding increment of the duration of the transfer to the operational orbit, but also with a small reduction of the total delta-V required to reach the operational orbit due to the lower gravity losses.

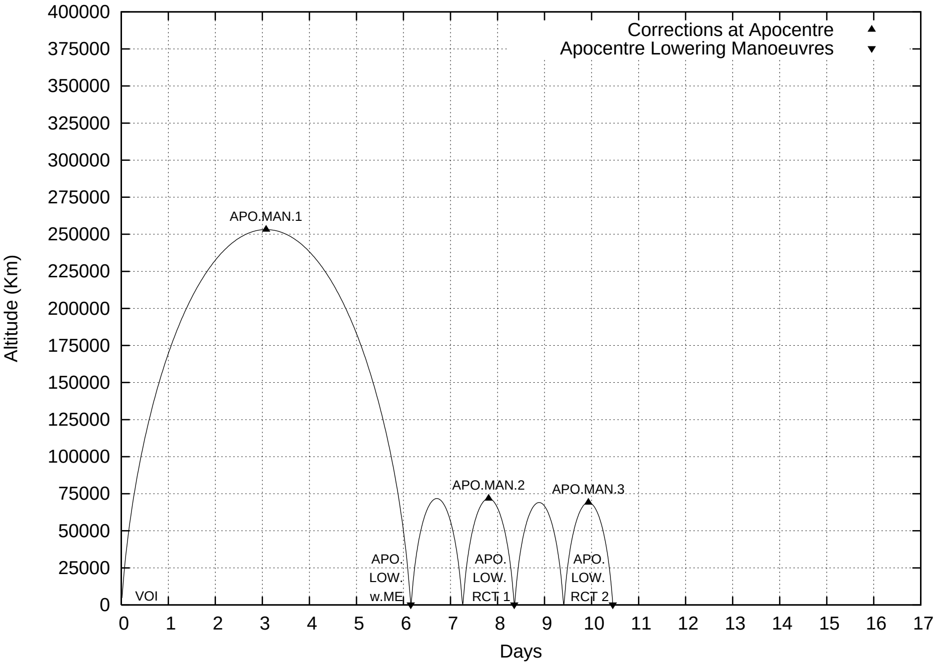

Figure 4-20: Altitude of the spacecraft during the capture and intermediate orbits (launch day 26/10/2005, capture orbit of 6.2 days)

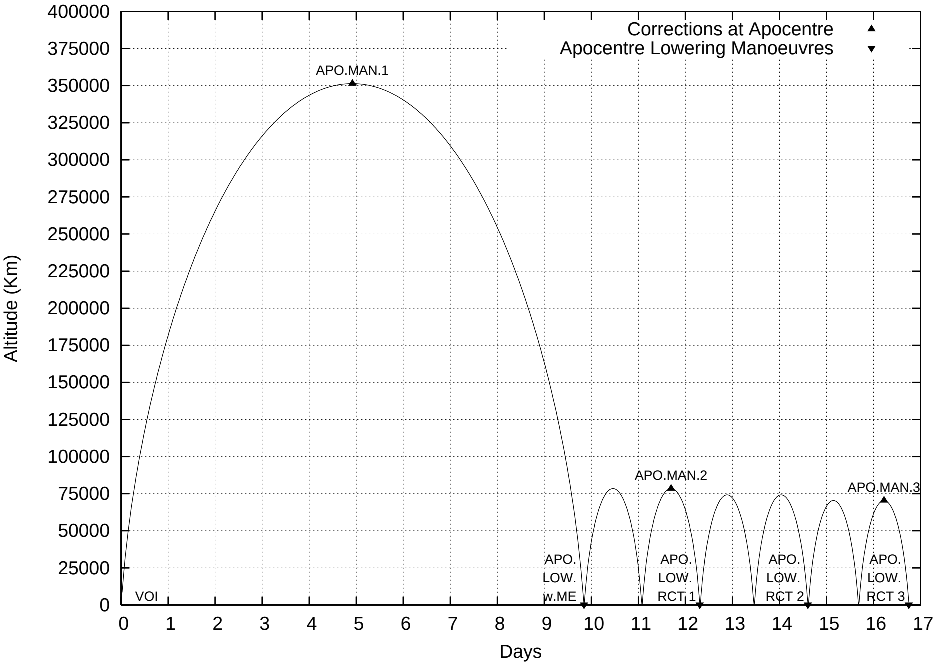

Figure 4-21: Altitude of the spacecraft during the capture and intermediate orbits (launch day 24/11/2005, capture orbit of 10 days)

Table 10 presents the keplerian orbital elements and the spacecraft mass at the arrival to the operational orbit after the capture and apocentre lowering manoeuvres.

Launch	Operational orbit						S/C M
	a (Km)	e	i	AN	PE	TA	(kg)
2005/10/26	39475.9	0.84024	90.16	103.46	101.09	0.00	723.8
2005/10/27	39475.9	0.84023	90.17	103.71	101.81	0.00	726.7
2005/10/28	39476.0	0.84024	90.16	103.99	101.51	0.00	729.0
2005/10/29	39475.9	0.84024	90.15	104.28	101.19	0.00	727.8
2005/10/30	39475.9	0.84023	90.17	104.49	101.90	0.00	727.0
2005/10/31	39475.9	0.84023	90.17	104.76	101.80	0.00	726.3
2005/11/01	39475.9	0.84023	90.17	105.00	101.71	0.00	725.9
2005/11/02	39475.9	0.84022	90.18	105.21	102.59	0.00	725.9
2005/11/03	39475.9	0.84022	90.18	105.44	102.63	0.00	726.0
2005/11/04	39475.9	0.84023	90.17	105.68	101.82	0.00	726.1
2005/11/05	39475.9	0.84021	90.20	106.86	103.62	0.00	726.6
2005/11/06	39476.0	0.84022	90.21	107.00	104.01	0.00	729.0
2005/11/07	39475.9	0.84023	90.19	107.01	102.25	0.00	724.2
2005/11/08	39475.9	0.84021	90.20	107.49	103.52	0.00	726.1
2005/11/09	39476.0	0.84022	90.20	107.69	103.40	0.00	729.4
2005/11/10	39475.9	0.84021	90.20	107.87	103.61	0.00	726.4
2005/11/11	39475.9	0.84021	90.22	108.01	103.34	0.00	721.2
2005/11/12	39475.9	0.84020	90.23	108.55	104.65	0.00	723.6
2005/11/13	39475.9	0.84022	90.21	108.74	103.83	0.00	727.7
2005/11/14	39475.9	0.84020	90.23	108.85	104.27	0.00	721.5
2005/11/15	39475.9	0.84021	90.21	109.31	103.00	0.00	719.3
2005/11/16	39475.9	0.84020	90.23	109.38	104.51	0.00	723.0
2005/11/17	39475.9	0.84019	90.24	109.71	104.53	0.00	718.2
2005/11/18	39475.9	0.84020	90.22	109.83	103.78	0.00	719.1
2005/11/19	39475.9	0.84019	90.23	110.11	103.92	0.00	714.3
2005/11/20	39475.9	0.84020	90.23	110.19	103.77	0.00	713.6
2005/11/21	39475.9	0.84019	90.25	110.39	105.03	0.00	720.5
2005/11/22	39475.9	0.84019	90.25	110.56	104.42	0.00	716.9
2005/11/23	39475.9	0.84018	90.28	110.66	106.23	0.00	719.7
2005/11/24	39475.9	0.84018	90.27	110.82	105.68	0.00	716.0

Table 10: Operational Orbit characteristics

4.4 *Use of 10 N thrusters after failure of Main Engine in VOI*

If the Main Engine fails during the execution of the Venus Insertion Manoeuvre, it may be possible to proceed with the manoeuvre by using the 10 N thrusters. The thrust level will be reduced from 414 N to at most 80 N if it is possible to use in parallel 8 thrusters of 10 N each. Depending on the time at which the ME fails, it will be possible to either totally complete the VOI, or to inject into an orbit with a higher apocenter. In the nominal case of fully using the Main Engine, the duration of the Venus Orbit Insertion Manoeuvre is about 3180 s, and the propellant used 420 kg, section 4.1. The following analysis supposes that after a given time "Time of failure", the ME fails and the 8 x 10 N thrusters start working. This time of failure is taken from 1900 s after start of VOI manoeuvre to the end of the manoeuvre. The available propellant is all available propellant except the last 40 m/s that is reserved for orbit maintenance. The nominal value of the specific impulse of the 10 N thrusters used. In a later analysis, it should be reduced to account for inefficiencies of the 10 N thrusters when used for a long period.

The parametric analysis is done by targeting after the failure orbits with apocenter height of 233000 km, 300000 km, 400000 km, 500000 km, and 600000 km.

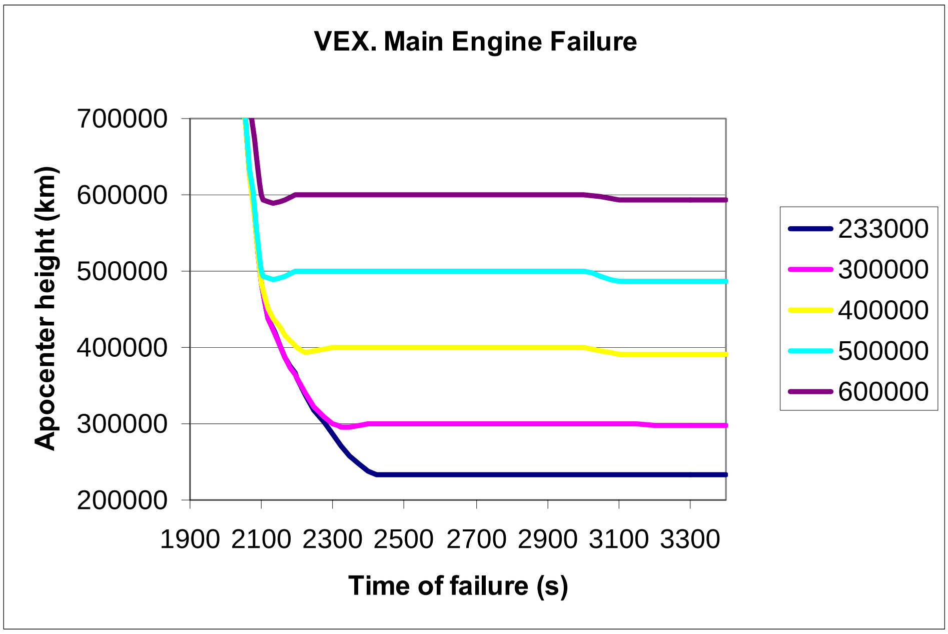

Figure 4-22: Apocenter of orbit after failure of Main Engine

The results show that if the ME failure occurs before 2100 seconds, even using all the available propellant, Figure 4-23, the final apocenter height will be greater than 700000 km, Figure 4-22. The spacecraft is left in an orbit extending almost outside of the sphere of influence of Venus. The thrusting time will be greater than 8000 s, Figure 4-24

If the ME failure takes place at about 2500 s, and the 10 N thrusters are used, it is still possible to inject into an orbit with apocenter height of 233000 km. In this case, all the propellant will be used. The mission will be severely degraded since the operational orbit will have a period of almost 5 days, and the pericenter height will increase very quickly.

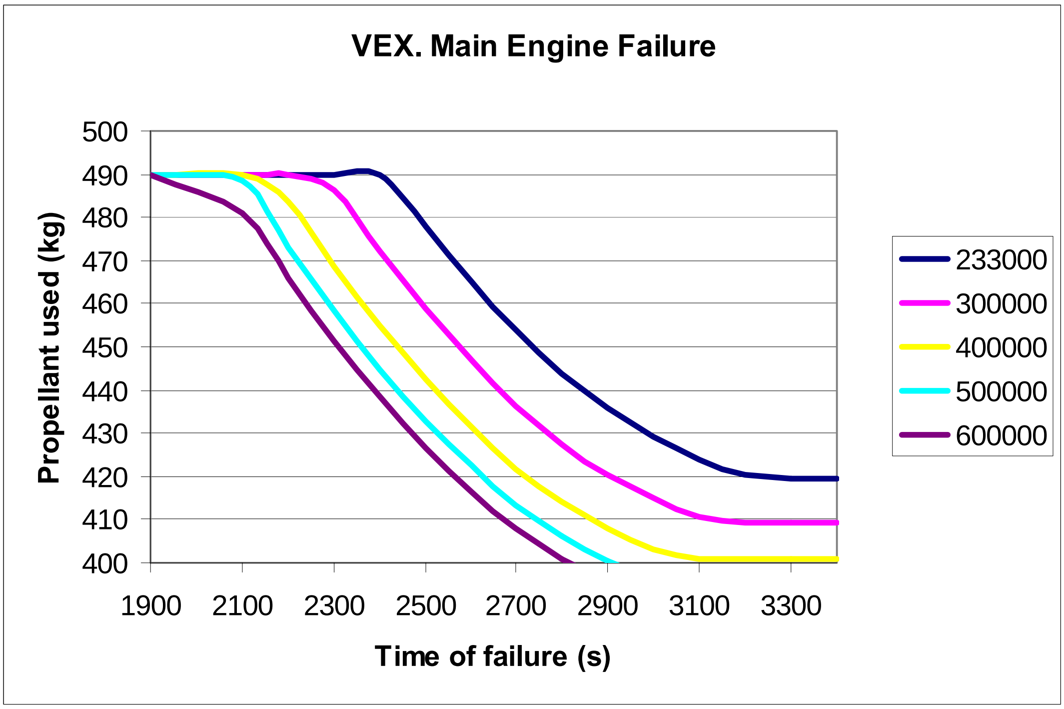

Figure 4-23: Propellant used after failure of Main Engine

If the ME failure takes place after 2900 s, it is possible to achieve an orbit with an apocenter height closer to the nominal.

A possible strategy is as follows:

- If the Main Engine failure takes place before 2100 + T1 seconds of thrusting, do not use the 10 N thrusters. The mission is severely degraded and a mission recovery shall be done.
- If the ME fails between 2100 + T1 s and 2900 + T2 seconds of thrusting, reach a predefined apocenter height and stop thrusting. After one revolution, the 10 N thrusters can be used much more efficiently to achieve an orbit that will partially satisfy some scientific requirements.
- If the ME fails after 2900 + T2 seconds of thrusting, do not use the 10 N thrusters. After one revolution, the 10 N thrusters can be used much more efficiently to achieve an orbit that will be closer to the nominal operational orbit.

The actual values of T1 and T2 can be fixed during the interplanetary phase, when the available propellant for VOI is well determined.

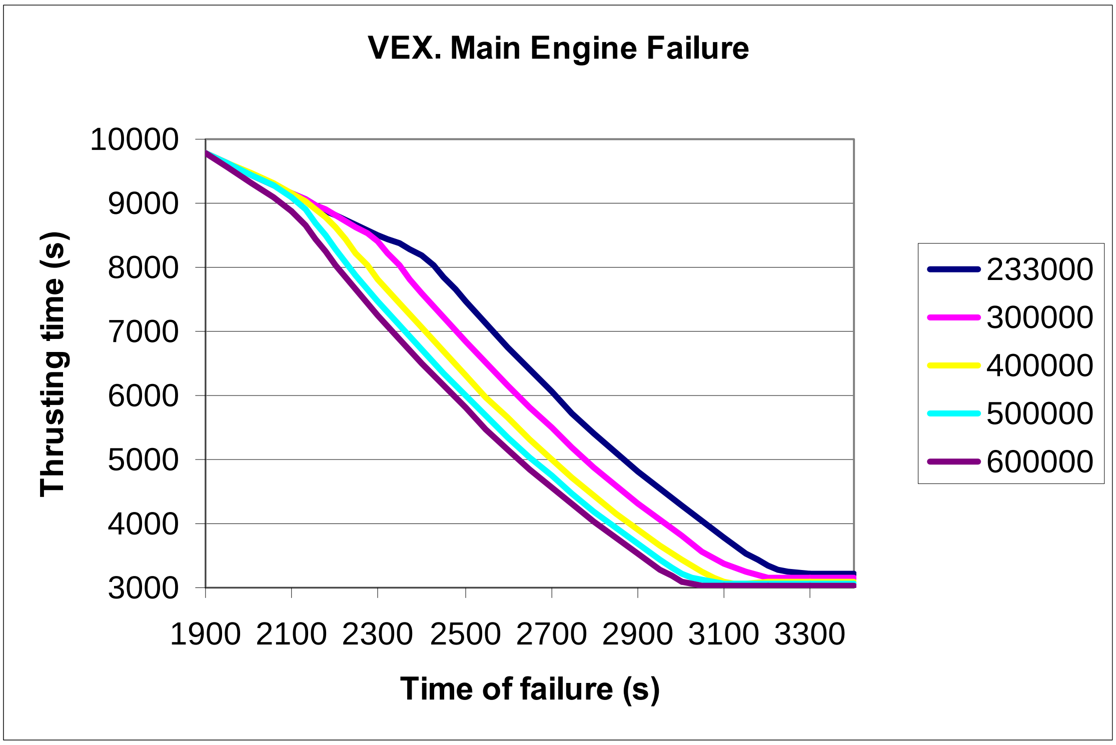

Figure 4-24: Time of thrust after failure of Main Engine

5. Interplanetary Trajectory

5.1 Earth-Venus Transfer

The main characteristics of the interplanetary trajectory do not depend very much on the particular day of launch. The spacecraft performs more than a half revolution around the Sun before reaching Venus. The trajectory presented in Figure 5-1 is for a launch on 05/11/2005.

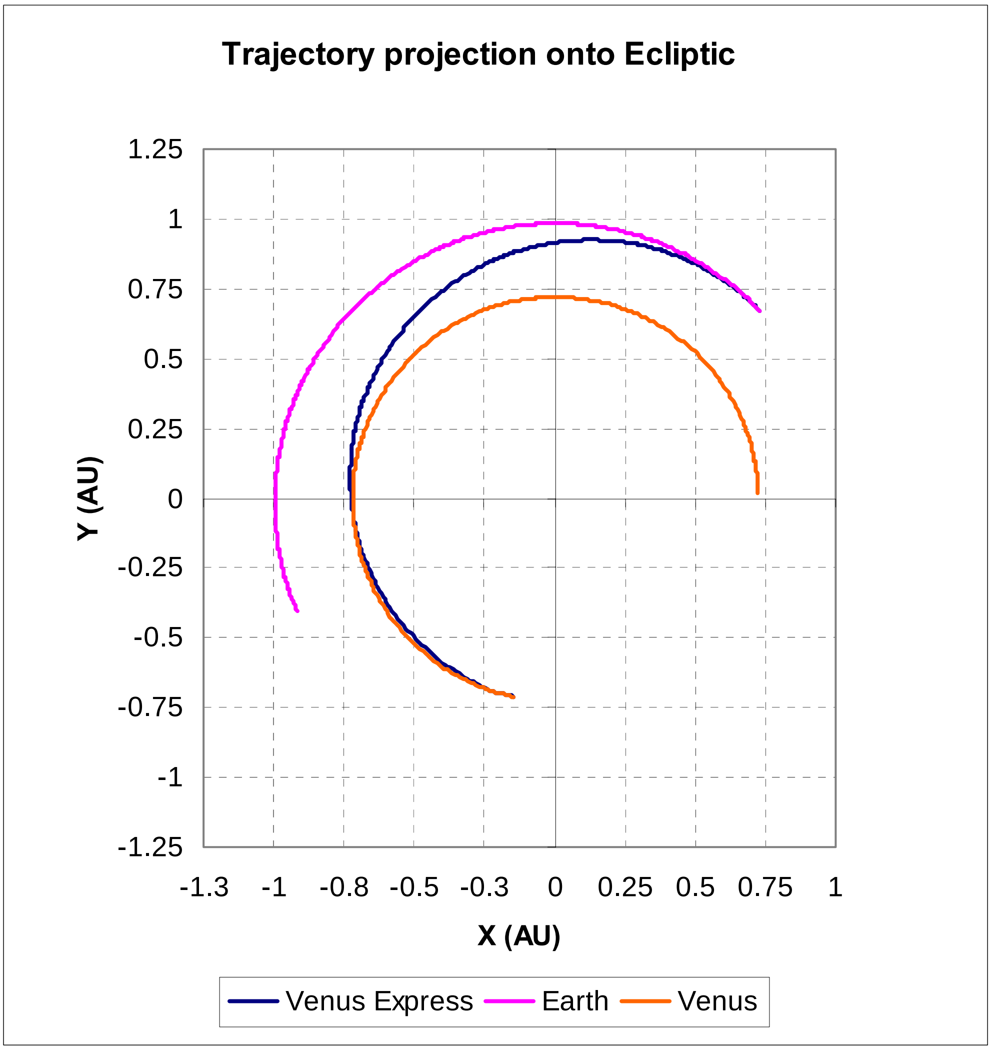

Figure 5-1: Orbits of Venus Express, Earth and Venus projected onto the ecliptic.

The evolutions of some geometrical parameters are presented in Figure 5-3 to Figure 5-5.

Figure 5-3 shows the distance of the spacecraft to the Earth, to Venus, and to the Sun. Figure 5-5 shows the distance rate in km/s of the spacecraft with respect to the Earth and to Venus. Finally, Figure 5-4 shows the angles Earth-spacecraft-Sun, and spacecraft-Earth-Sun. When this angle has a small value, 2º - 3º, the spacecraft will be in conjunction with the Sun and radio-communication with the spacecraft will be difficult.

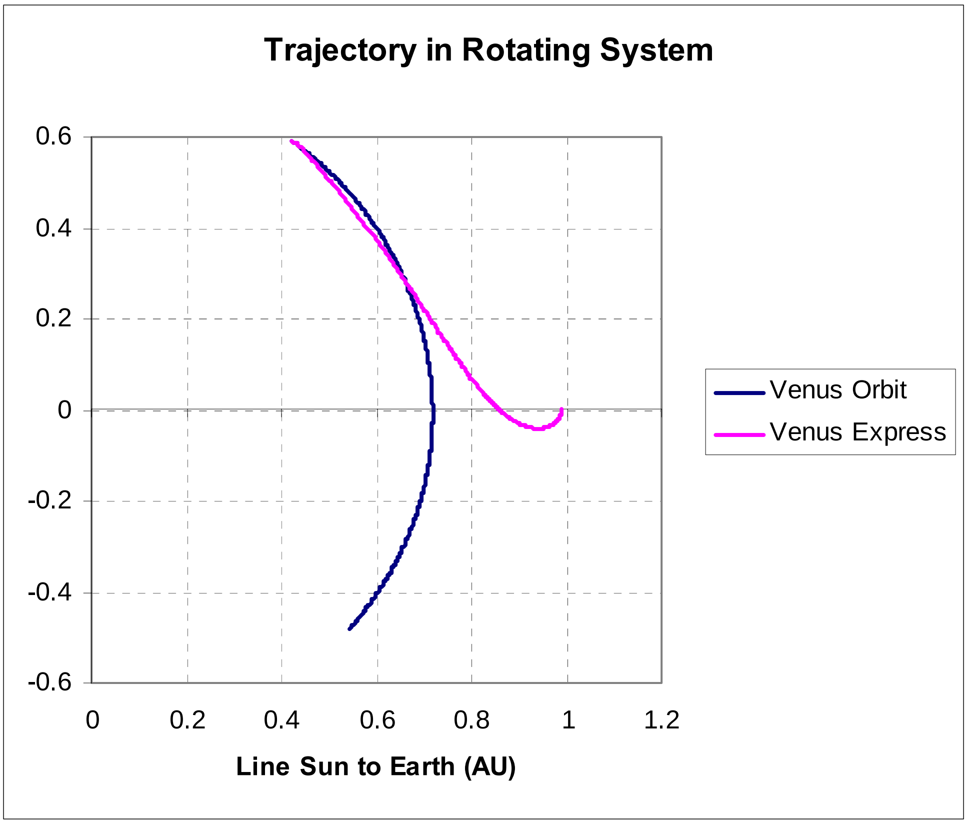

Figure 5-2: Orbit of Venus Express projected onto the ecliptic in a rotating system with axis fixed from Sun to Earth (Sun is at point (0,0), Earth is at (1,0))

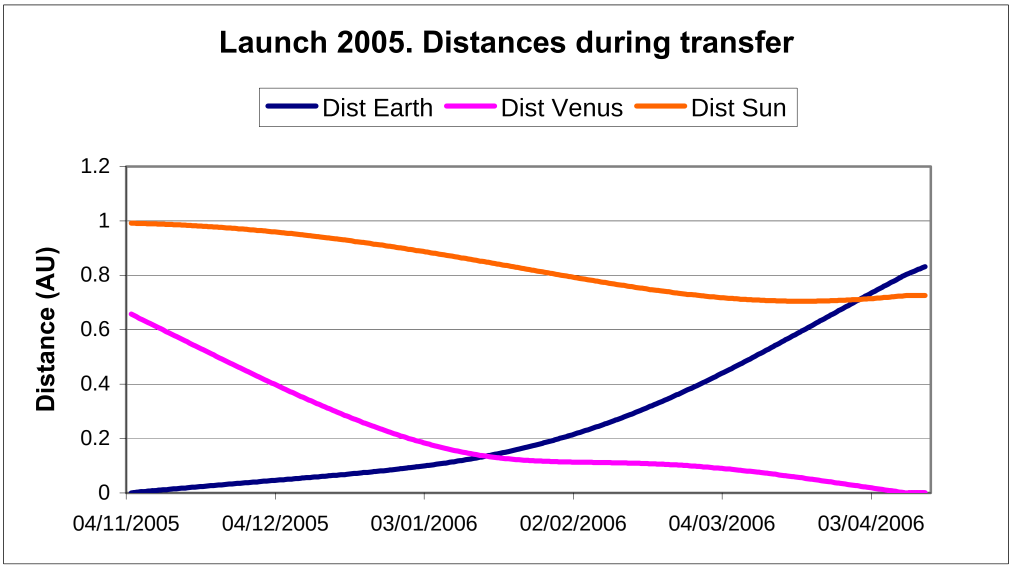

Figure 5-3: Distance of Venus Express to Sun, Earth, and Venus. Launch 2005

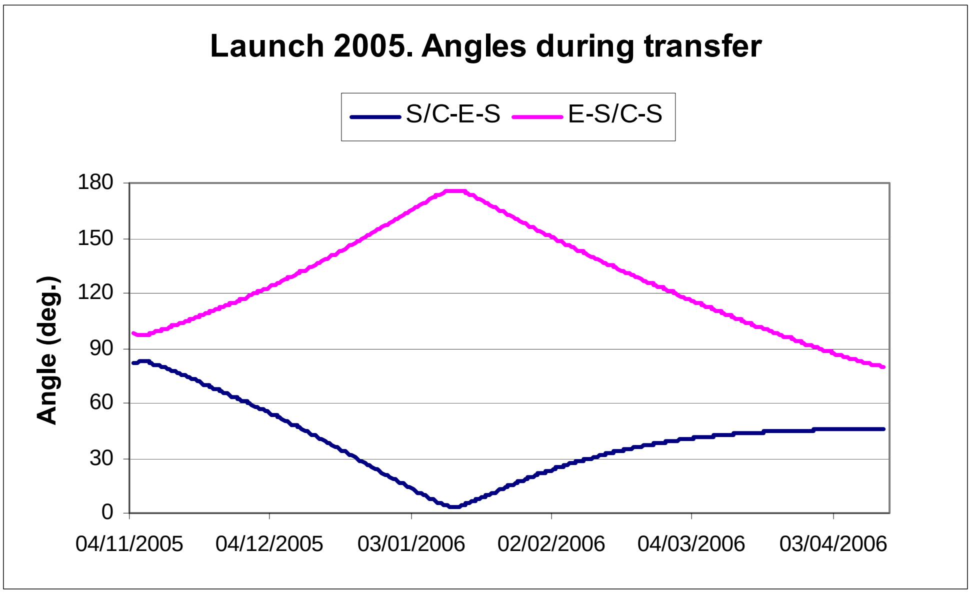

Figure 5-4: Communication angles during interplanetary trajectory. Launch 2005.

The radio signal will be affected by the Doppler shift due to the relative velocity in the direction of the line of sight from the Earth to the spacecraft. Figure 5-5 shows this relative velocity is in the range of 2.6 km/s to 17.4 km/s. To obtain the Doppler effect the rotational velocity of the Earth has to be added.

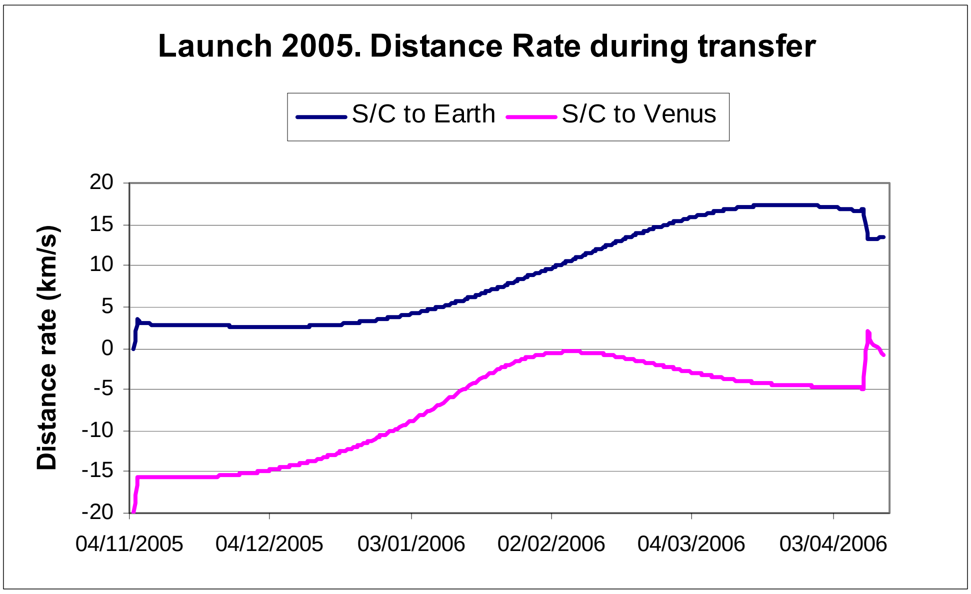

Figure 5-5: Rate of change of the distance Venus Express-to-Earth during the interplanetary phase. Launch 2005

5.2 *Ground Station Coverage during transfer arc*

This section presents the results of a study on the coverage from the ground stations during the interplanetary transfer from the Earth to Venus. The primary ground station is Cebreros (Spain), while the station in New Norcia (Perth, Australia) is considered as back up.

Figure 5-6 and Figure 5-7 present the entry (AOS) and exit (LOS) of the visibility interval during each day for each ground station. The vertical axis is the UTC hour and the visibility from the ground station is obtained during the shadowed zone. The continuous lines show the AOS/LOS for a minimum elevation angle of 5 degrees, for which it is most likely to establish communication with the spacecraft. A minimum elevation of 10 degrees is normally required for commanding and the AOS/LOS for this elevation are shown by means of the dashed lines.

Figure 5-8 presents the duration of the visibility from both ground stations versus the calendar date. The low declination of the spacecraft during the trajectory allows it to be observed during more than 12 hours per day from New Norcia. Visibility duration from Cebreros is lower and ranges from 7 hours to 10 hours at the arrival at Venus.

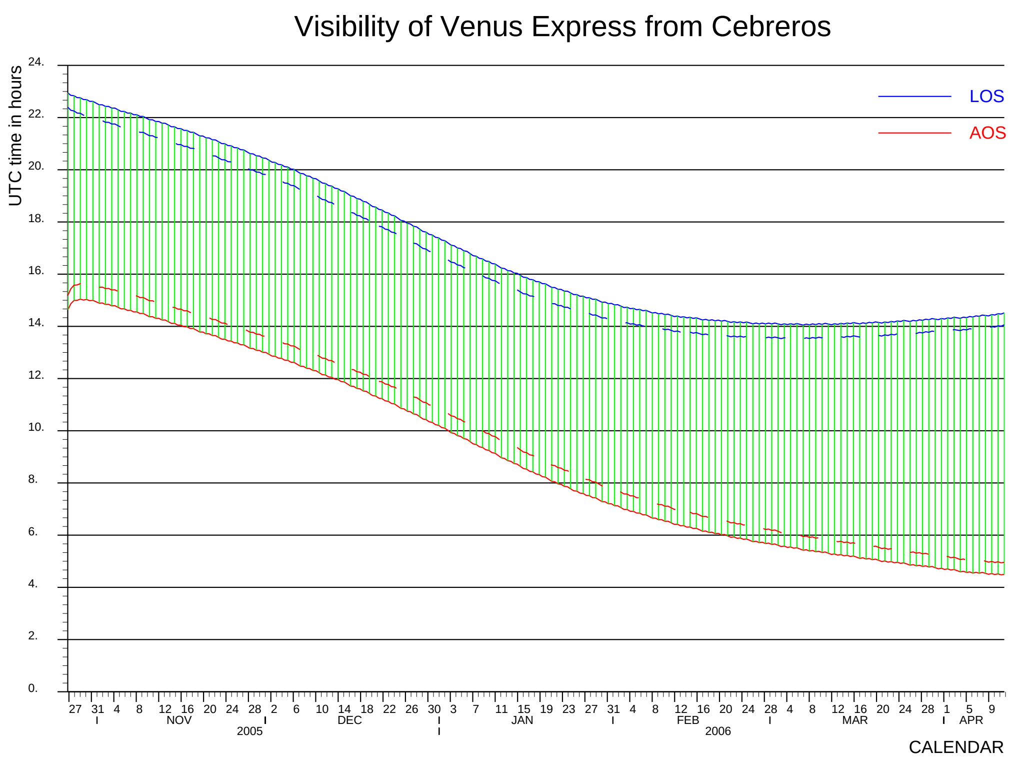

Figure 5-6: Visibility from Cebreros during the transfer arc Earth-Venus

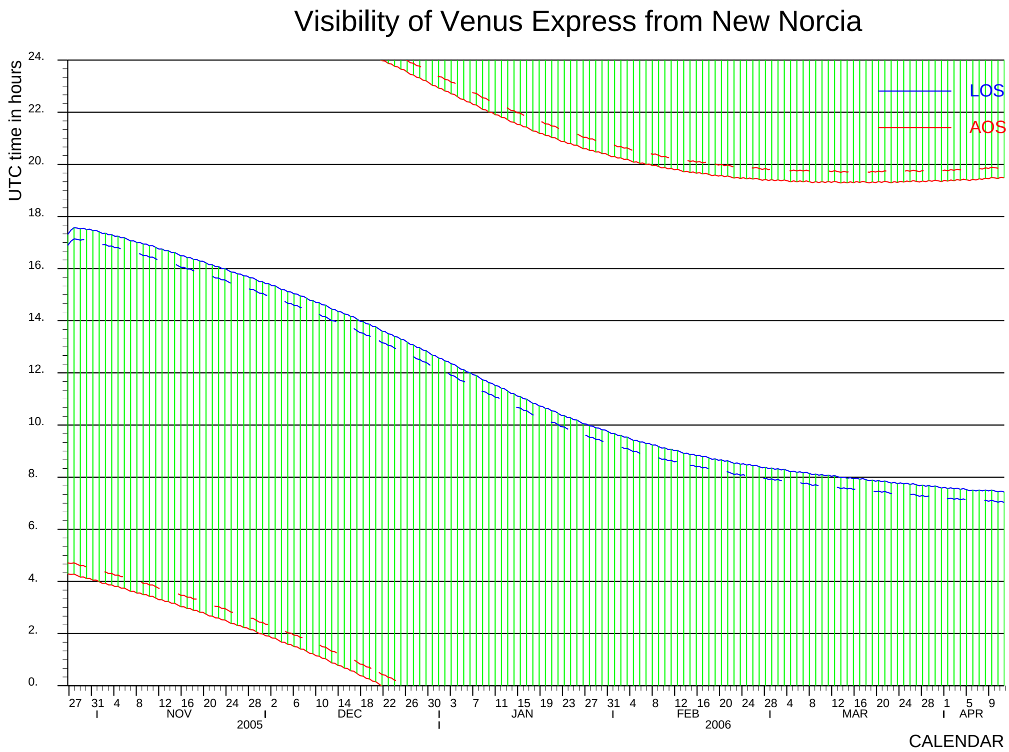

Figure 5-7: Visibility from New Norcia during the transfer arc Earth-Venus

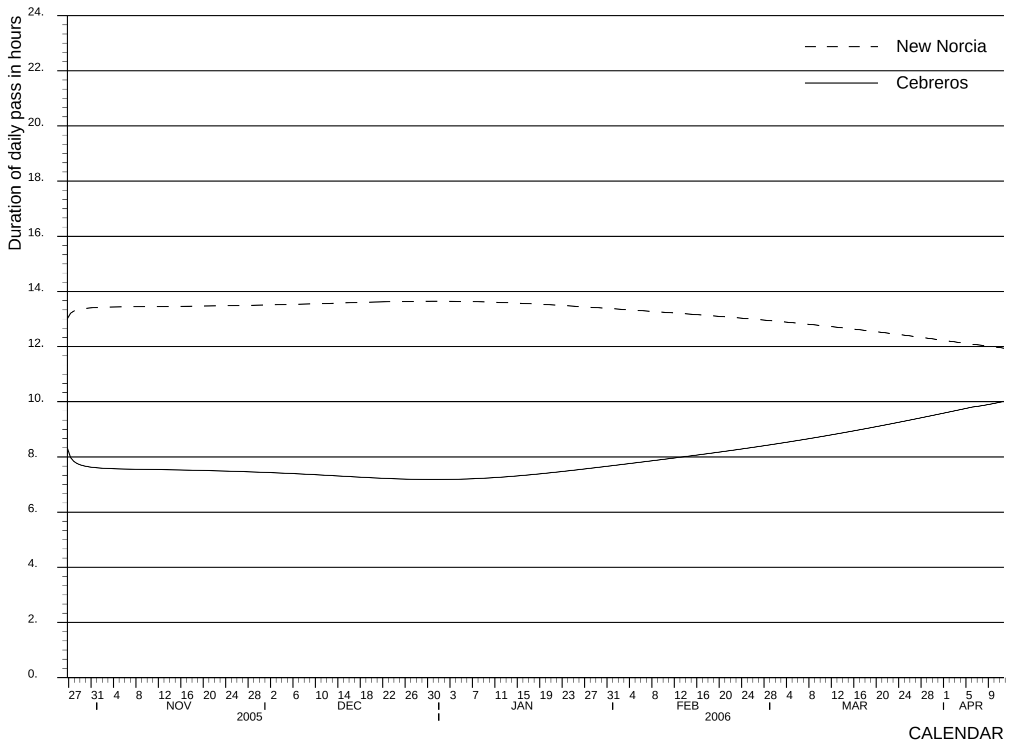

Figure 5-8: Duration of the visibility during the transfer arc Earth-Venus

6. Evolution and Control of Operational Orbit

6.1 Earth-Venus Geometry during Operational Orbit

An extended mission lifetime of 1000 days corresponds to about four and a half revolutions of Venus around the Sun (period of Venus is 225 days). During this time, the configuration of the Earth-Venus line performs almost two cycles, as corresponds to the synodical period of the Earth-Venus relative movement of 586 days. Figure 6-1presents the relative positions of the Earth and Venus for the first four Venus years.

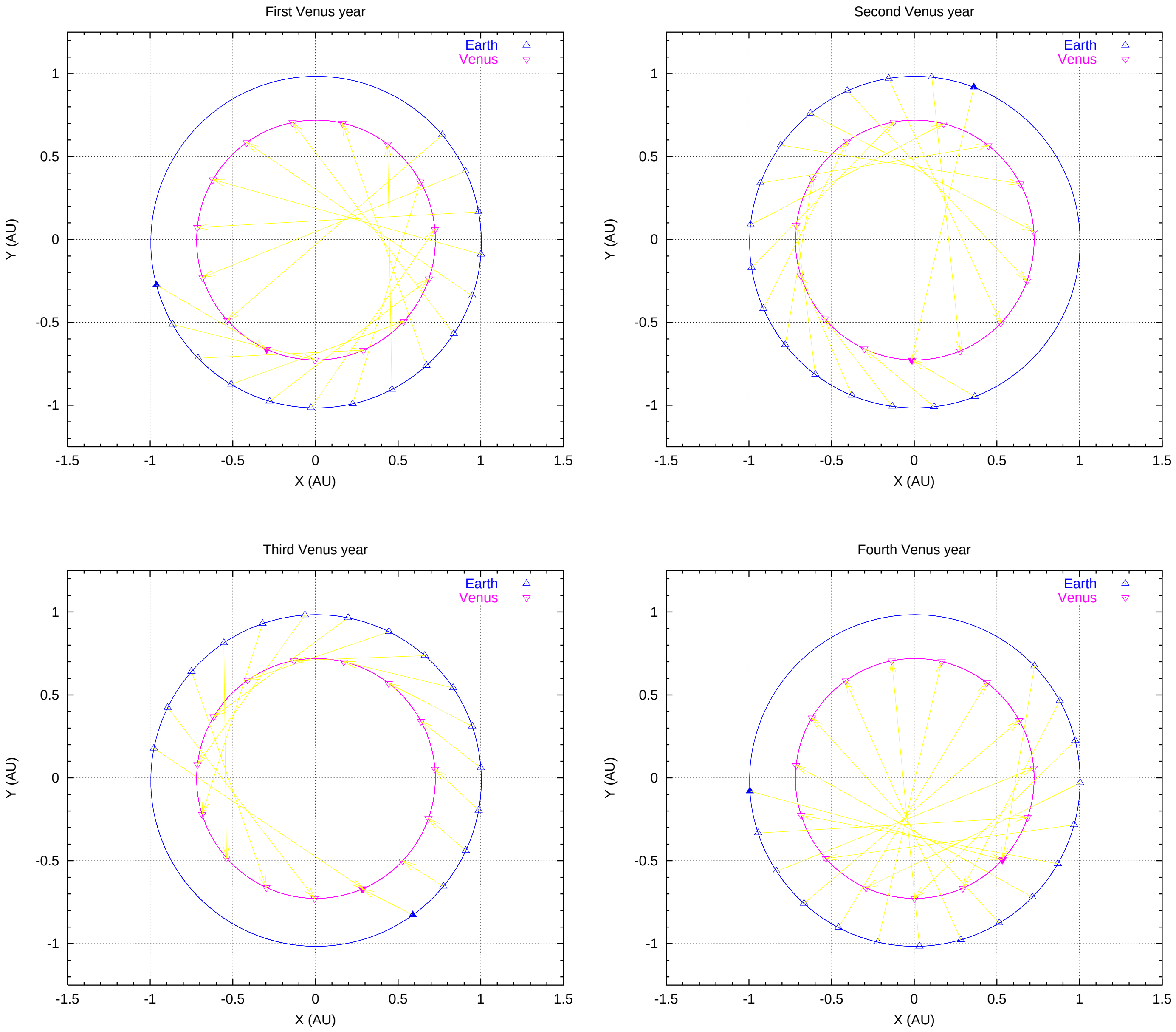

Figure 6-1: Geometric configuration of Earth and Venus on the ecliptic plane. Tick marks every 15 days

Figure 6-2 presents the evolution of the distance from the Earth to Venus. During the extended mission lifetime there is an inferior conjunctions at 19/08/2007 and two superior conjunctions, the first at 28/10/2006 and the second at 08/06/2008. The Venus-Earth-Sun angle is almost zero degrees for these superior conjunctions preventing communications with the spacecraft for a few days. The evolution of the geometric angles is presented in Figure 6-3.

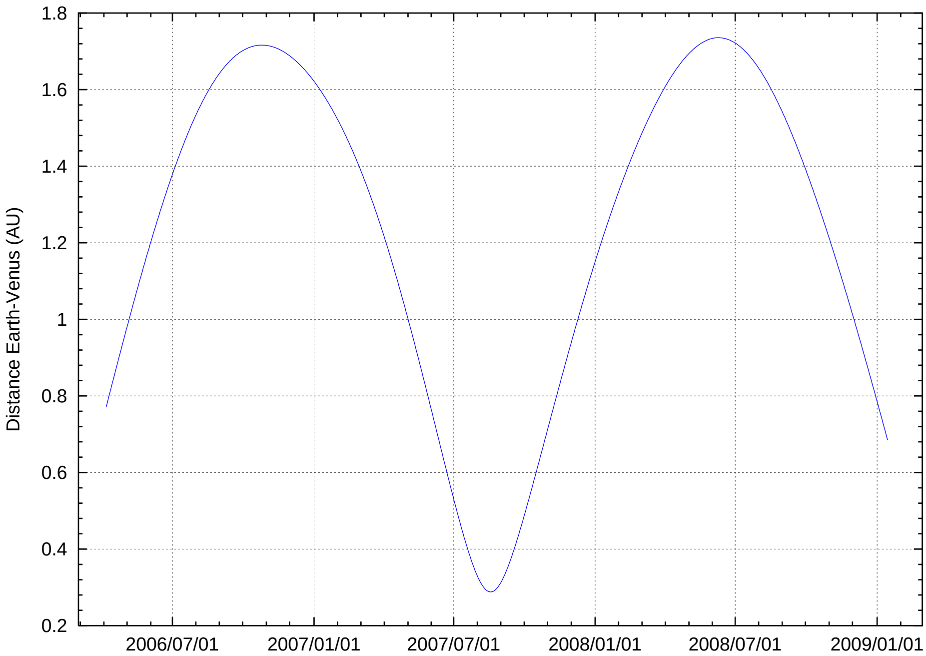

Figure 6-2: Earth-Venus Distance during the extended mission lifetime

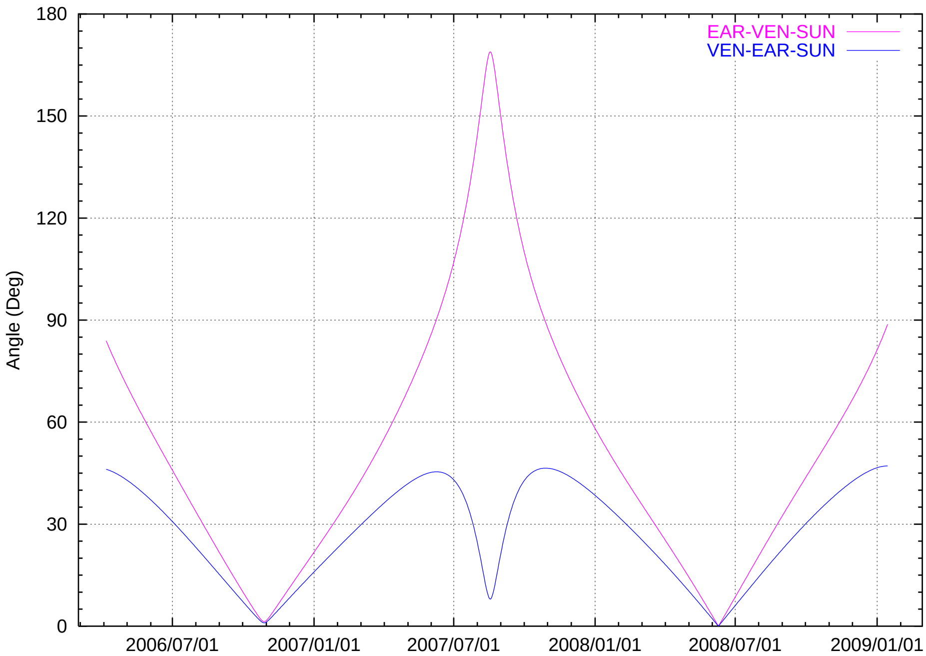

Figure 6-3: Geometric angles of Sun, Earth and Venus during the extended mission lifetime

6.2 Orbital evolution as Function of the Launch Day

6.2.1 Evolution of orbital elements

As it was discussed in Chapter 4, the angular parameters r.a. of ascending node, Ω, Figure 6-4 and argument of pericenter, ω, Figure 4-17, of the operational orbit change with the launch day. As a result, the evolution of the orbit during the mission lifetime can be very different.

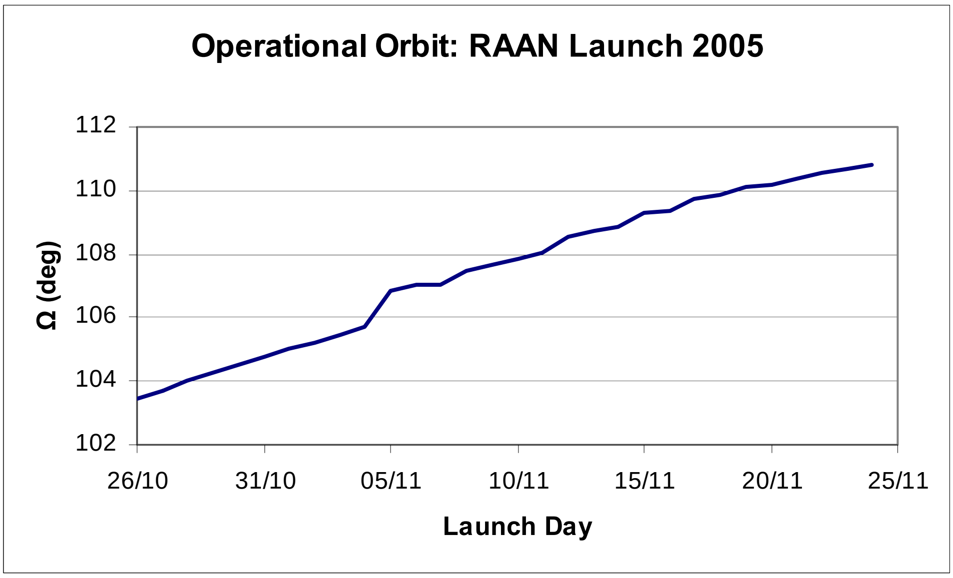

Figure 6-4: Range of initial r.a. ascending node of the operational orbit around Venus

In the previous issue of the CRMA, the argument of pericenter of the operational orbit, which is the driven parameter for the evolution of the altitude of the pericenter, had a variation over the launch window of about 15 deg. Therefore, the evolution of the orbital parameters for the extended mission lifetime of 1000 days is very different for a launch on 26/10/2005 to a launch on 24/11/2005. If the height of pericenter is not controlled, the Sun perturbations will change it and after 1000 days, its final value would vary from about 1000 Km for a launch on 26/10 to about 3000 Km for a launch on 24/11. The height of pericenter would increase about 3 times faster at the end of the launch window and the cost of controlling the pericenter height would also be multiplied by this factor.

However, the new capture strategy makes use of asymmetrical thrusts with the main engine to modify the initial argument of pericenter of the operational orbit in such a way that the variation over the launch window will be of only 5 deg. The rate of increase of the altitude of the pericenter is reduced at the end of the launch window. The final altitude of the pericenter after 1000 days will vary between 1000 Km and 1700 Km, Figure 6-5. Therefore, the number of manoeuvres to control the pericenter height at the end of the launch window is smaller with the new capture strategy.

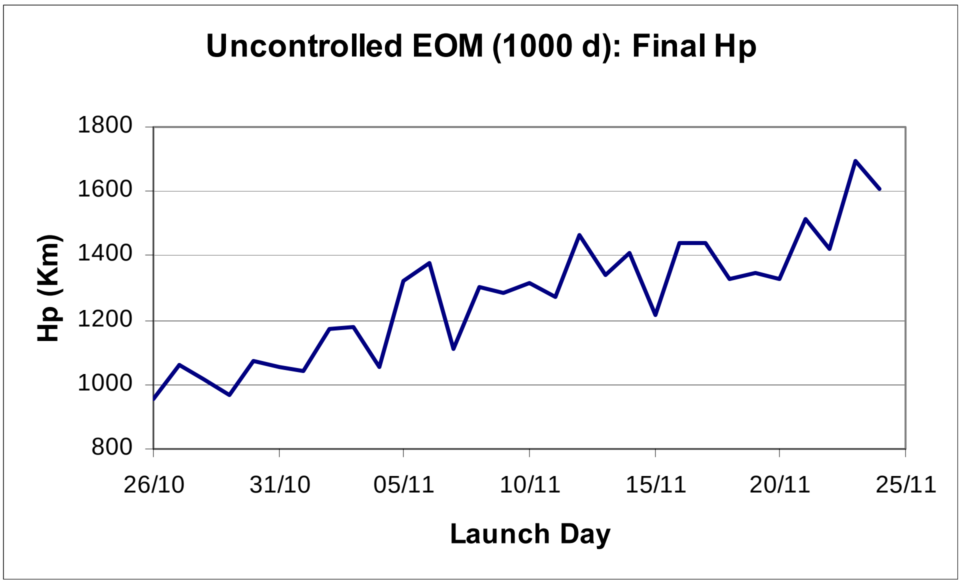

Figure 6-5: Final height of pericenter after 1000 days if no pericenter control is performed

The final values of the orbital angular parameters i, Ω, and ω do not depend on whether the pericenter height is controlled or not, Figure 6-6, Figure 6-7. In all cases the pericenter latitude drifts towards the North Pole about 8.8 deg in 1000 days and the right ascension of the ascending node decreases between 0.7 and 1.0 deg, while the inclination stays very close to 90º.

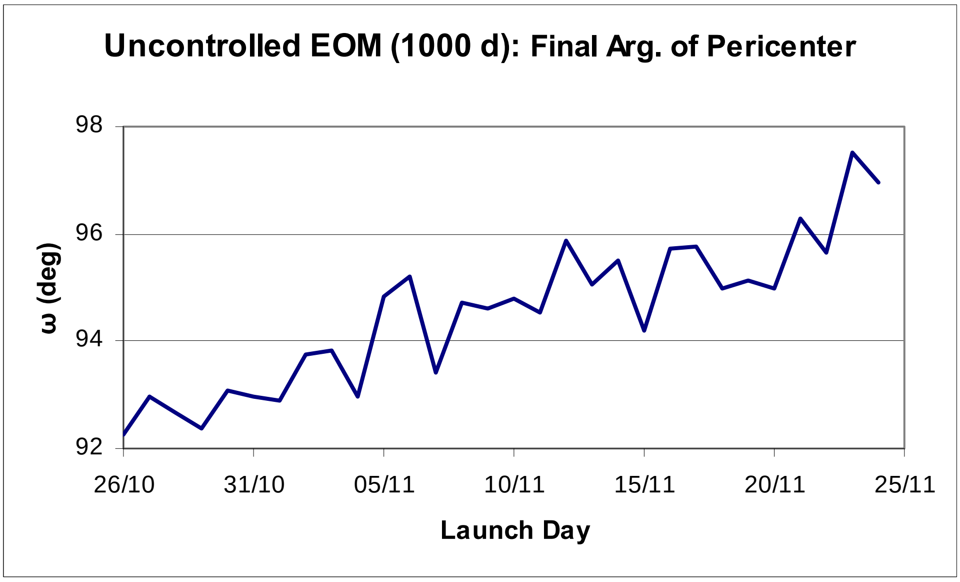

Figure 6-6: Final argument of pericenter after 1000 days when no pericenter control is performed

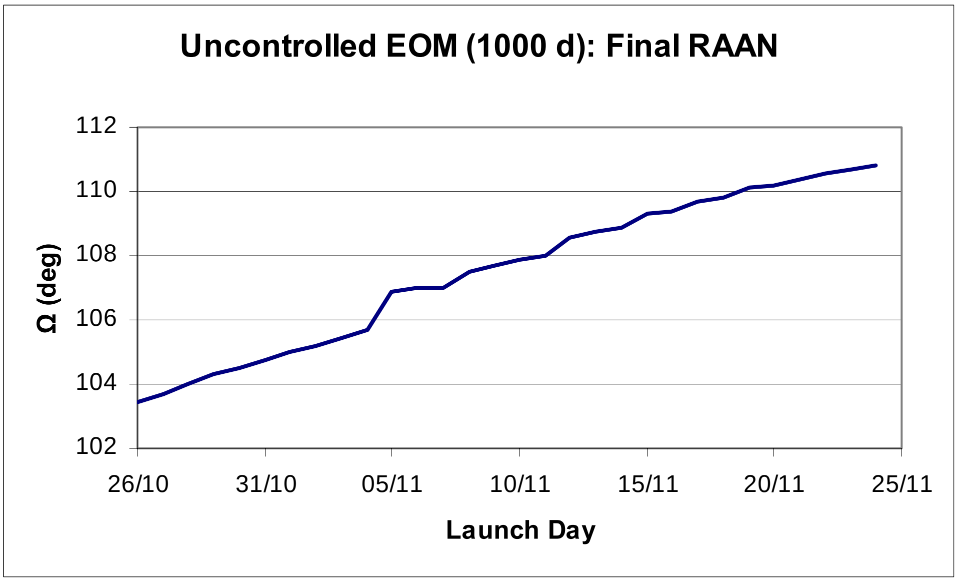

Figure 6-7: Final right ascension of ascending node after 1000 days when no pericenter control is performed

6.2.2 Variation of eclipse time and Earth occultation

The variation of the eclipse time, and of the Earth occultation as a function of the launch day is shown in Figure 6-8. The maximum eclipse time increases by about only 4 minutes from a launch on 26/10 to a launch on 25/11/2005.

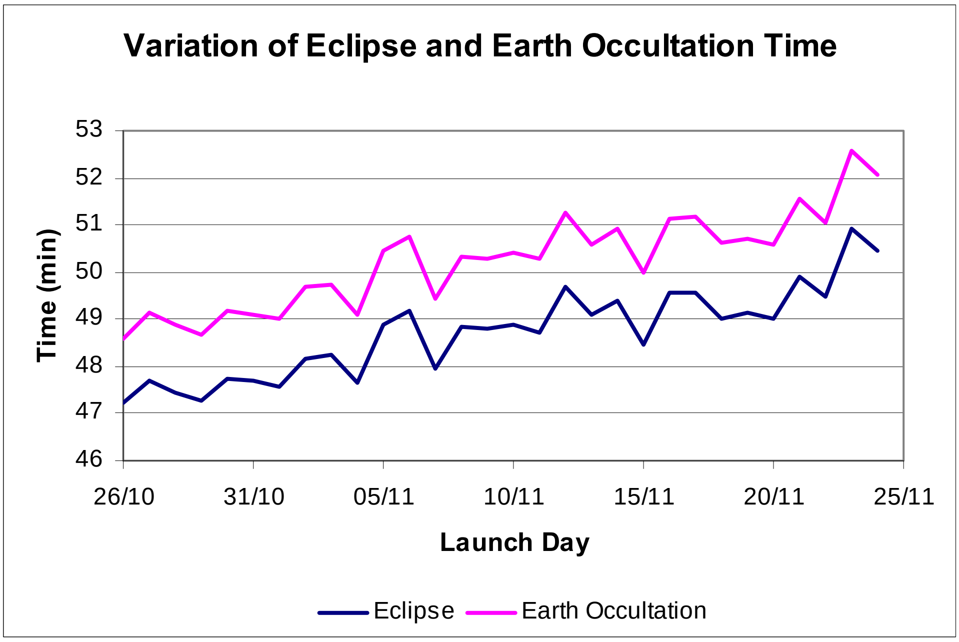

Figure 6-8: Maximum eclipse and Earth occultation times as function of the launch day

6.3 Control of pericenter height

Most of the scientific observations will be done near pericenter at short distances from the surface of Venus. As a compromise between the observations and the atmospheric drag, the pericenter height at BOL is selected to 250 km. However, due to the Sun gravitation there are periodic and secular variations of the height of pericenter, such that the pericenter increases and the apocenter decreases by the same value. The effect of the gravity pull of the Sun makes the pericenter height to increase as shown in Figure 6-5.

To compensate this effect a series of manoeuvres must be implemented whenever the pericenter rises to about a set limit Hp+Hmax. While two manoeuvres would be needed only to correct the pericentre altitude, a series of three consecutive manoeuvres is needed to maintain the pericenter time and thus the phasing with the ground station. Several manoeuvre strategies have been studied and are presented in Table 11, where every strategy is named with the initials of the sequence of manoeuvres meaning A apocenter and P pericenter.

Sequence	**A-P-A**	**P-A-P**	**A-P-P**	**P-P-A**	**P-A-A**	**A-A-P**
Initial Orbit	hp+Hmax x ha-Hmax					
Orbit 1	hp+Hmax/2 x ha-Hmax	hp+Hmax x ha-Hmax/2	hp x ha-Hmax	hp+Hmax x ha-3·Hmax/2	hp+Hmax x ha	hp+3·Hmax/2 x ha-Hmax
Orbit 2	hp+Hmax/2 x ha	hp x ha-Hmax/2	hp x ha+Hmax/2	hp+Hmax x ha	hp-Hmax/2 x ha	hp x ha-Hmax
Final Orbit	hp x ha					
ΔV (m/s/Km)	0.0668	0.0668	0.0722	0.0722	0.129	0.129
Time (periods)	1.0	1.0	1.5	1.5	1.5	1.5

Table 11: Summary of pericenter height control possible strategies.

The strategies that are clearly better are APA and PAP, resulting in a reduction of the ΔV needed per correction manoeuvre of about a 7 % with respect to the APP strategy suggested in the first issue of the CRMA. Another advantage of these strategies is that they are implemented in only one orbit instead of the one and a half of the others, which means that the spacecraft is quicker ready for new observations. The APA strategy is the most desirable operationally as there are more manoeuvres at the apocenter, where the visibility from the ground station allows monitoring the manoeuvre and as well, there are no eclipses or occultations. The last two strategies are the worst in terms of ΔV and also the PAA strategy is unfeasible because it forces pericenter altitudes below the nominal of 250 Km.

For the selected nominal strategy APA and the parameters of the operational orbit, the total ΔV needed per correction is: 0.0668·Hmax, where Hmax is km and ΔV in m/s. The current used values are: Hp = 250 km, and Hmax = 150 km, that means about 10 m/s per correction.

As well as the manoeuvres to control the pericenter height, a couple of manoeuvres have to be implemented in order to maintain the visibility configuration from the nominal ground station in Cebreros. As explained in Section 10, an orbital period change of about 150 seconds (equivalent to about 50 km in semi-major axis) is needed during the period of time when the relative movement of Venus to the Earth induces a secular rate in the visibility. This can be achieved by single manoeuvres of about 0.7 m/s. This correction manoeuvre can also be coupled with a pericenter height control manoeuvre occurring at about the same time with a little ΔV saving.

The different rate of increase of the height of pericenter as function of the launch day, result in different characteristics of the orbit control. This is shown in the following Table 12. The important characteristics are: Number of corrections of the pericenter height and ΔV needed for controlling the orbit during the lifetime; minimum and maximum interval between control manoeuvres. The ΔV has been subdivided in the required just for pericentre control manoeuvres, just for visibility phasing corrections or for mixed manoeuvres with both effects.

Orbit Control Simulation:				Pericentre Height (250-400 Km) Visibility from Cebreros			
Launch Day	Correction Manoeuvres					Time between man. (days)	
	Number	ΔV Total (m/s)	ΔV hp (m/s)	ΔV Visib. (m/s)	ΔV Mixed (m/s)	Min	Max
26-10-2005	4	40.0	29.3	1.4	9.3	80.6	349.2
27-10-2005	5	47.9	35.7	1.4	10.8	61.0	234.7
28-10-2005	5	44.4	32.3	1.3	10.7	52.0	319.7
29-10-2005	4	40.9	38.8	2.1	0.0	30.5	399.2
30-10-2005	5	48.7	36.5	1.4	10.8	61.0	231.6
31-10-2005	5	47.5	35.4	1.4	10.7	60.0	235.7
01-11-2005	5	46.3	34.2	1.3	10.7	55.0	244.7
02-11-2005	6	56.5	45.7	1.4	9.4	54.4	319.7
03-11-2005	6	56.9	54.8	2.1	0.0	50.4	249.7
04-11-2005	5	47.2	35.2	1.3	10.7	56.0	238.7
05-11-2005	7	69.0	39.6	0.0	29.4	84.6	274.7
06-11-2005	7	72.2	60.2	1.3	10.7	39.0	199.6
07-11-2005	5	51.5	39.4	1.4	10.7	63.0	226.1
08-11-2005	6	67.4	47.3	0.0	20.1	68.5	196.6
09-11-2005	6	62.2	50.0	1.4	10.8	24.5	210.1
10-11-2005	7	68.5	39.0	0.0	29.5	68.5	240.7
11-11-2005	6	62.3	60.2	2.1	0.0	23.5	209.6
12-11-2005	7	82.4	81.0	1.4	0.0	49.5	178.6
13-11-2005	7	71.2	50.3	0.7	20.2	58.5	215.6
14-11-2005	7	72.3	60.2	1.3	10.8	40.0	168.1
15-11-2005	6	59.7	57.6	2.1	0.0	49.4	222.6
16-11-2005	7	72.4	70.3	2.1	0.0	25.0	186.1
17-11-2005	7	72.5	70.4	2.1	0.0	24.0	185.1
18-11-2005	7	69.7	48.8	0.7	20.2	25.5	231.6
19-11-2005	7	71.4	59.2	1.4	10.8	20.5	213.6
20-11-2005	7	69.2	48.4	0.7	20.0	31.4	231.6
21-11-2005	8	82.4	80.4	2.1	0.0	31.5	185.6
22-11-2005	7	72.4	60.2	1.3	10.8	39.0	176.1
23-11-2005	9	92.7	80.5	1.4	10.8	34.4	134.6
24-11-2005	8	82.8	70.7	1.4	10.8	44.5	162.1

Table 12: Control of the pericenter height for a period of 1000 days.

Figure 6-9 and Figure 6-10 graphically present the values of Table 12. In the previous issue of the CRMA, the requirement of delta-V for controlling the pericenter height increased up to 172 m/s for a launch on 25/11/2005. For a launch on the first launch day, 26/10/2005, 4 manoeuvres are needed to control the pericenter and 40 m/s required during the EML. The new capture strategy reduces drastically the required delta-V for control at the end of the launch window. The maximum requirement is for a launch on 23/11/2005, when 9 manoeuvres are needed and the total delta-V grows up to 93 m/s.

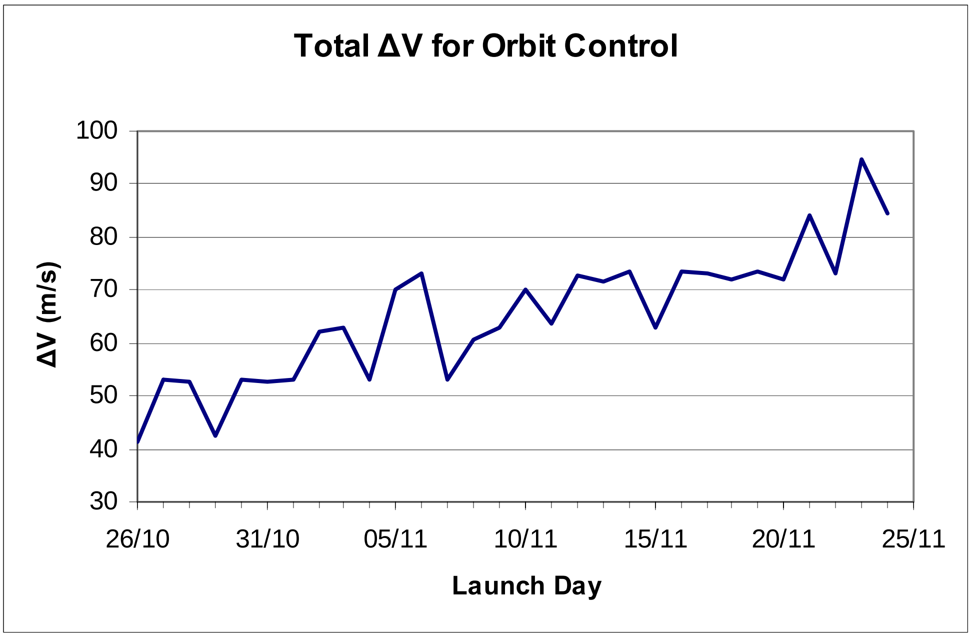

Figure 6-9: Total Delta-V required to maintain the orbit in the interval 250-400 km for a mission lifetime of 1000 days.

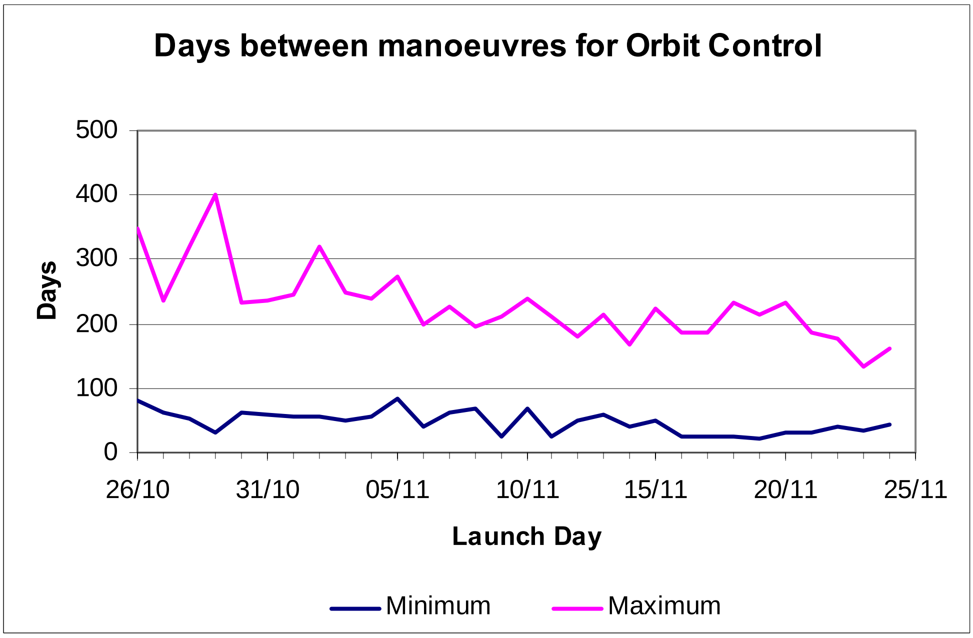

Figure 6-10: Maximum and minimum interval without orbit maintenance manoeuvres

6.3.1 Simulation of Pericenter Control

The control as described in Section 6.3 has been applied to the cases of launch days 26/10/2005, 05/11/2005, 15/11/2005, and 25/11/2005, Figure 6-11 to Figure 6-14, respectively. The control is applied automatically whenever the height is near the limit of 400 km, and it takes into consideration the changes in rate of increase of the pericenter height. These variations are due to the effect of the Sun, and they become very flat periodically.

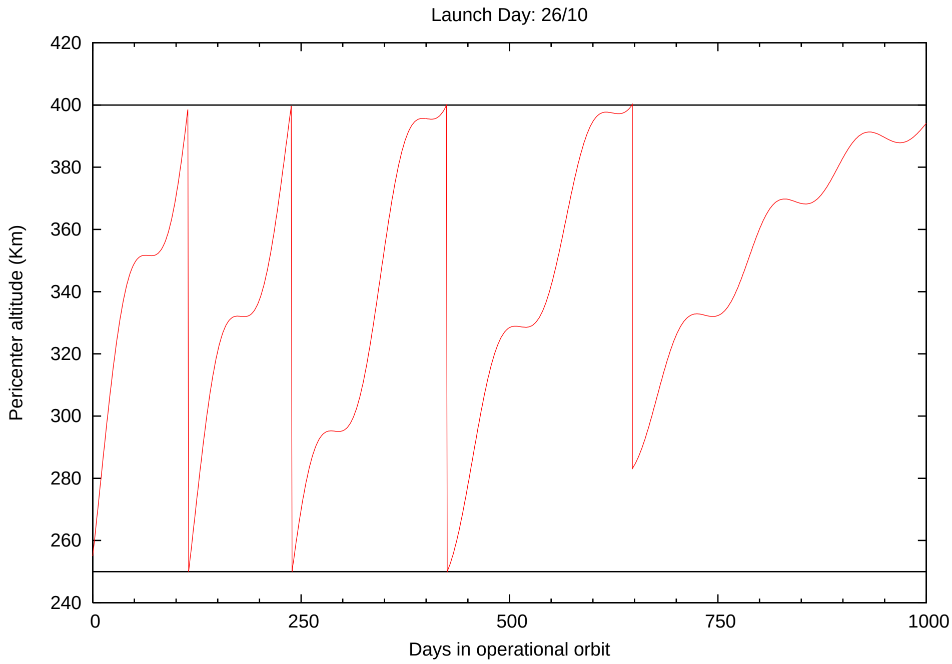

Figure 6-11: Simulation of control of the pericenter height. Launch 26/10/2005

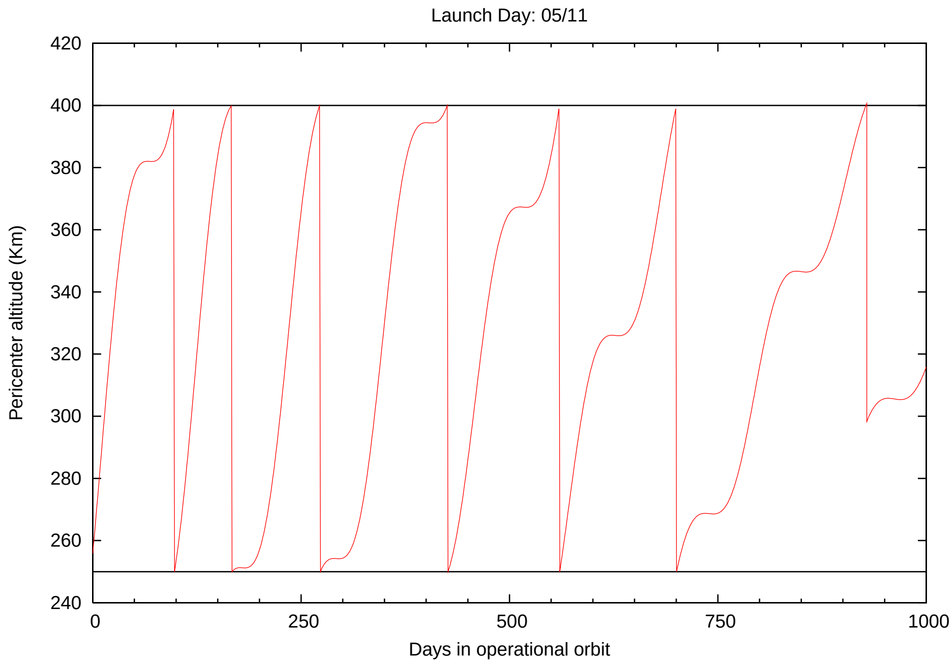

Figure 6-12: Simulation of control of the pericenter height. Launch 05/11/2005

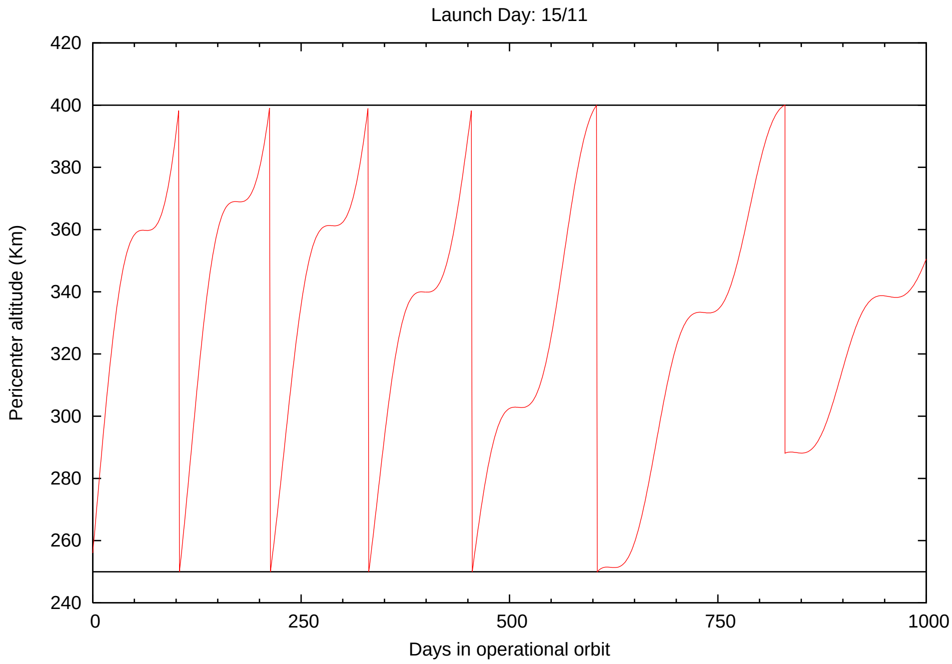

Figure 6-13: Simulation of control of the pericenter height. Launch 15/11/2005

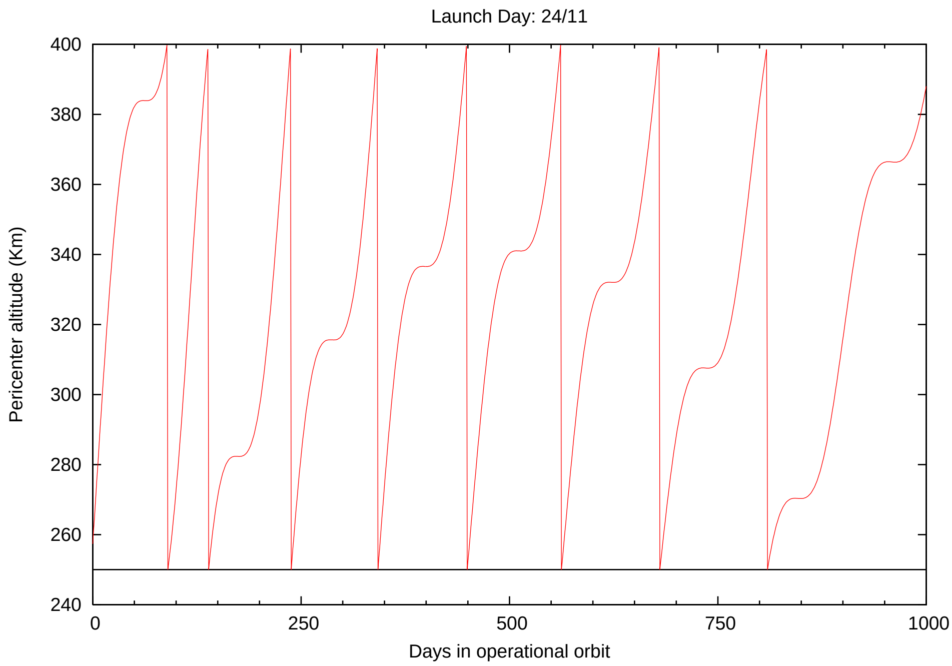

Figure 6-14: Simulation of control of the pericenter height. Launch 24/11/2005

As a result of the higher rate of change of the altitude of the pericenter as the launch day moves to the end of November, the frequency of the manoeuvres and, therefore, the total propellant used increases.

6.3.2 Extension of Mission Operational Time

The requirement MISS 070 of section 3.3 establishes that the consumables shall be capable of maintaining the spacecraft in operation for at least 4 Venus days, equivalent to 1000 days. This is the extended mission lifetime (EML) used in the previous sections. However, the spacecraft will have the possibility to extend the operations until the on-board propellant is exhausted.

After the capture and insertion into the operational orbit, the mass of the spacecraft can be computed and also the available ΔV for maintenance, discounting the propellant reserves for wheel off-loadings and safe modes, which together represent about 10 Kg. The ΔV required for the maintenance of the operational orbit during the EML has been presented in Table 12. The difference of the available and the required gives the ΔV budget for the mission. A positive value indicates that the spacecraft could continue operating after 1000 days in the operational orbit.

An estimation of the complete duration of the mission is obtained extending the propagation of the operational orbit until the propellant is finished. The pericenter altitude of the spacecraft would continue to grow until the argument of pericenter with respect to the Venus ecliptic plane gets below 90 degrees. After that the pericenter altitude decreases with time. Whenever the pericenter altitude gets higher than 400 Km or lower than 250 Km a correction manoeuvre is implemented. In addition, the last manoeuvre before the pericenter altitude starts decreasing can be computed in order to optimize its magnitude in such a way that the altitude increases up to a maximum of 400 km and then decreases.

It is assumed that in the propellant tanks a total of 7.7 Kg of residuals is left (propellant that can not be used). After the EML, the propellant needed for wheel off-loadings, safe modes and changes of the phasing with the ground station are discounted at a rate of 0.032 m/s per day. The mission operations can be extended until there is no more propellant left. Then, the spacecraft will have no possibility to correct the orbit and the pericenter altitude will get lower until the entry into Venus atmosphere.

A complete analysis of the optimization of the total mission lifetime is presented in R 8. Figure 6-15 to Figure 6-18 present the evolution of the pericenter height of the operational orbit for launches on 26/10, 05/11, 15/11 and 24/11. The first vertical line represents the EML, while the second vertical line represents the end of the spacecraft capability to control the orbit. For the launches on 26/10, 05/11 and 15/11 the total mission lifetime extends beyond 2000 days. However, for a launch on the last launch window day, 24/11, the number of manoeuvres required during the EML is much bigger and the controlled duration of the mission is about 1250 days.

Table 13 presents the ΔV available for control at the arrival to the operational orbit and the ΔV margin once the requirement for the EML has been discounted. Furthermore, it shows the total number of manoeuvres to maintain the pericenter altitude between 250 and 400 Km, the mission lifetime with control capability and duration of the orbit of the spacecraft around Venus.

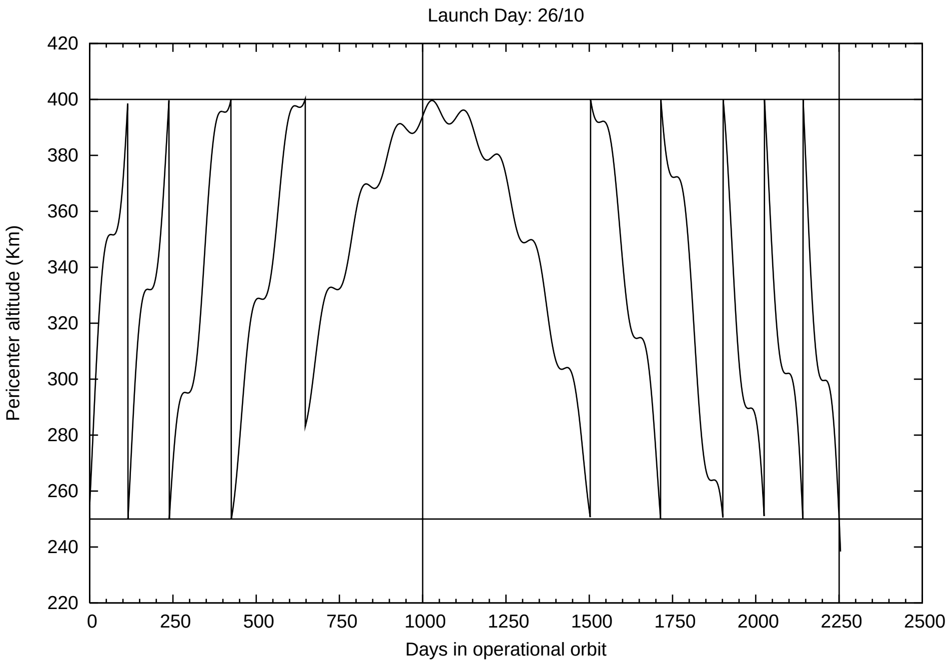

Figure 6-15: Evolution of the pericenter height until the end of mission. Launch 26/10/2005

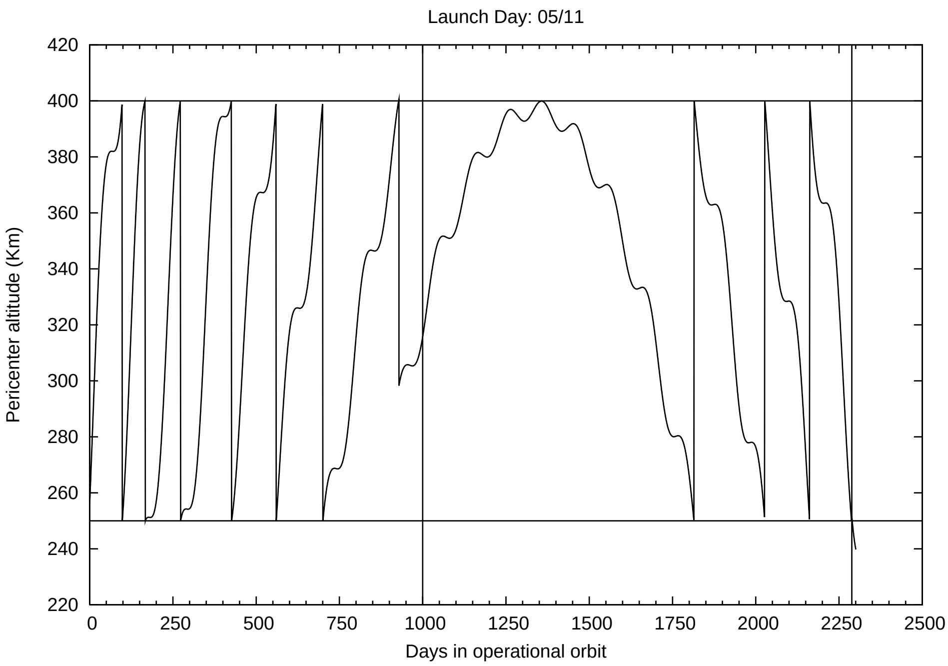

Figure 6-16: Evolution of the pericenter height until the end of mission. Launch 05/11/2005

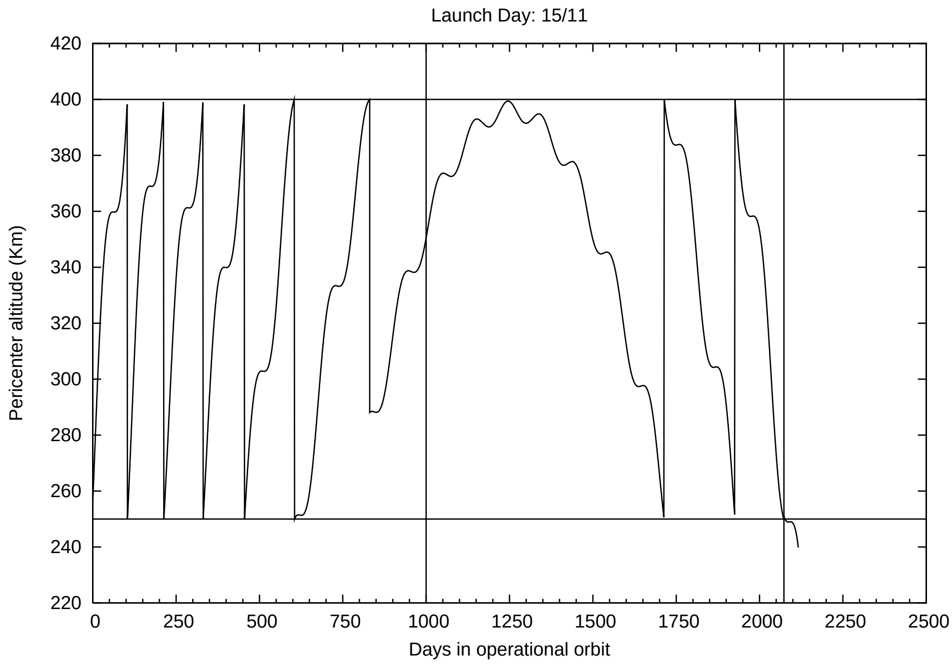

Figure 6-17: Evolution of the pericenter height until the end of mission. Launch 15/11/2005

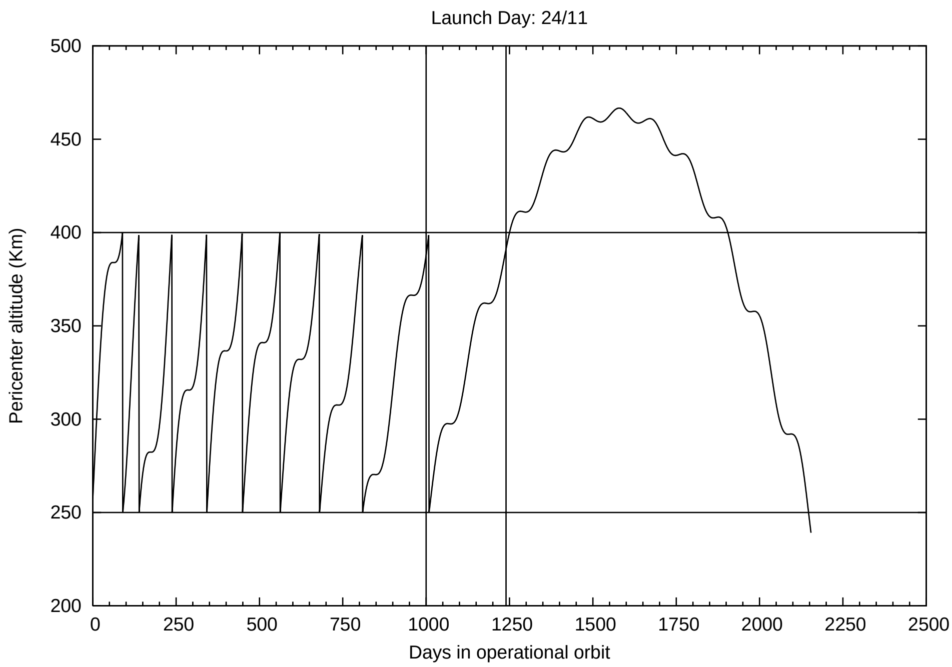

Figure 6-18: Evolution of the pericenter height until the end of mission. Launch 24/11/2005

Launch dd/mm	ΔV_{LEFT} m/s	ΔV_{MARG} m/s	N_{MAN} Total	Control days	End days	LP
26/10	127.8	87.8	9	2250.3	2254.3	1
27/10	138.1	90.3	10	2337.4	2348.4	
28/10	146.2	101.8	12	2381.4	2384.4	
29/10	142.0	101.1	11	2360.4	2364.4	
30/10	139.4	90.7	11	2349.4	2355.4	
31/10	136.8	89.3	11	2301.4	2343.4	
01/11	135.3	89.1	11	2289.4	2298.4	
02/11	135.2	78.7	11	2286.4	2293.4	
03/11	135.8	78.9	12	2290.4	2298.4	
04/11	136.2	88.9	11	2296.4	2332.4	
05/11	137.9	68.9	11	2288.4	2300.4	2
06/11	146.4	74.2	12	2363.4	2374.4	
07/11	129.3	77.8	9	2257.3	2262.3	
08/11	136.0	68.7	11	2277.4	2284.4	
09/11	147.5	85.4	12	2384.4	2389.4	
10/11	137.1	68.6	10	2282.4	2290.4	
11/11	118.6	56.3	9	2133.3	2146.3	
12/11	127.1	44.7	12	2143.3	2149.3	3
13/11	141.7	70.5	11	2342.4	2352.4	
14/11	119.8	47.4	10	2075.2	2126.3	
15/11	112.0	52.3	9	2073.2	2116.3	4
16/11	125.0	52.7	10	2132.3	2144.3	
17/11	107.9	35.4	9	1919.1	1928.1	5
18/11	111.2	41.5	9	2029.2	2040.2	
19/11	94.1	22.7	8	1755.0	1836.1	6
20/11	91.5	22.4	7	1746.0	1822.1	
21/11	116.2	33.8	10	1955.2	2009.2	7
22/11	103.4	31.0	8	1838.1	1919.1	8
23/11	113.5	20.8	10	1357.8	2231.3	9
24/11	100.0	17.2	9	1239.7	2154.3	10

Table 13: ΔV at operational orbit and estimations of mission duration

For every launch day the propellant margin is positive. The worst case is for launch on 24/11/2005 when only 17.2 m/s will be available at the end of the EML and the mission could be extended for about 240 days. However, after exhausting the propellant, the pericenter height will grow up to 460 Km and then fall into Venus at about 2150 days after the beginning of the operational orbit. For most of the launch days, the spacecraft will be in orbit around Venus for more than 2000 days. The worst case is for launch on 20/11/2005 with 1820 days.

A possible solution to obtain a better performance for the last days of the launch window is presented in R 8. For these days, the operational orbit benefits from a lower argument of pericenter, which can be obtained more efficiently using a larger intermediate orbit of 10 days.

For the case of a 10-days capture orbit, Table 14 presents a summary of the characteristics of the capture and transfer, as well as of the operational orbit. In comparison to the 6.2-days capture orbit, the transfer time to the operational orbit is larger, the delta-V for the capture is smaller, the inclination of the operational orbit is bigger because of the more intense effect of the Sun perturbation to a capture orbit with higher apocenter and the argument of the pericenter is about 2 degrees lower.

Launch	Capture and apocentre lowering					Operational Orbit				
Launch dd/mm	Arrival dd/mm	D_{TRF} days	ΔV_C m/s	ΔV_{1R} m/s	ΔV_{RCT} m/s	Arrival dd/mm	i deg	Ω deg	ω deg	LP
17/11	13/04	17.6	1251.8	288.5	55.2	01/05	90.53	108.72	103.55	5
18/11	13/04	15.2	1249.9	300.6	41.9	28/04	90.57	108.75	104.30	
19/11	14/04	17.6	1255.3	286.2	58.8	01/05	90.57	109.03	104.32	6
20/11	14/04	17.7	1259.5	283.3	63.0	01/05	90.57	109.12	104.08	
21/11	14/04	17.5	1270.1	296.3	48.7	01/05	90.56	109.37	103.81	7
22/11	14/04	17.5	1265.7	299.9	49.4	02/05	90.63	109.42	104.83	8
23/11	14/04	17.7	1282.5	283.9	62.6	02/05	90.57	109.67	104.15	9
24/11	14/04	17.8	1284.8	281.3	66.8	02/05	90.61	109.76	104.71	10

Table 14: Characteristics of transfer and operational orbit at the end of the launch window using a 10-days period intermediate orbit

Table 15 presents the delta-V for control and available at arrival to the operational orbit, the number of pericenter correction manoeuvres during the EML and the total lifetime and the estimations of the mission lifetime with control capability and the total duration of the orbit around Venus. The mission duration is improved specially for launches on 23/11/2005, now with 1854 days instead of 1358 days, and on 24/11/2005, with 1445 days which is about 45 % more than the initial requirement.

Launch dd/mm	ΔV_{CON} m/s	ΔV_{LEFT} m/s	ΔV_{MARG} m/s	N_{MAN} EML	Control days	N_{MAN} Total	End days	LP
17/11	62.4	107.2	44.8	6	2015.2	10	2022.2	5
18/11	72.4	120.2	47.8	7	2112.3	9	2122.3	
19/11	72.2	104.5	32.3	7	1907.1	9	1916.1	6
20/11	72.3	100.4	28.2	7	1886.1	8	1899.1	
21/11	70.0	113.3	43.4	7	2031.2	9	2100.2	7
22/11	82.5	113.4	31.0	8	1941.2	10	2004.2	8
23/11	72.4	100.2	27.9	7	1854.1	9	1903.1	9
24/11	82.5	96.7	14.2	8	1445.9	9	1930.1	10

Table 15: Operational orbit ΔV and mission duration at the end of the launch window using a 10-days period intermediate orbit

Figure 6-19 presents the evolution of the pericenter height of the operational orbit for launch on 24/11/2005 when an intermediate orbit with a period of 10 days is used. The improvement w.r.t. the evolution presented in Figure 6-18 comes from the need of one manoeuvre less and from the fact that the last manoeuvre is just before the end of the EML.

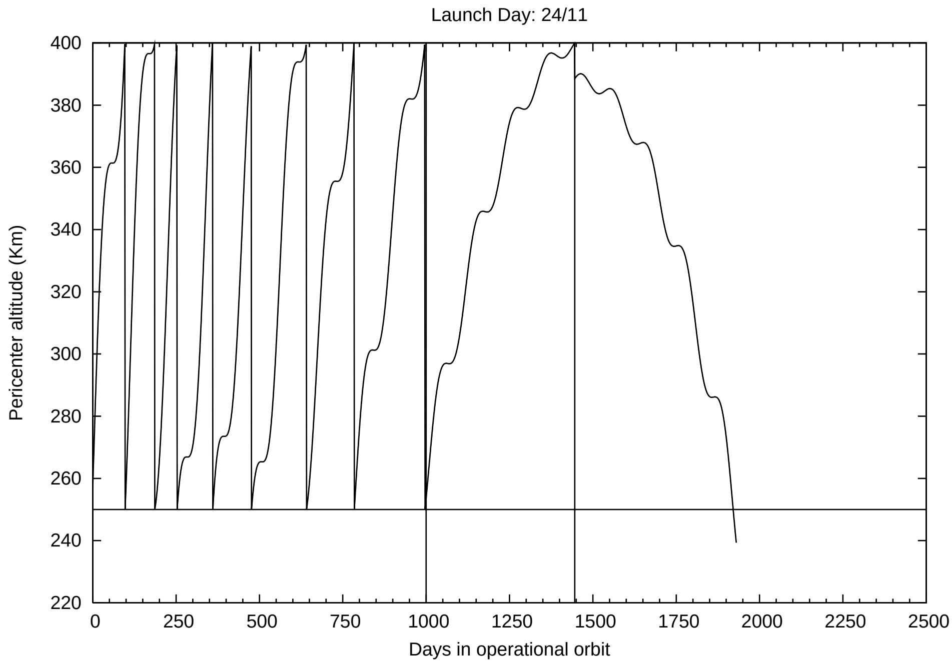

Figure 6-19: Evolution of the pericenter height until the end of mission. Launch 24/11/2005

The strategy for the capture and transfer to the operational orbit that has been finally selected for the operations is based on the one presented in R 11. The capture orbit is 10 days for every launch day, while the capture manoeuvre and the apocentre lowering manoeuvre with the ME are very assymetrical w.r.t. the pericentre and the direction of the thrust presents a tilt angle w.r.t. the velocity direction. The operational orbit has a fixed number of 4 pericentre reductions during the extended mission lifetime of 1000 days.

The total mission lifetime with control capability has been estimated also for the previous strategy. Figure 6-20 compares the estimations from the results given in R 11 with the possible mission extension with the capture, transfer strategy and operational orbits presented in this document, which have been optimized in order to maximize the mission duration for each launch window day. The line that appears in the plot presents the number of additional days that the spacecraft can be controlled in the operational orbit, that is, the reference level of 0 corresponds to the total mission duration with control capability estimated from R 11. For example, for a launch on the 26/10, the mission duration with control estimated for a 10-days capture orbit and an operational orbit with 4 pericentre reductions is 2150 days, while the total duration given in Table 13 is about 2250 days. Therefore 100 additional days of operations could be obtained. The mission lifetime with control capability can be extended up to 230 days and for the most of the launch days more than 50 additional days can be obtained. Only for the last launch window day 24/11, there is no increment of the mission duration.

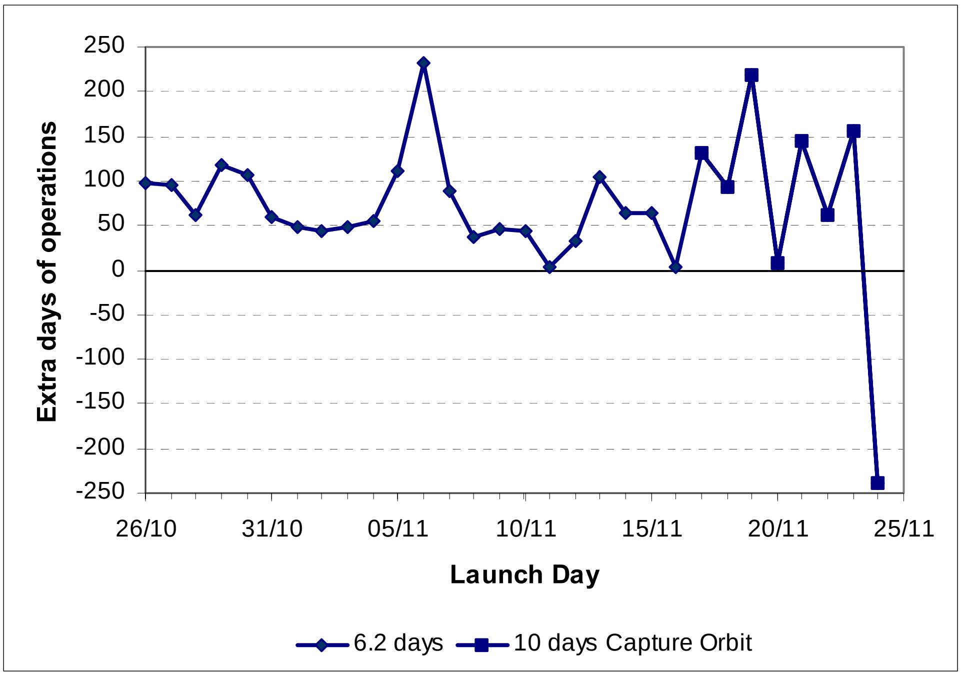

Figure 6-20: Extra days of operations as a function of the launch day w.r.t. 10-days capture orbit and operational orbit with 4 pericentre correction manoeuvres

7. CHARACTERISTICS OF THE OPERATIONAL ORBIT

7.1 General Characteristics

The initial orbital elements of the operational orbit around Venus are given in Table 10, where the right ascension of the ascending node, and the argument of pericenter are values depending on the launch and arrival date, whereas the others are independent of the launch day.

Because Venus flattening is very small (the J_2 term of the gravitational potential is approximately 250 times smaller than the one of the Earth) the rotation of the apsidal line, and the nodal regression are very small in comparison with the values that will appear in an orbit around the Earth or Mars. The only important perturbation is the Sun gravity pull that will increase the pericenter height, and decrease the apocenter height as described in section 6.2.

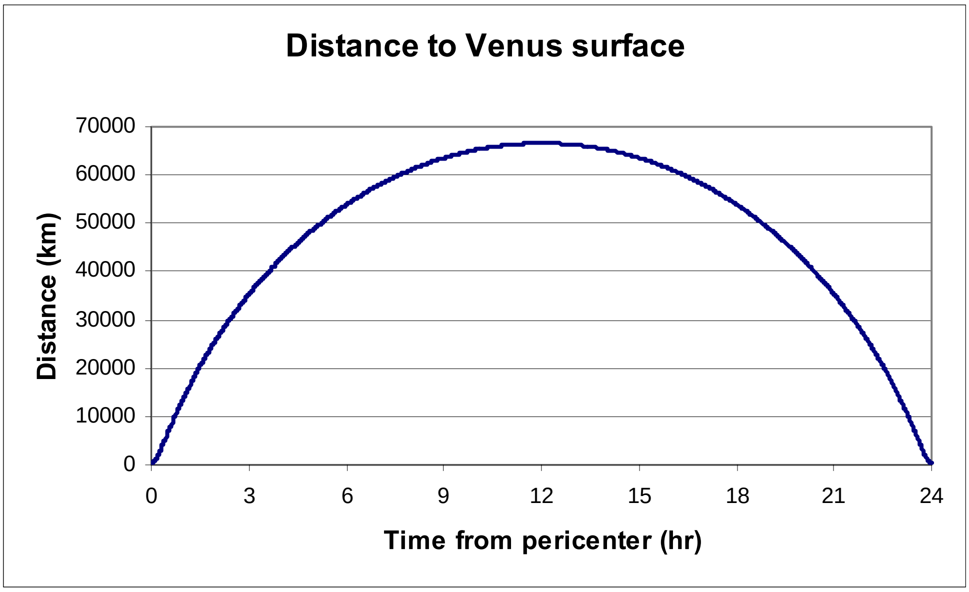

Figure 7-1: Variation of height over one revolution

Figure 7-1 shows the variation of the distance to the surface of Venus over one revolution as a function of time from pericenter, while Figure 7-2 shows the variation of the velocity.

The evolution of the latitude is presented in Figure 7-3.

During the extended lifetime, the longitude of pericenter will perform almost 5 revolutions, Figure 7-9, but the latitude of pericenter will remain in a narrow band of about 8.5°, Figure 7-10.

The distance to the surface of Venus as a function of time from pericenter is presented in, Figure 7-5, for distances of up to 15000 km.

The latitude of the sub-satellite point on the Venus surface region and the altitude at that point is shown in Figure 7-6, for the full orbit, and in Figure 7-7 for the part of the orbit with height less than 15000 km. The latitude region that can be observed at height less than 15000 km is from -47° to 90°. Figure 7-8 presents the variation of the altitude against the latitude of the sub-satellite point with the launch day. For a launch on 24/11/2004 the region that can be observed at height less than 15000 km is slightly smaller from -43° to 90°.

The ground track of the sub-satellite point on Venus surface over the first orbit is shown in Figure 7-4. In section 8 it will the shown the variations of the surface coverage along the mission.

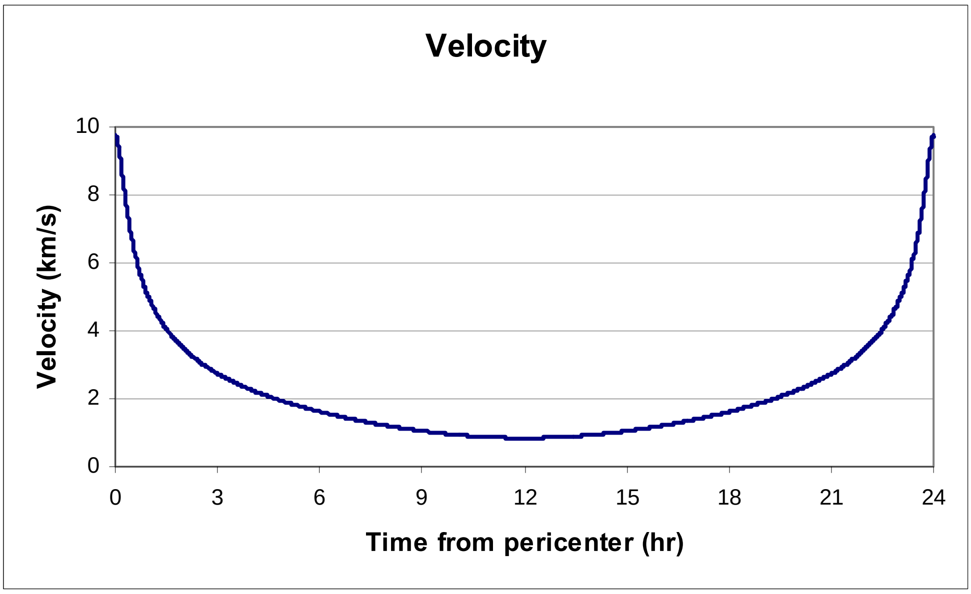

Figure 7-2: Variation of orbital velocity over one orbit

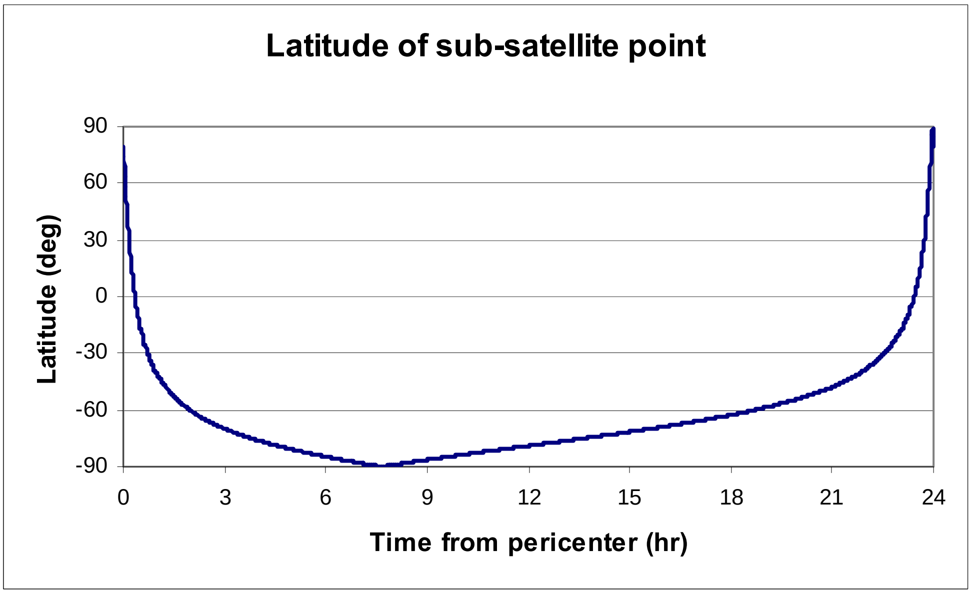

Figure 7-3: Variation of the latitude of the sub-satellite point.

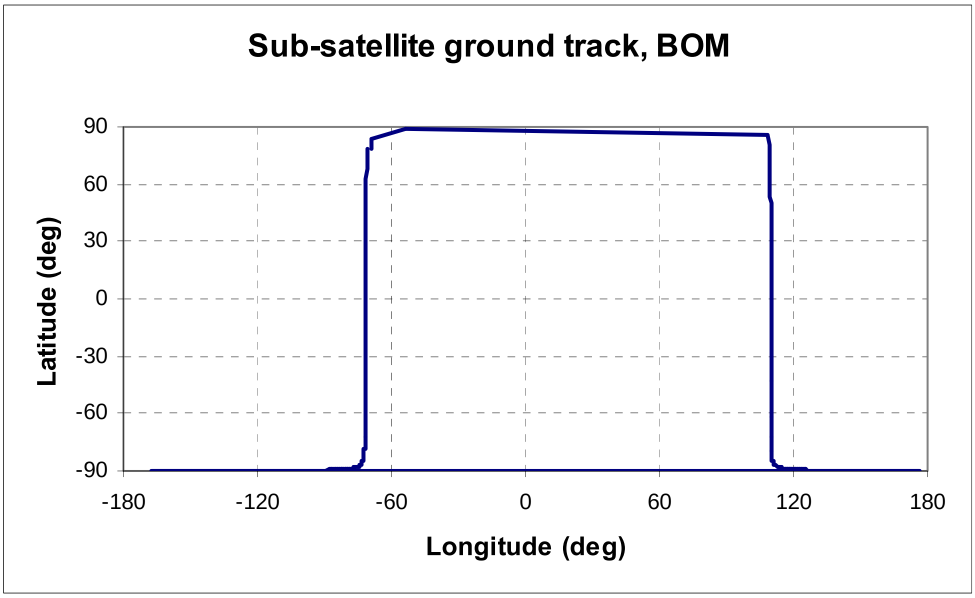

Figure 7-4: Ground track at BOM

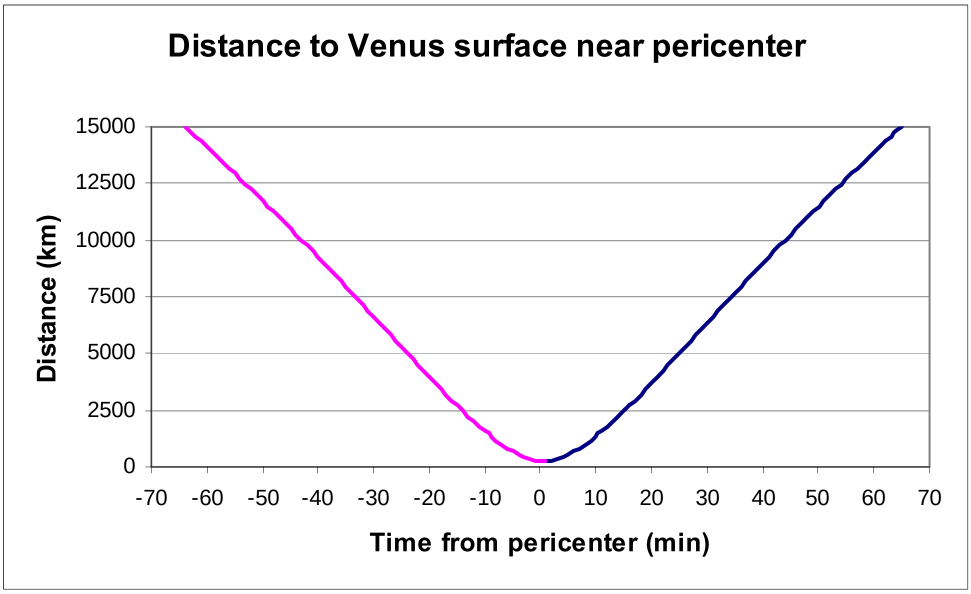

Figure 7-5: Height over Venus surface near the pericenter

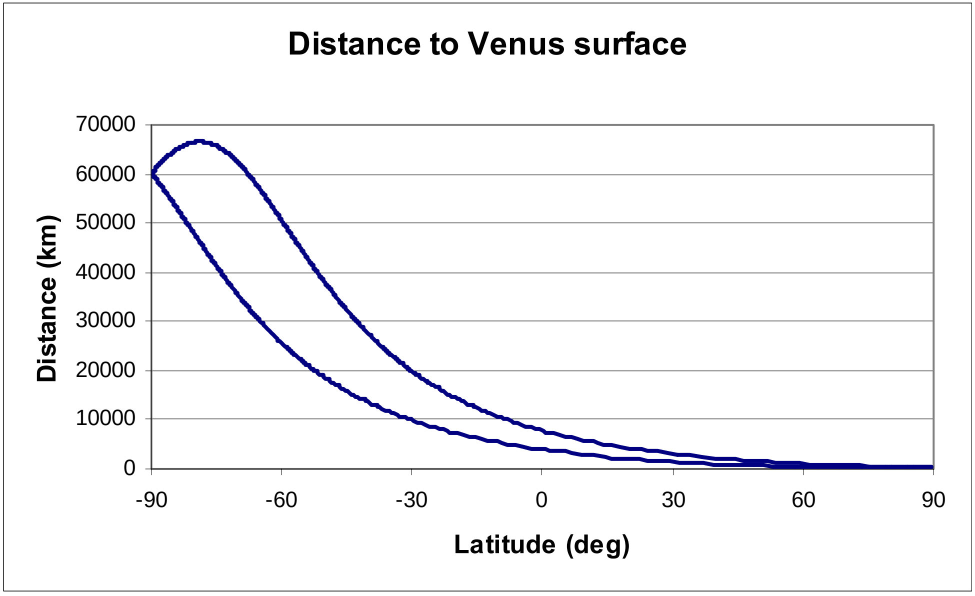

Figure 7-6: Height over Venus surface as function of the latitude of the sub-satellite point.

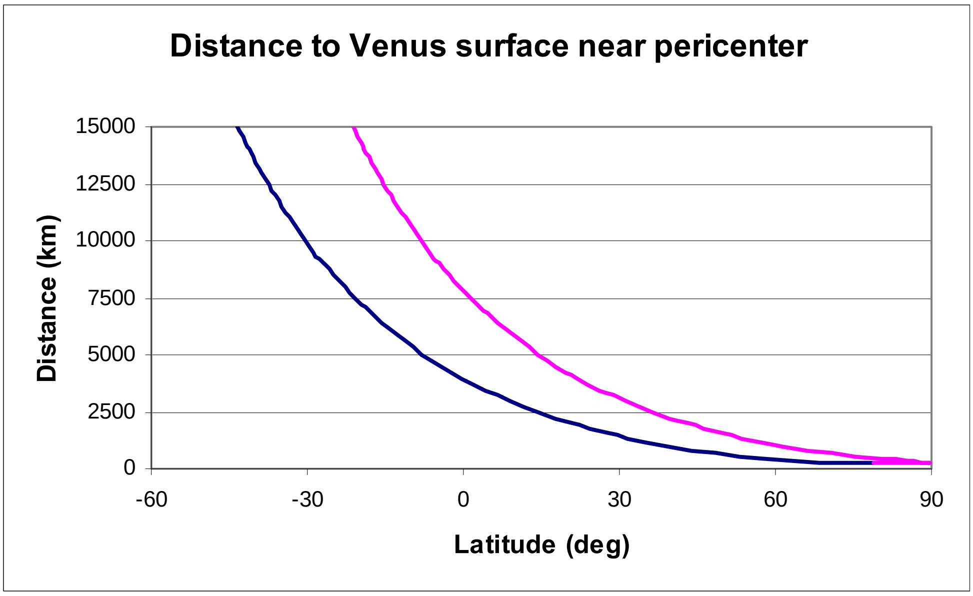

Figure 7-7: Height over Venus surface as function of the latitude of the sub-satellite point for the arc near pericenter.

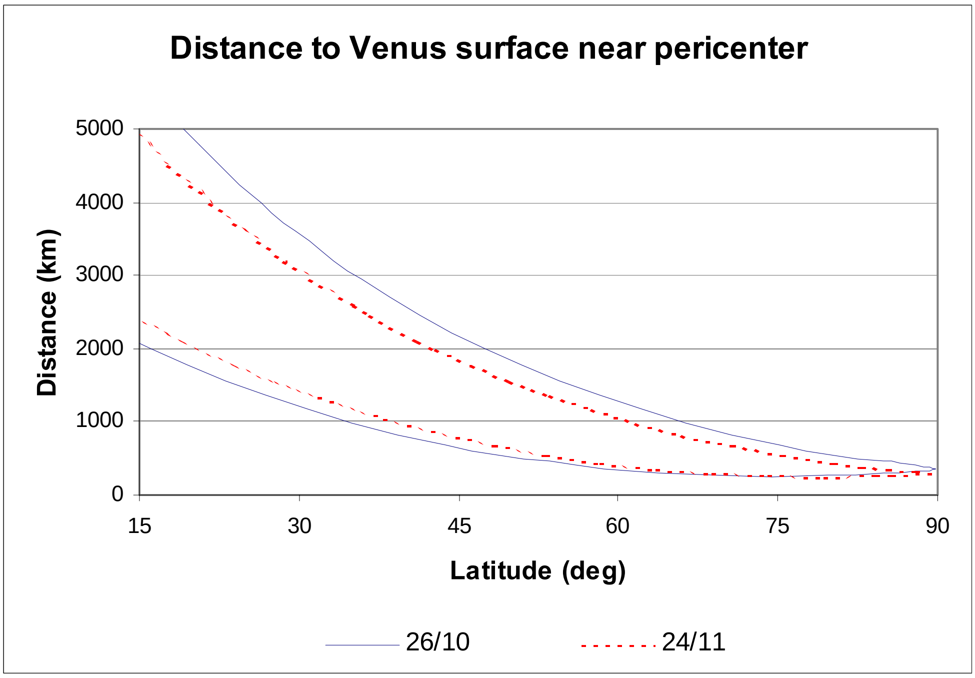

Figure 7-8: Detail of height over Venus surface as function of the latitude of the sub-satellite point for the arc near pericenter for launches on 26/10/2005 and 24/11/2005.

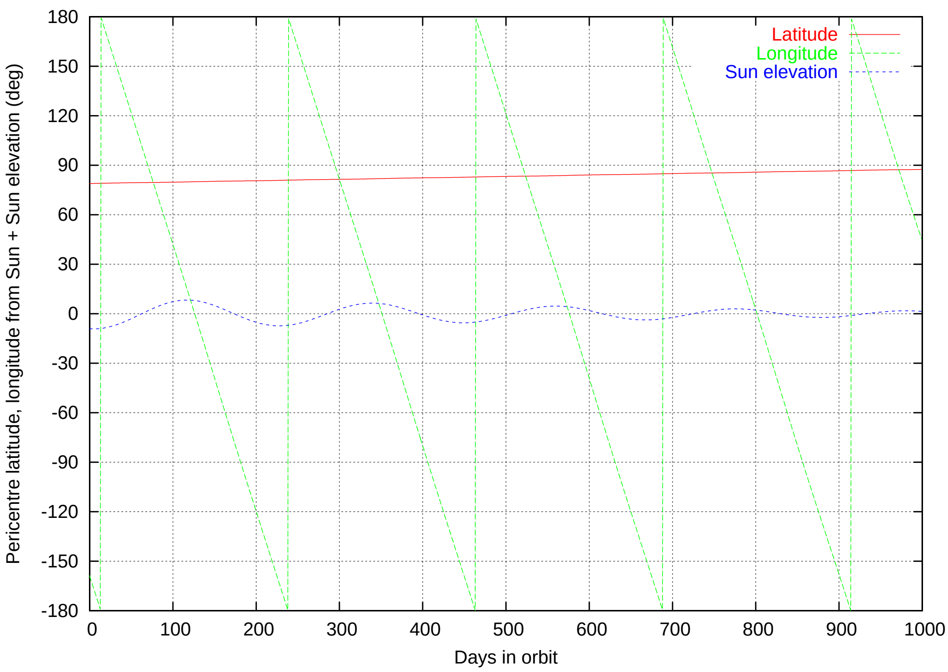

Figure 7-9: Evolution of the pericenter latitude, longitude, and the Sun elevation at the sub-satellite point at pericenter over the extended mission lifetime. Launch 26/10/2005

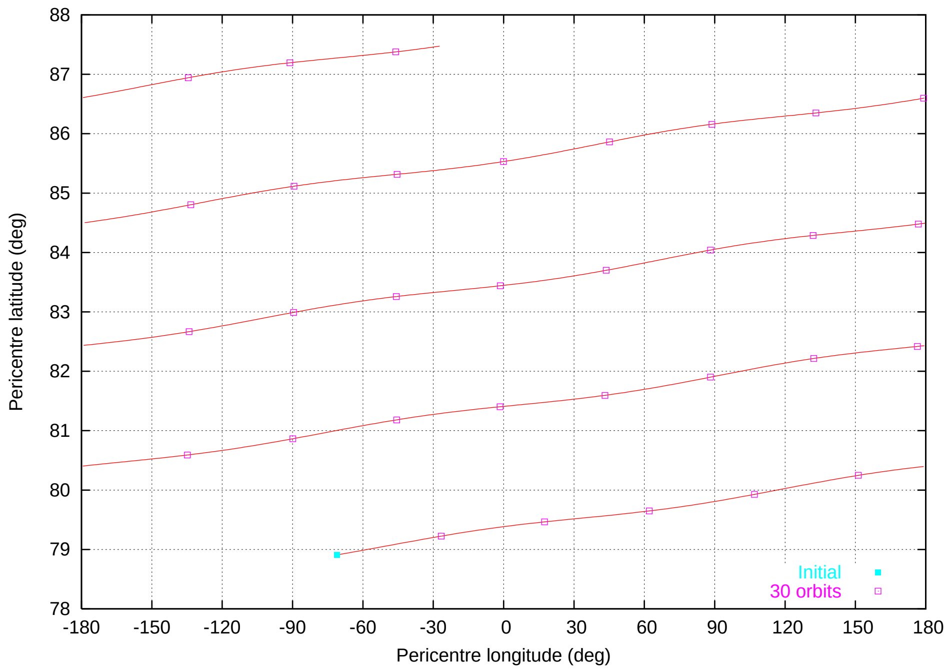

Figure 7-10: Evolution of pericenter latitude and longitude during the extended mission lifetime. Launch 26/10/2005

For the extreme cases of launch, 26/10 and 24/11/2005 Figure 7-11 to Figure 7-14 present the evolution of the right ascension of the ascending node, and of the argument of pericenter. The evolution of these angular elements of the orbit does not depend on whether the pericenter height is controlled or not. The orbital plane does not change more than 1º, but the argument of pericenter changes about 8º.

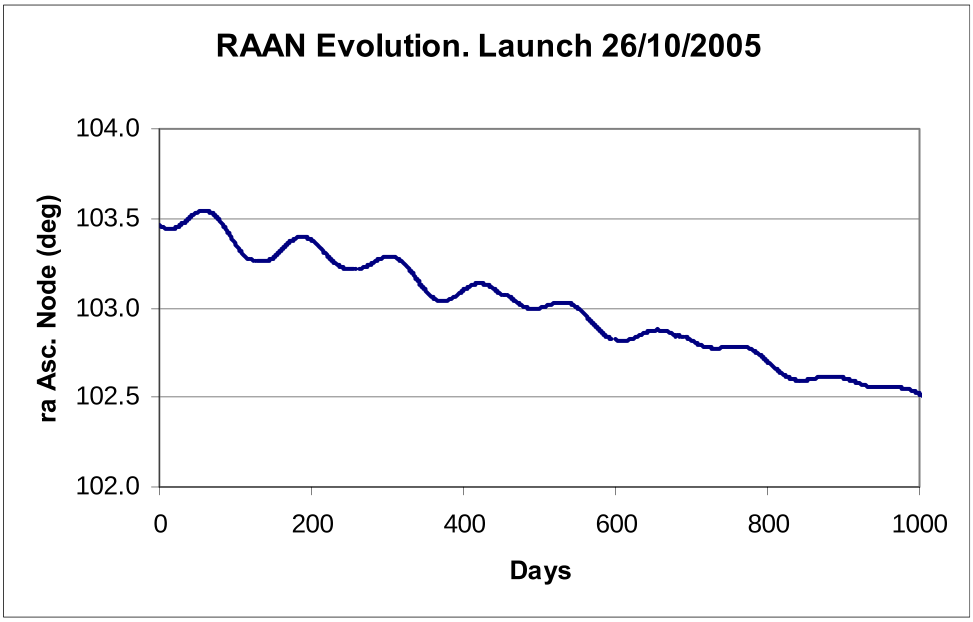

Figure 7-11: Evolution of the right ascension of the ascending node. Launch 26/10/2005

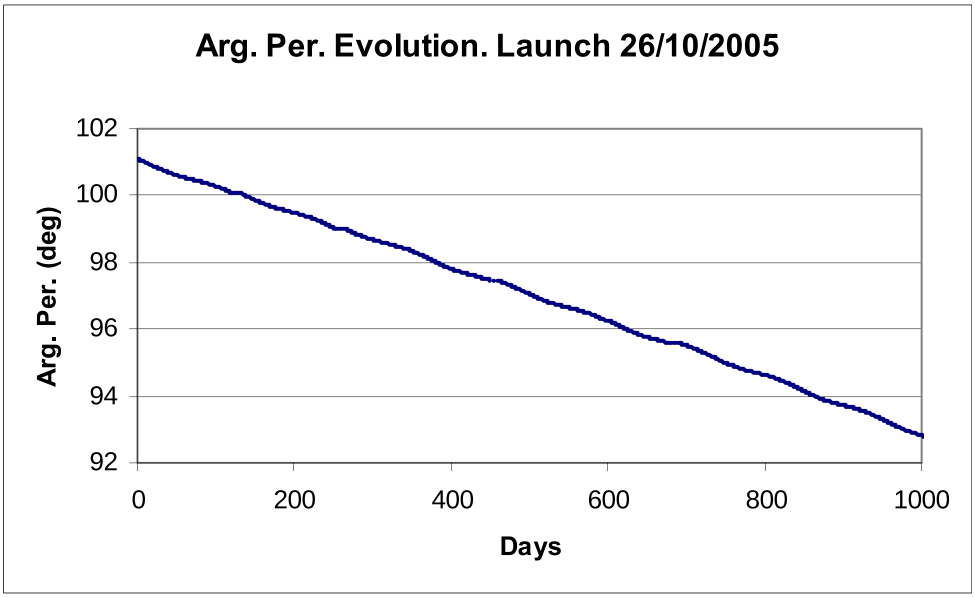

Figure 7-12: Evolution of argument of pericenter. Launch 26/10/2005

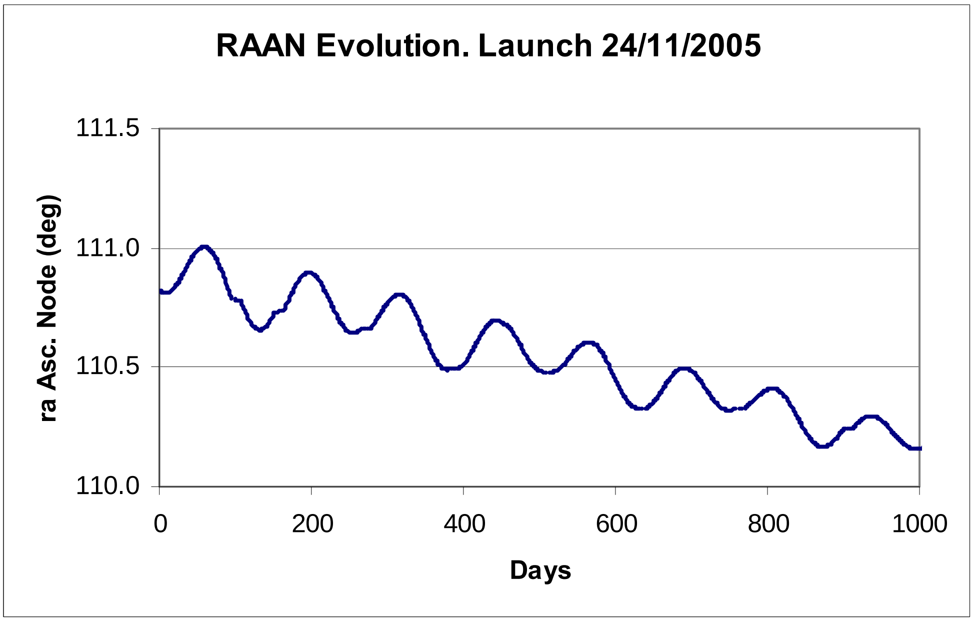

Figure 7-13: Evolution of the right ascension of the ascending node. Launch 24/11/2005

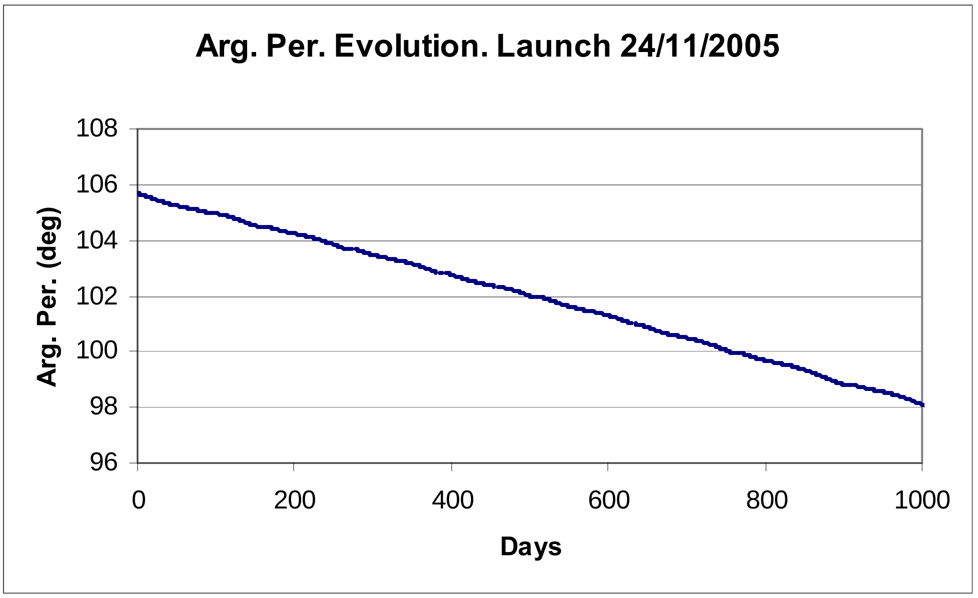

Figure 7-14: Evolution of argument of pericenter. Launch 24/11/2005

7.2 Evolution of Pericenter Characteristics

For the scientific observations near pericenter, it is of interest to know the illumination conditions of the subsatellite point, and the location of the Sun with respect to the point. Figure 7-9 and Figure 7-15 to Figure 7-17 present the variation of the Sun elevation, latitude, and longitude with respect to the Sun of the sub-satellite point at pericenter. Each figure show the evolution for launches on 26/10, 05/11, 15/11 and 25/11/2005, respectively.

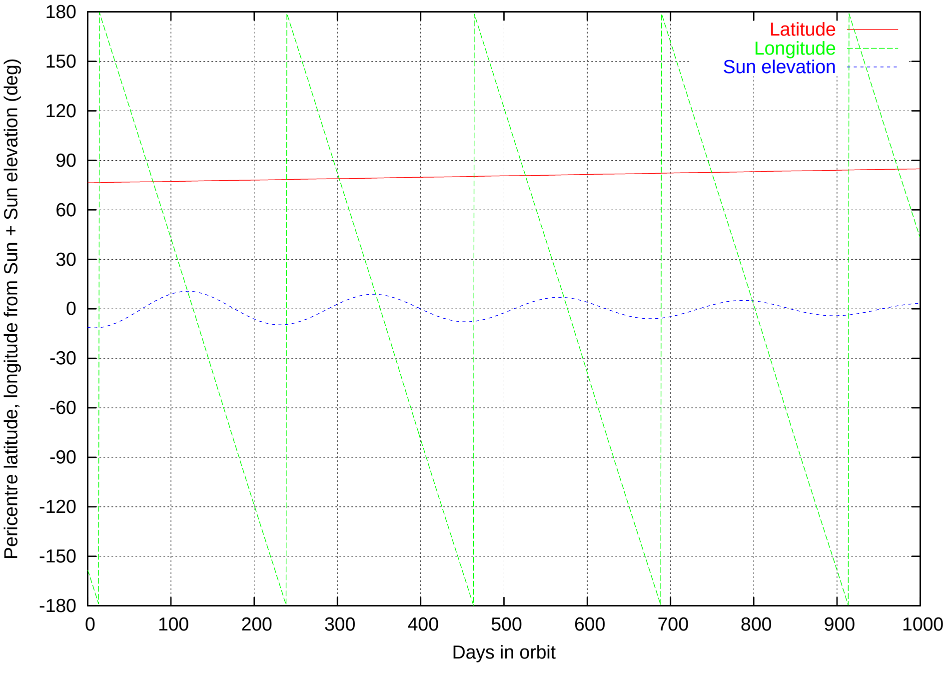

Figure 7-15: Evolution of the pericenter latitude, longitude, and the Sun elevation at the sub-satellite at pericenter over the extended mission lifetime. Launch 05/11/2005

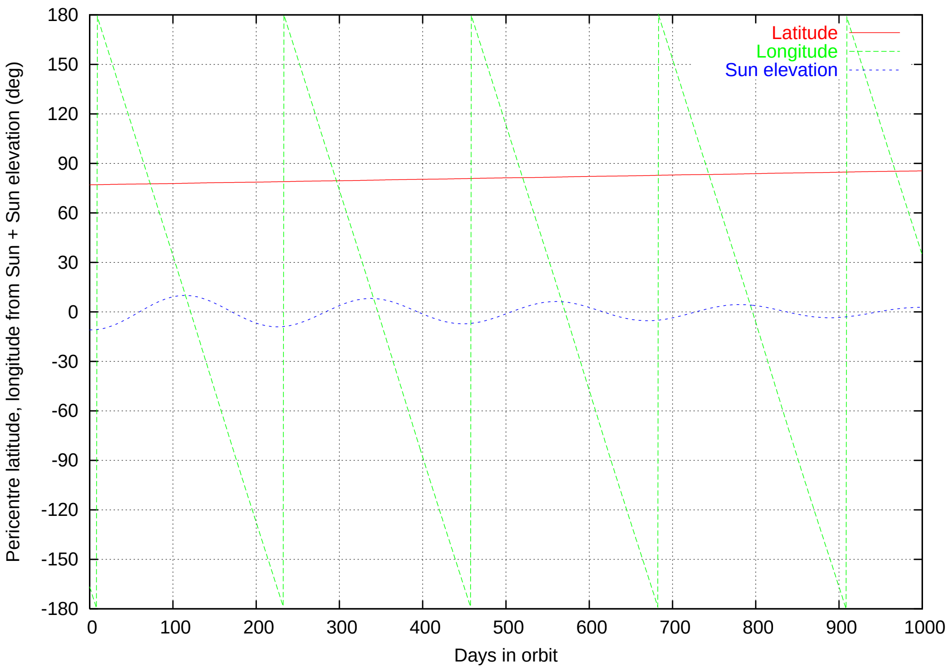

Figure 7-16: Evolution of the pericenter latitude, longitude, and the Sun elevation at the sub-satellite at pericenter over the extended mission lifetime. Launch 15/11/2005

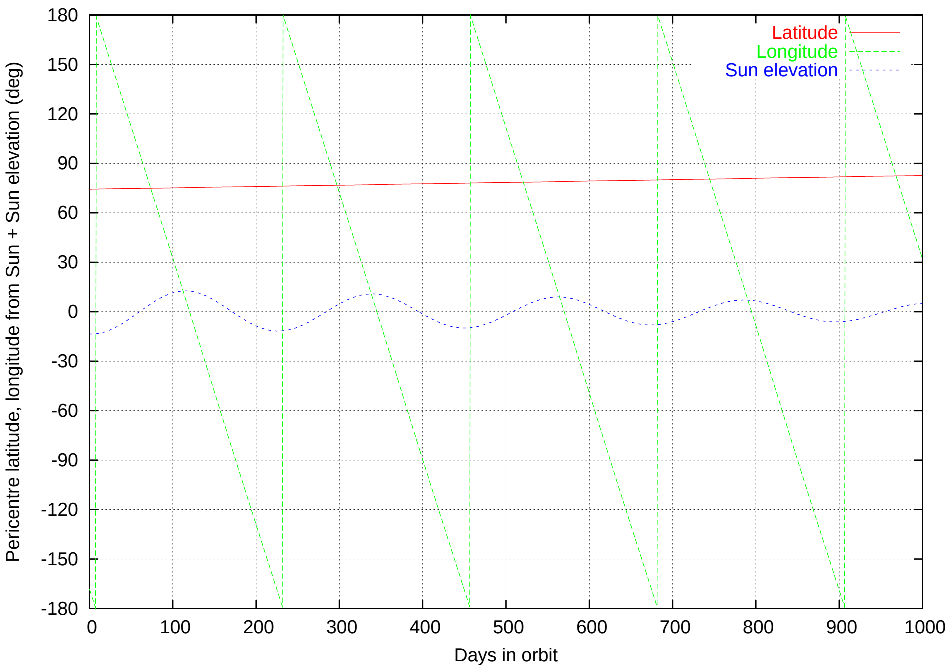

Figure 7-17: Evolution of the pericenter latitude, longitude, and the Sun elevation at the sub-satellite at pericenter over the extended mission lifetime. Launch 24/11/2005

The evolution of the pericenter latitude as a function of the Sun elevation on the sub-satellite point at pericenter is shown from Figure 7-18 to Figure 7-21, for the launch dates as above. The evolution is presented for an extended mission lifetime of 1000 days.

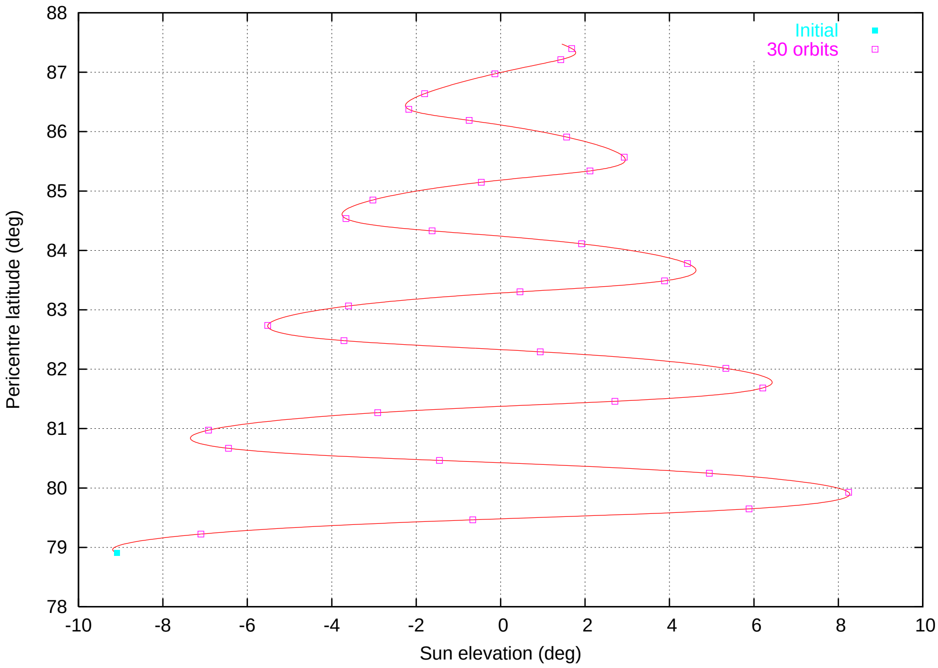

Figure 7-18: Latitude of pericenter vs. Sun elevation at the sub-satellite point. Launch 26/10/2005

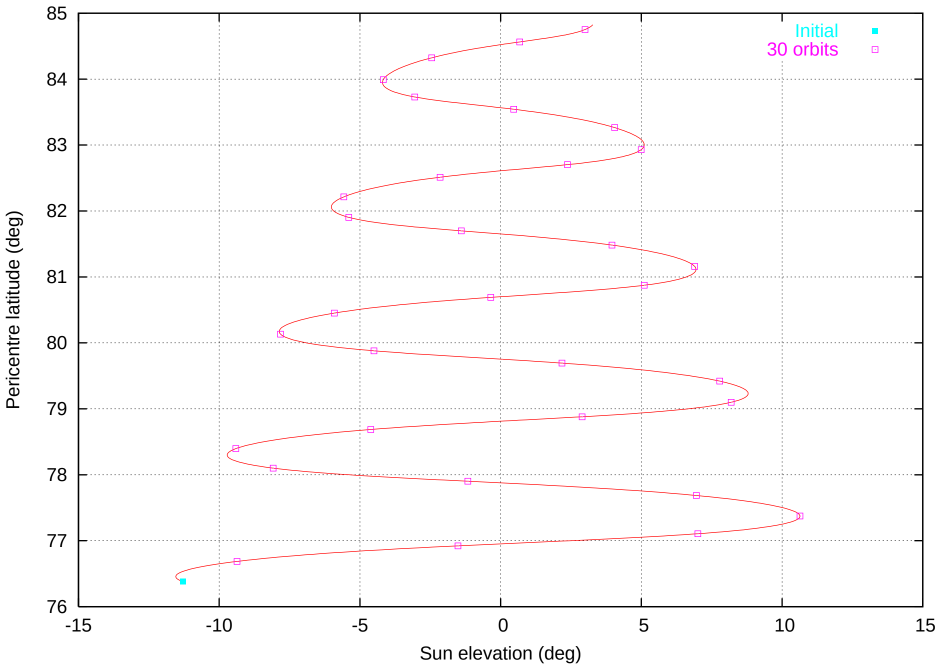

Figure 7-19: Latitude of pericenter vs. Sun elevation at the sub-satellite point. Launch 05/11/2005

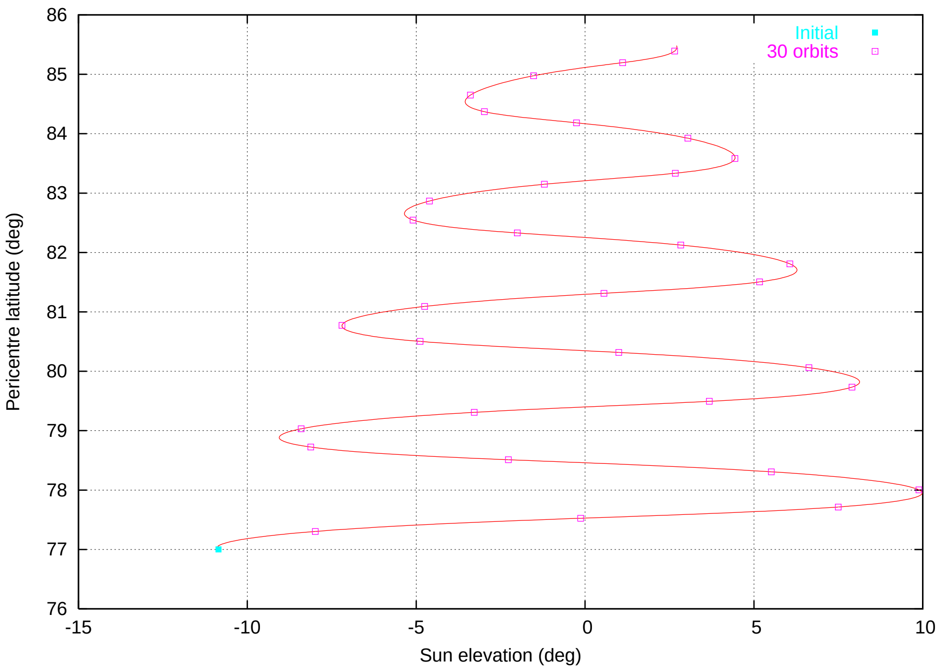

Figure 7-20: Latitude of pericenter vs. Sun elevation at the sub-satellite point. Launch 15/11/2005

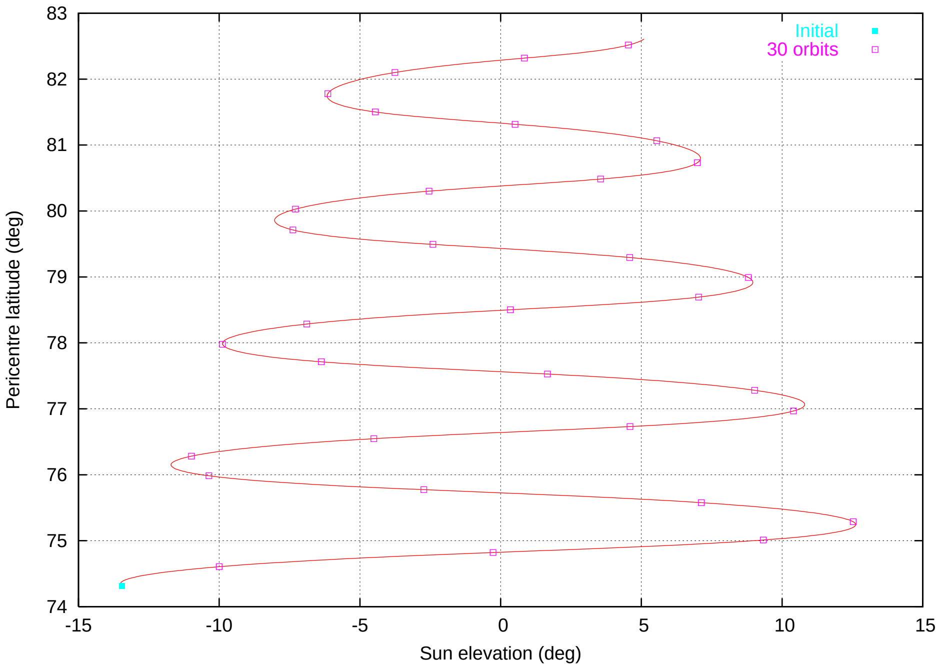

Figure 7-21: Latitude of pericenter vs. Sun elevation at the sub-satellite point. Launch 24/11/2005

The evolution of the angle of the direction to the Sun with the orbital plane of the spacecraft is presented in Figure 7-22. The equivalent angle with respect to the Earth is presented in Figure 7-23. Eclipse and occultation seasons occur when the angle of the Sun w.r.t. the orbit, the angle of the Earth w.r.t. the orbit, respectively, is near zero. The evolution of both angles is presented for an extended mission lifetime of 1000 days.

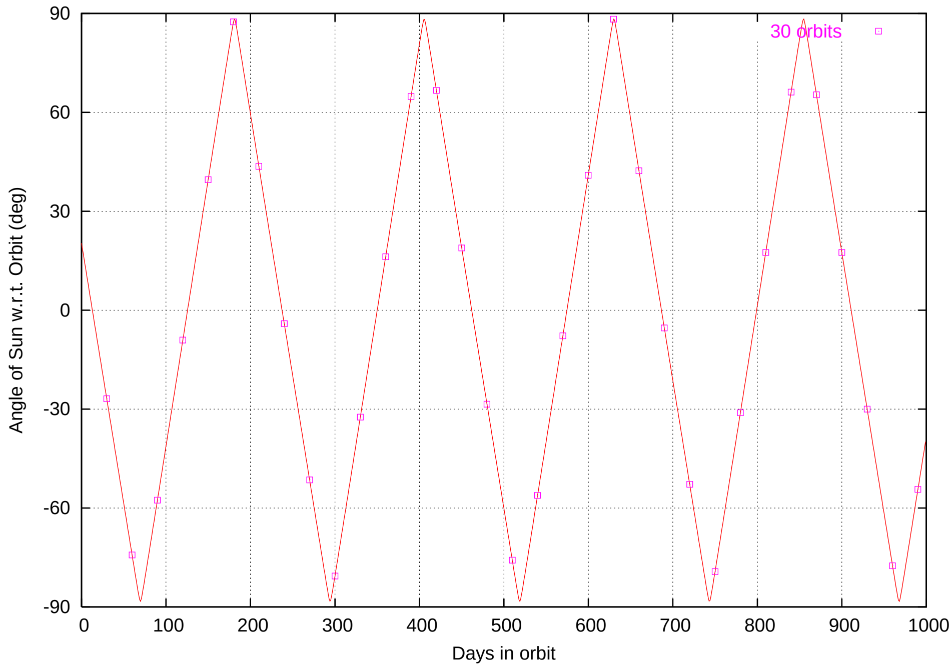

Figure 7-22: Angle of the Sun direction w.r.t. the orbital plane

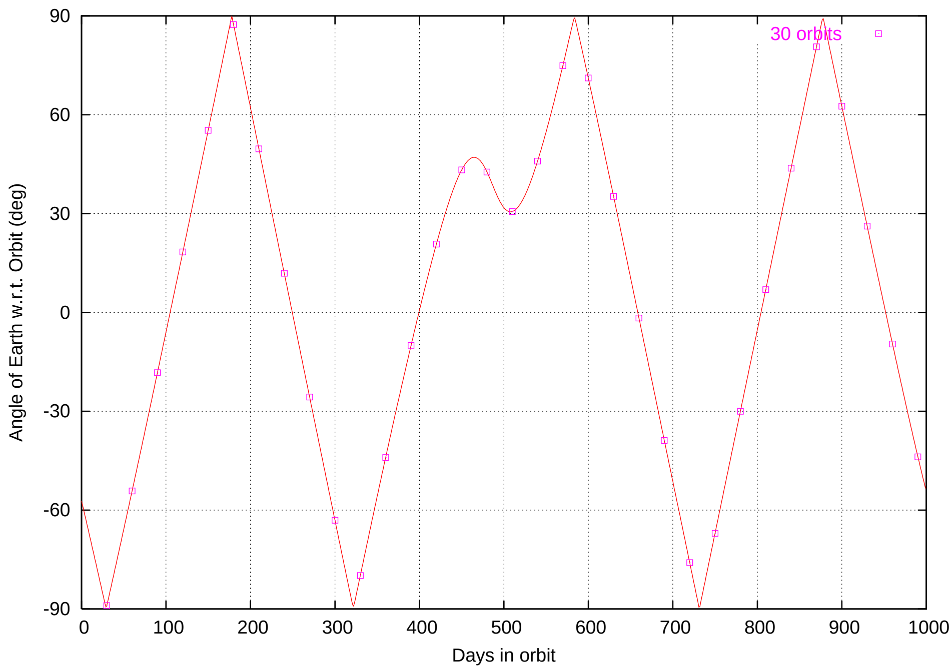

Figure 7-23: Angle of the Earth direction w.r.t. the orbital plane

8. SURFACE COVERAGE

As the orbit is polar (inclination = 90º), and Venus rotates very slowly, the ground track of the sub-satellite points are composed of almost meridional lines Figure 8-1.

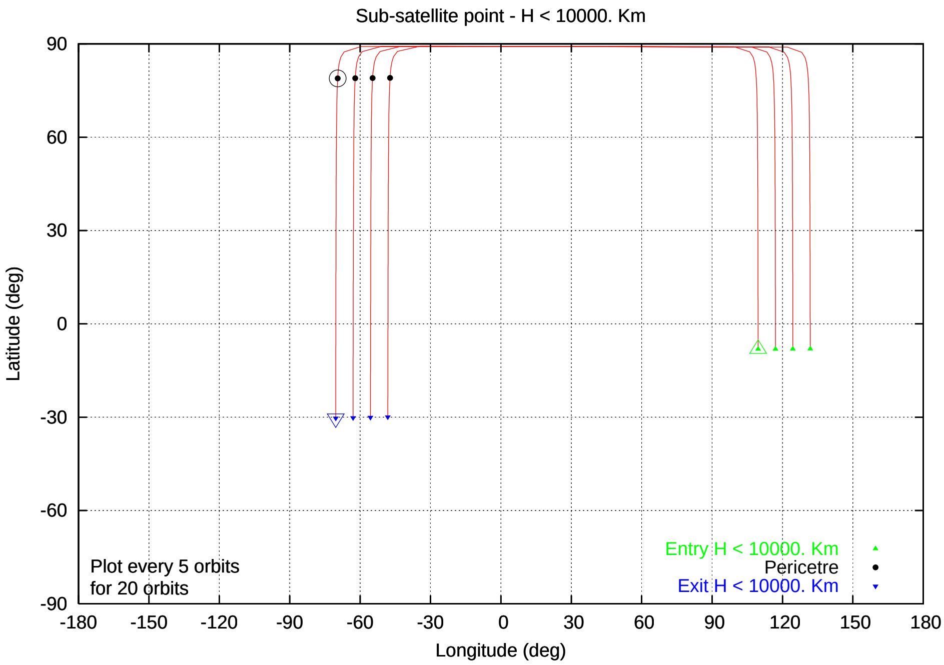

Figure 8-1: Typical sub-satellite ground track for height < 10000 km.

Because the orbit plane is almost fixed in an inertial system of coordinates, and that Venus makes a revolution around the Sun in about 225 days, the ground track is moving in longitude. The evolution of the longitude of the South-North crossing, and of the North-South crossing is shown in Figure 8-2.

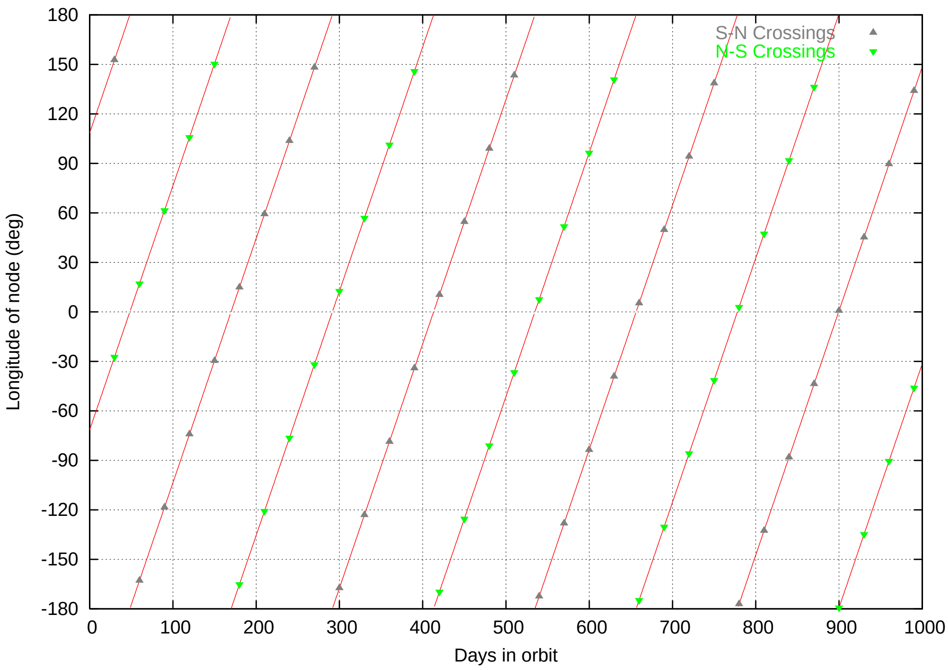

Figure 8-2: Longitude of the S-N and N-S crossing of the ground tracks.

The surface is covered in longitude in about 150 days.

When control is applied to maintain the height of pericenter Figure 8-3 show the coverage of the surface of Venus for height of the satellite less than 10000 km, and Figure 8-4 for height less than 2000 km.

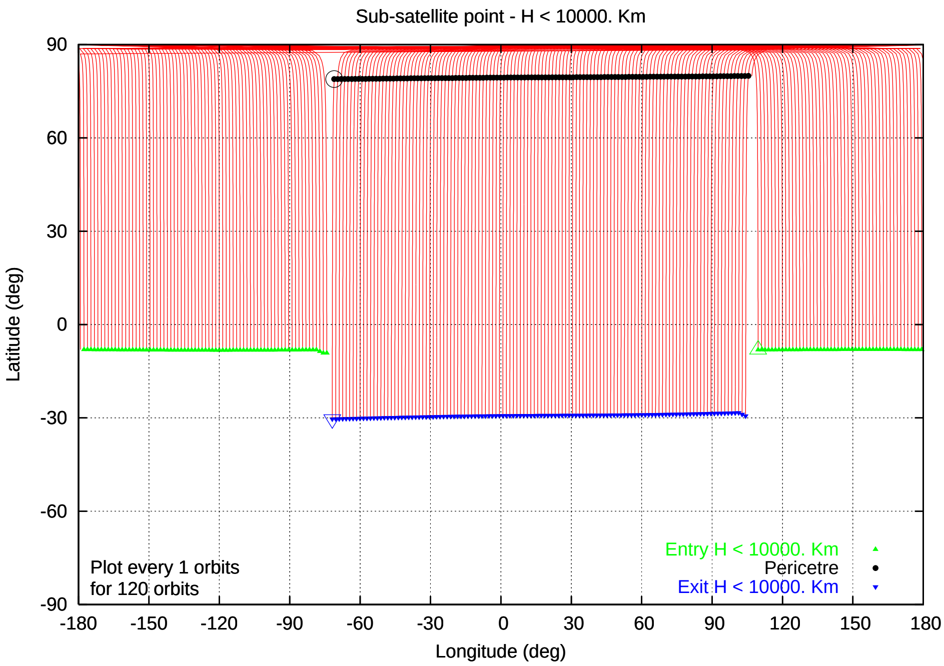

Figure 8-3: Sub-satellite point during first period of pericenter rising for height less than 10000 km

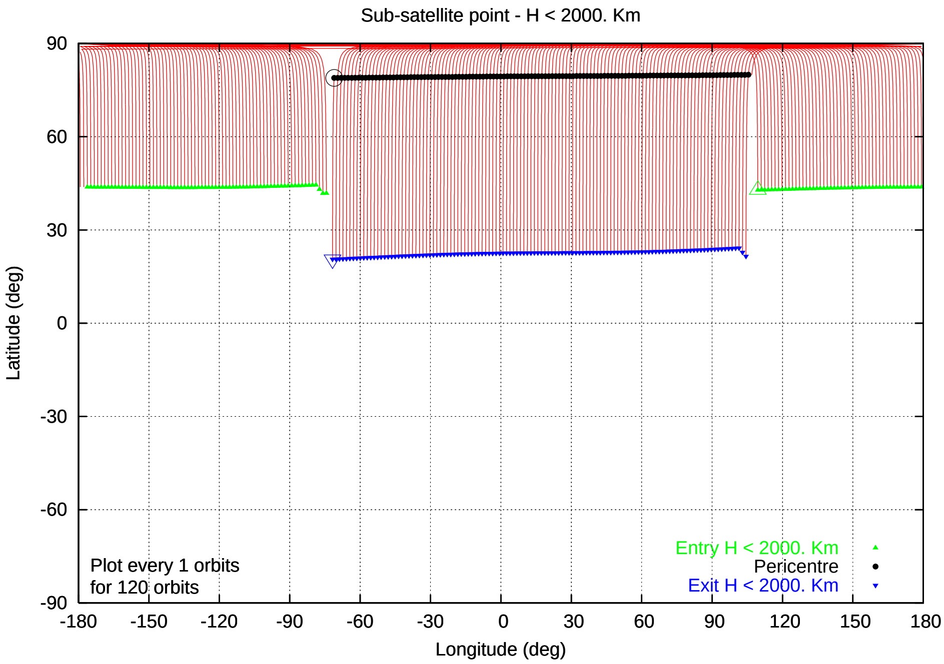

Figure 8-4: Sub-satellite point during first period of pericenter rising for height less than 2000 km

The duration of the observation phase below 2000 Km of altitude for each orbit is presented in Figure 8-5. This parameter is modulated by the change of the pericentre altitude and the strong variations due to the manoeuvres of the pericentre control. The duration of the phase of the orbit above 10000 Km of altitude is around 22.5 hours and its variation for the 1000 days extended lifetime is presented in Figure 8-6.

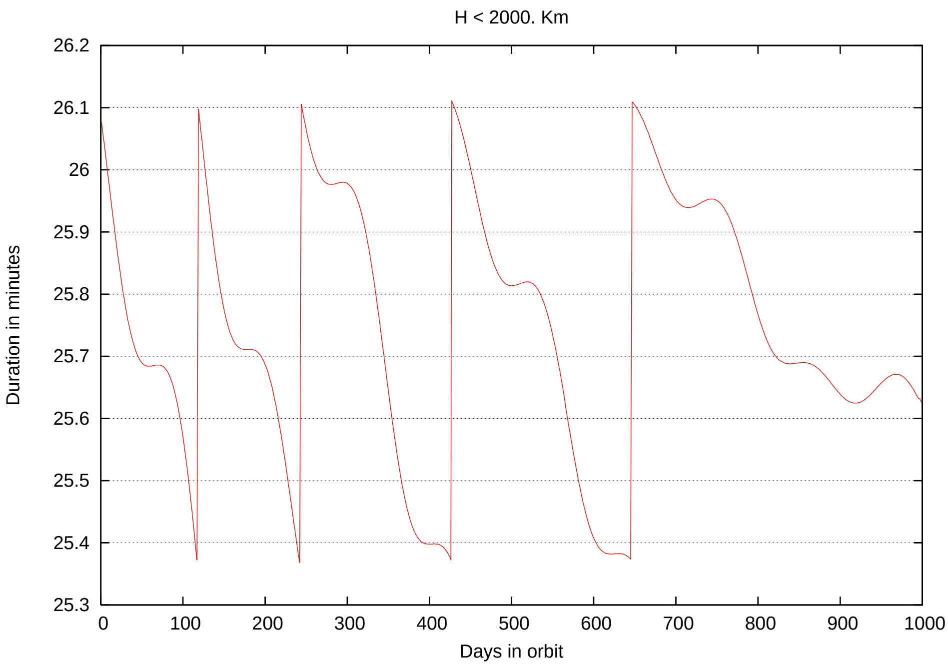

Figure 8-5: Duration of orbit arc with altitude below 2000 Km. Observation phase

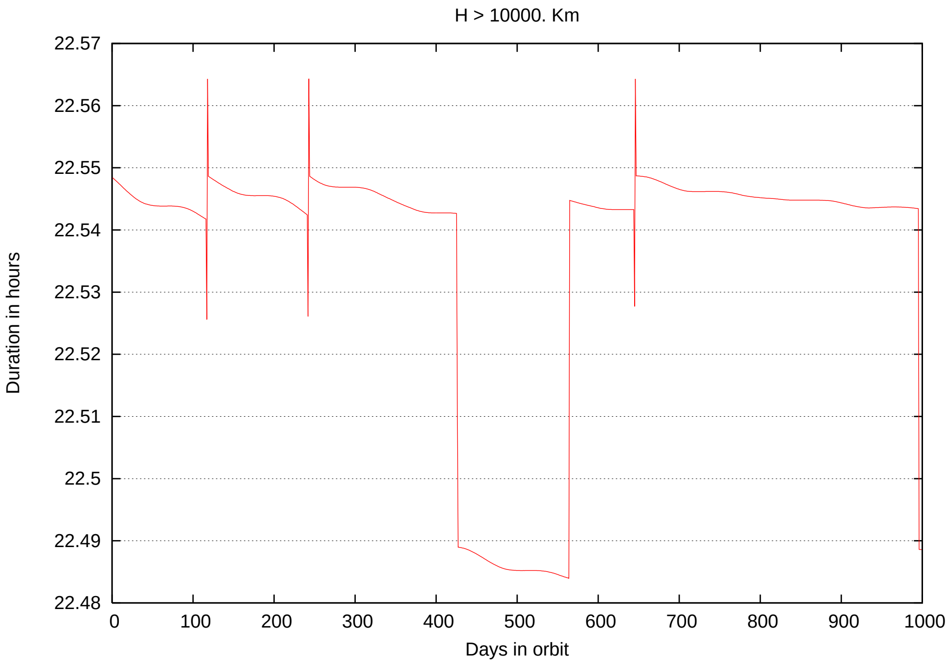

Figure 8-6: Duration of orbit arc with altitude above 10000 Km.

For the observations that will be carried out at altitudes lower than 2000 Km it is important to determine the illumination conditions of the sub-satellite point. Figure 8-7 presents information for the point of each orbit in which the spacecraft enters the observation region. The latitude of this point decreases slightly during the lifetime. The longitude of the spacecraft w.r.t. the Sun direction gives at least four revolutions. The elevation of the Sun at the sub-satellite point determines whether the observed point is at day –positive elevations- or at night –negative elevations. Periods of night/day of about 110 days can be observed. The same information for the end of the observation phase is presented in Figure 8-8.

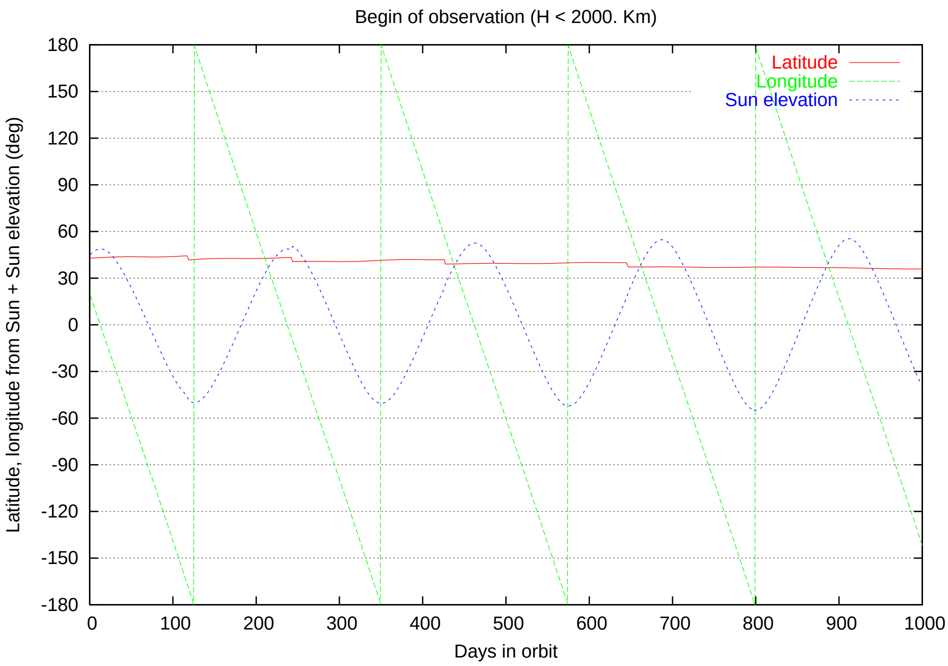

Figure 8-7: Longitude, latitude and elevation of the Sun at the sub-satellite point for the beginning of the observation phase.

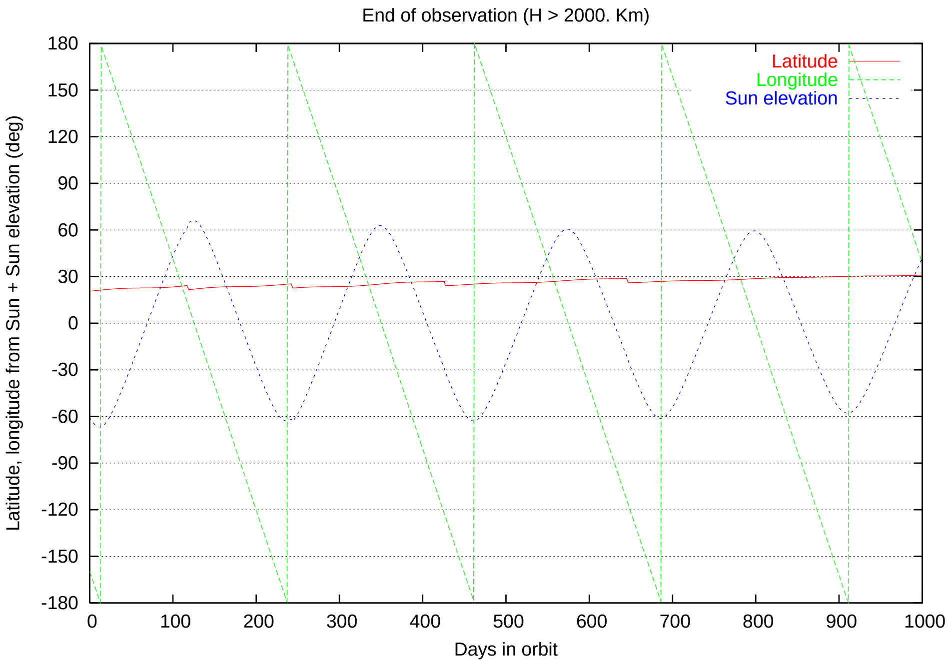

Figure 8-8: Longitude, latitude and elevation of the Sun at the sub-satellite point for the end of the observation phase.

The illumination conditions for each observation period are presented in Figure 8-9. The vertical axis represents the time measured from the pericentre and the dark areas are the periods for which the observed zone is at night. The duration of the observation at night or day is presented in Figure 8-10.

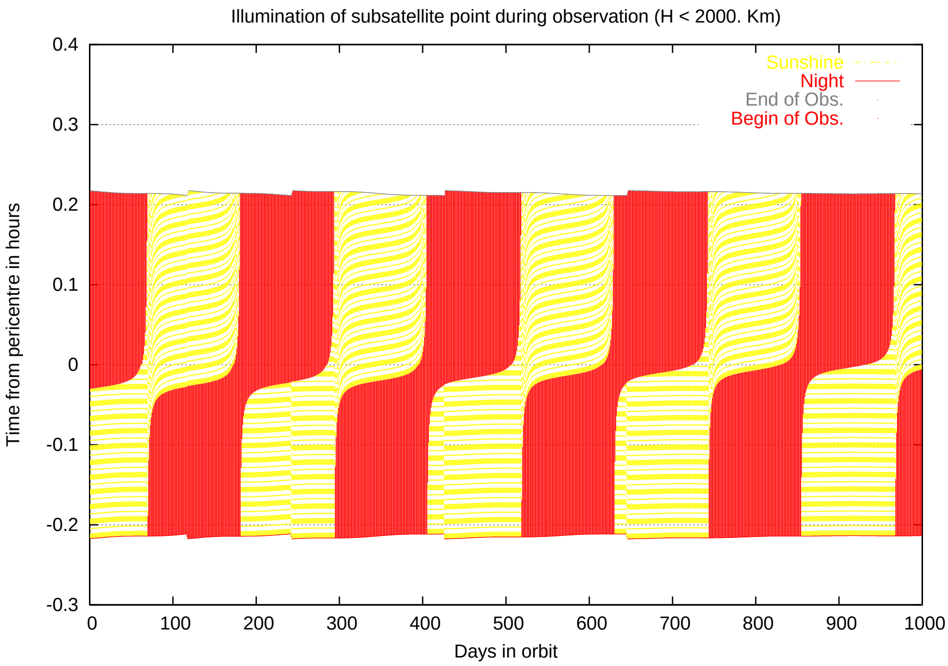

Figure 8-9: Illumination conditions day-night of the sub-satellite points during the observation phase. Time measured from pericentre

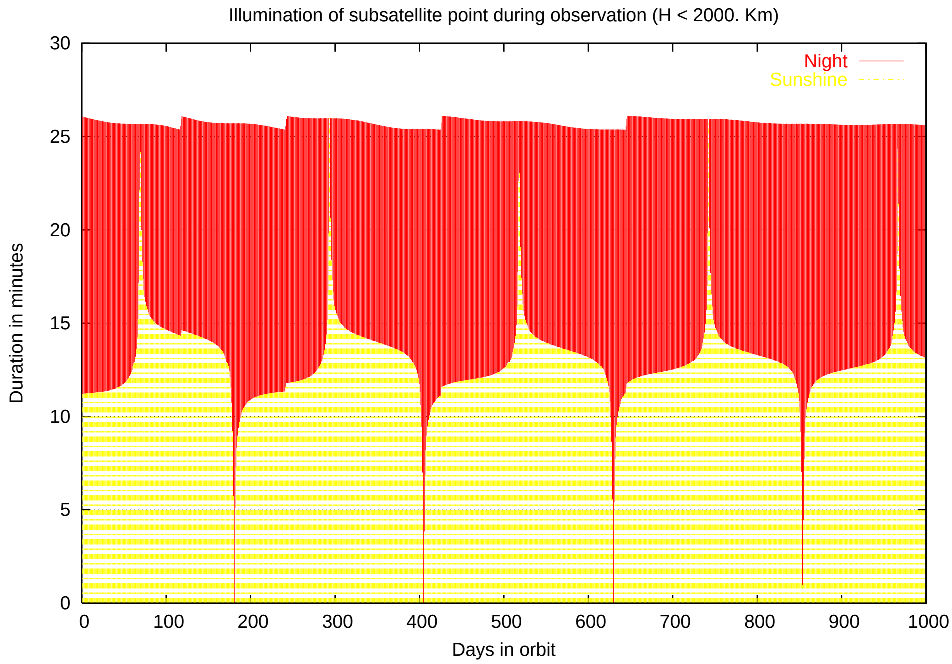

Figure 8-10: Duration of the day-night of the sub-satellite points for the observation phase

Figure 8-11 presents the coverage of the spacecraft at the beginning of the mission indicating the regions at which the observed sub-satellite point is at day or night. The day/night regions evolve with the apparent movement of the Sun on Venus surface.

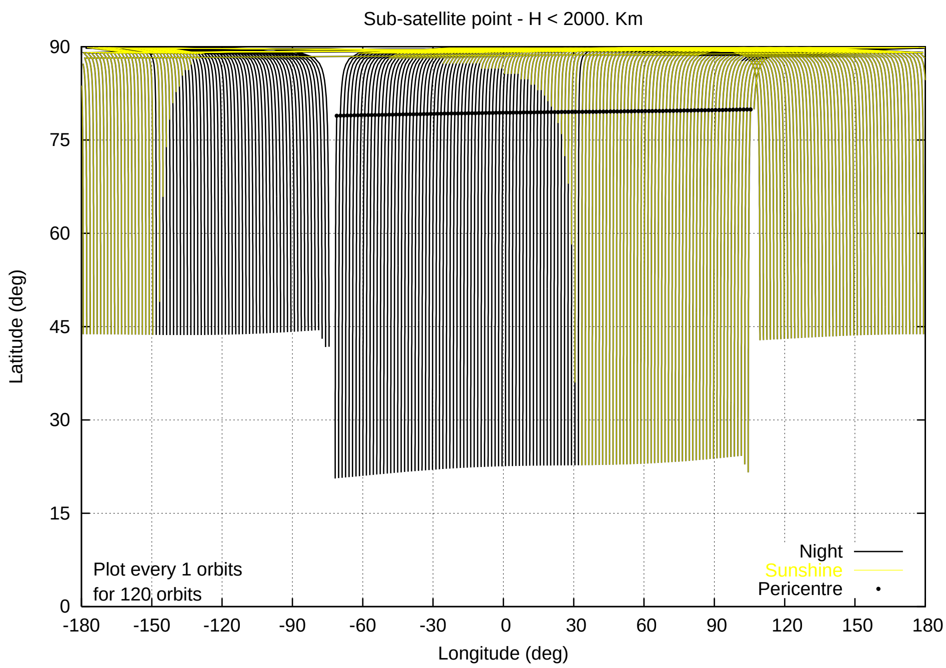

Figure 8-11: Ground tracks for height below 2000 Km with illumination conditions. Northern hemisphere

9. ECLIPSES AND EARTH OCCULTATION

During the extended mission lifetime there are nine seasons of eclipse, the maximum eclipse duration is 50 to 65 min, depending on the launch day. The eclipses after pericenter are shorter that the ones before pericenter, but the number of consecutive orbits with eclipse is greater, Figure 9-1 and Figure 9-2. The first season of eclipses start at BOM with eclipses just past pericenter. The next season has eclipses just before pericenter, Figure 9-1, and Figure 9-2. Due to the change in argument of pericenter drift towards the North Pole. As a result, the duration of the eclipses before and after pericenter becomes almost the same. Figure 9-3 shows the UTC time of the eclipses when control is applied to keep the visibility phasing from Cebreros.

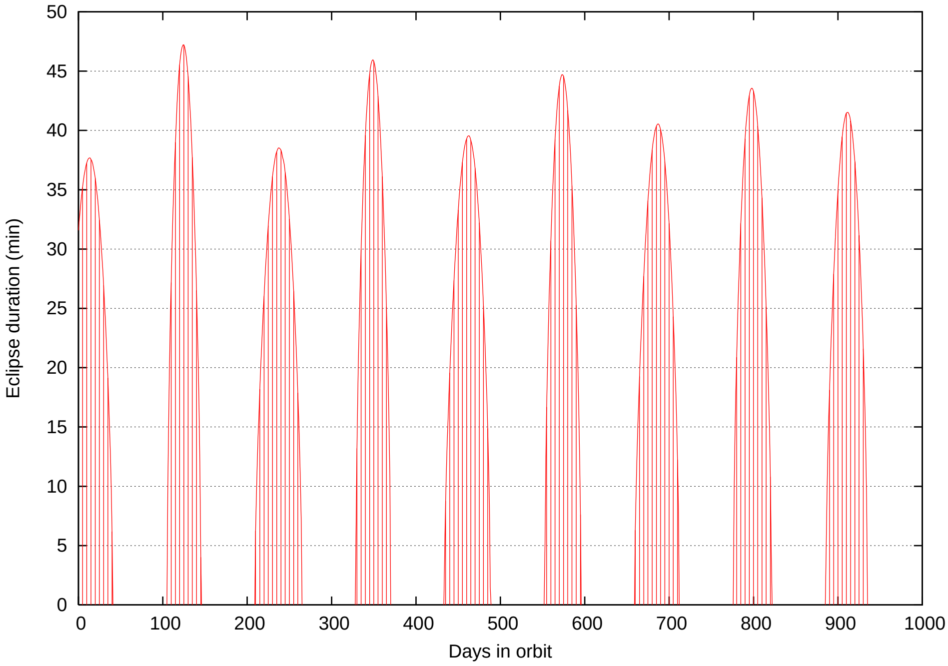

Figure 9-1: Eclipse duration along the mission of 1000 days for a launch on 26/10/2005. Duration is given in minutes.

The VeRa, radio science experiment, will study the atmosphere of Venus during the periods where Venus occults the Earth as seen from the spacecraft. Figure 9-4 and Figure 9-5 shows the period of occultation and the position of the Earth occultation with respect to pericenter. There are six seasons of Earth occultation with duration between 1 to 2.5 months. Each occultation last slightly longer than the eclipses.

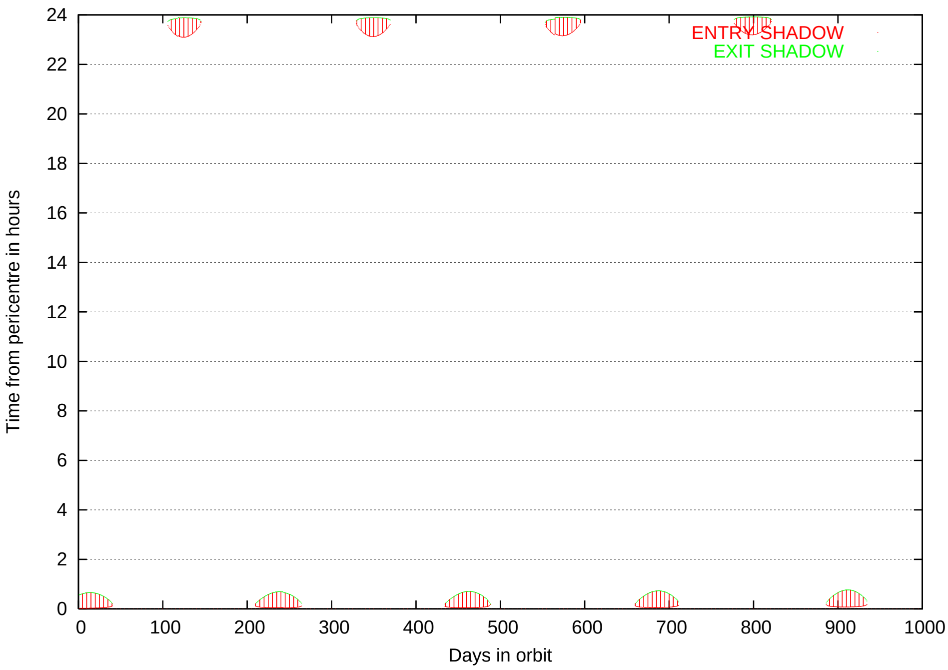

Figure 9-2: Eclipse position along the orbit. Time is measured from pericenter.

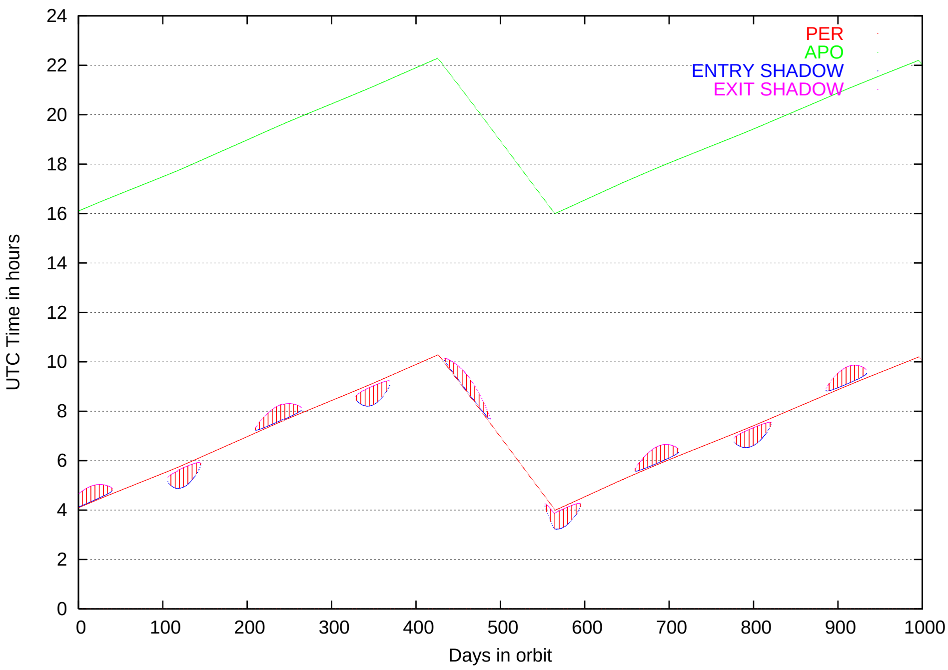

Figure 9-3: Eclipse positions along the day with control of the visibility phasing from Cebreros. Time is UTC

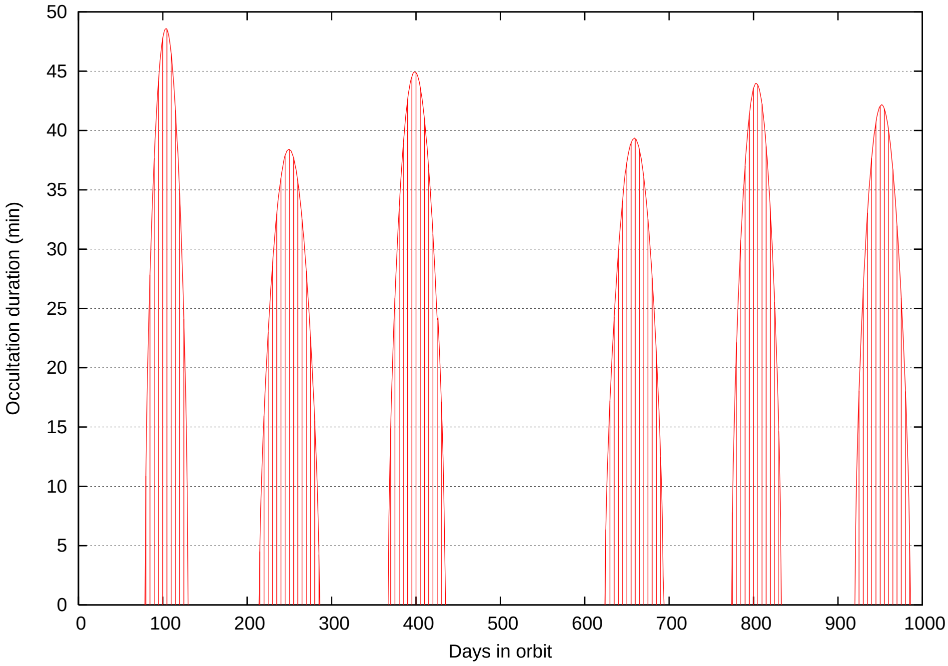

Figure 9-4: Periods of Earth occultation as seen from the spacecraft.

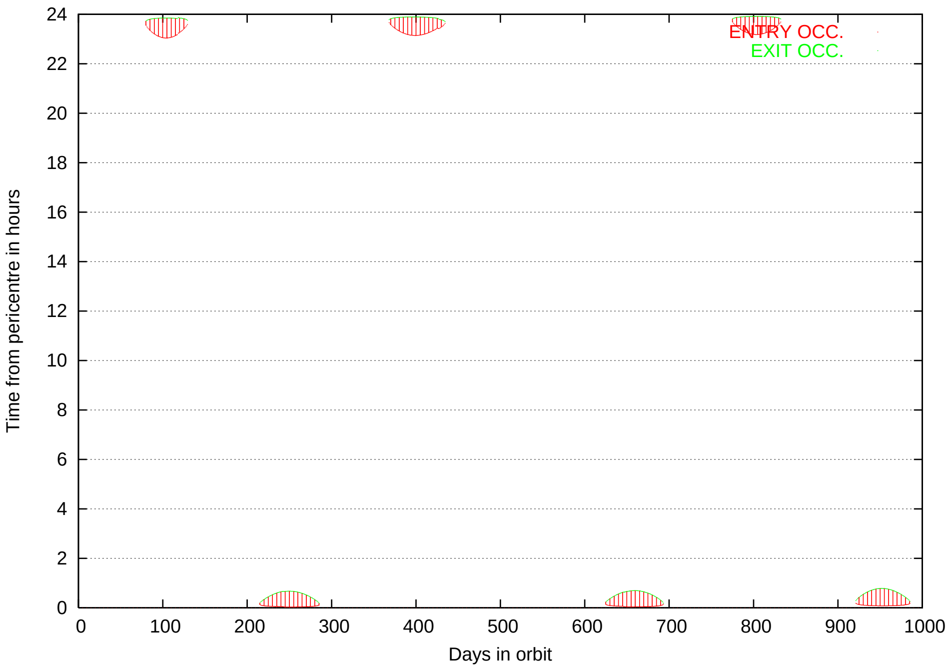

Figure 9-5: Earth occultation position along the orbit. Time is measured from pericentre

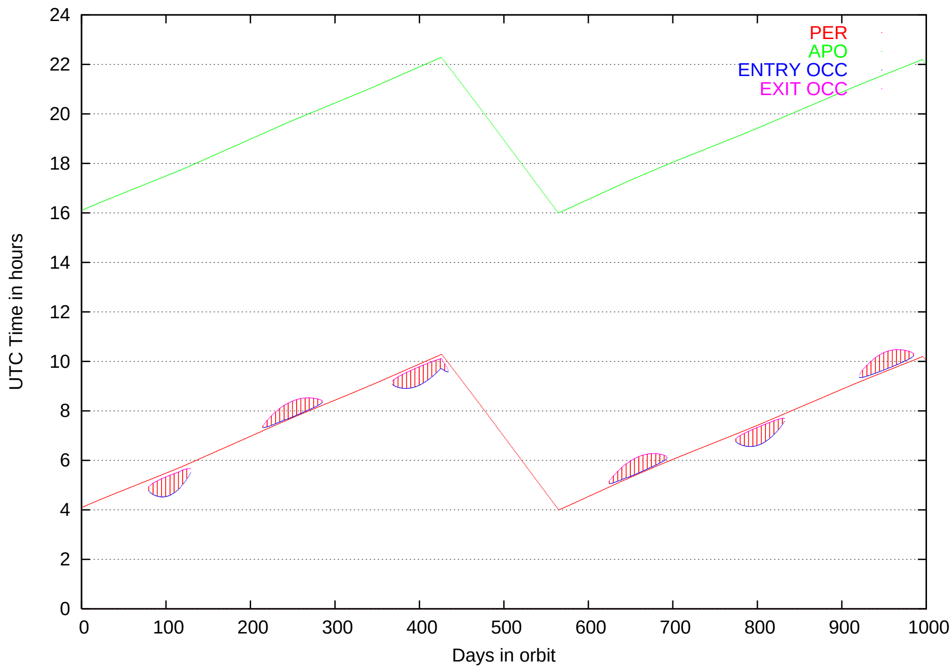

Figure 9-6: Earth occultation position along the day with control of the visibility phasing from Cebreros. Time is UTC

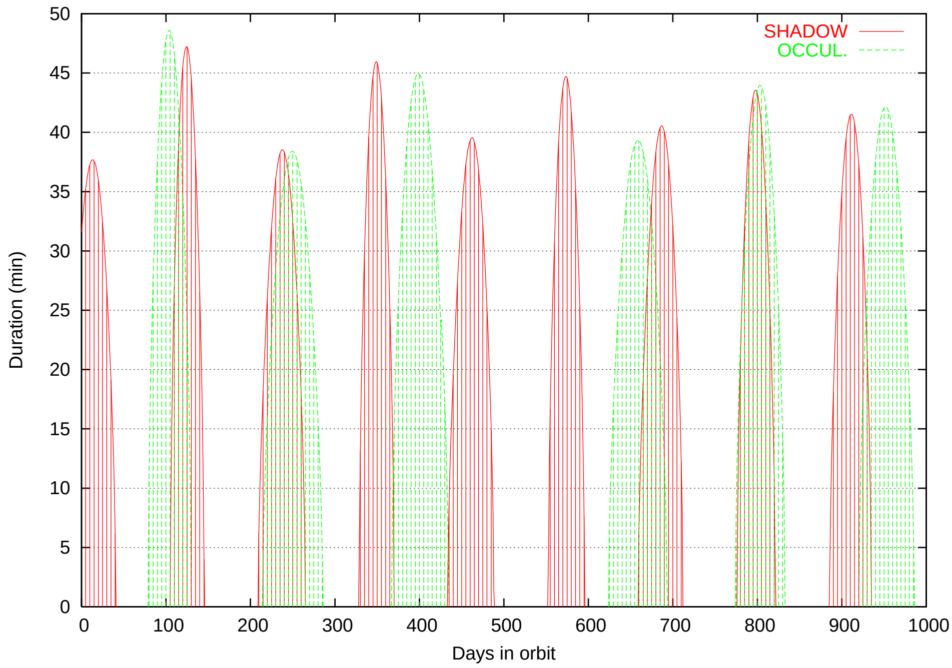

Figure 9-7: Periods of Eclipse and Earth occultation as seen from the spacecraft.

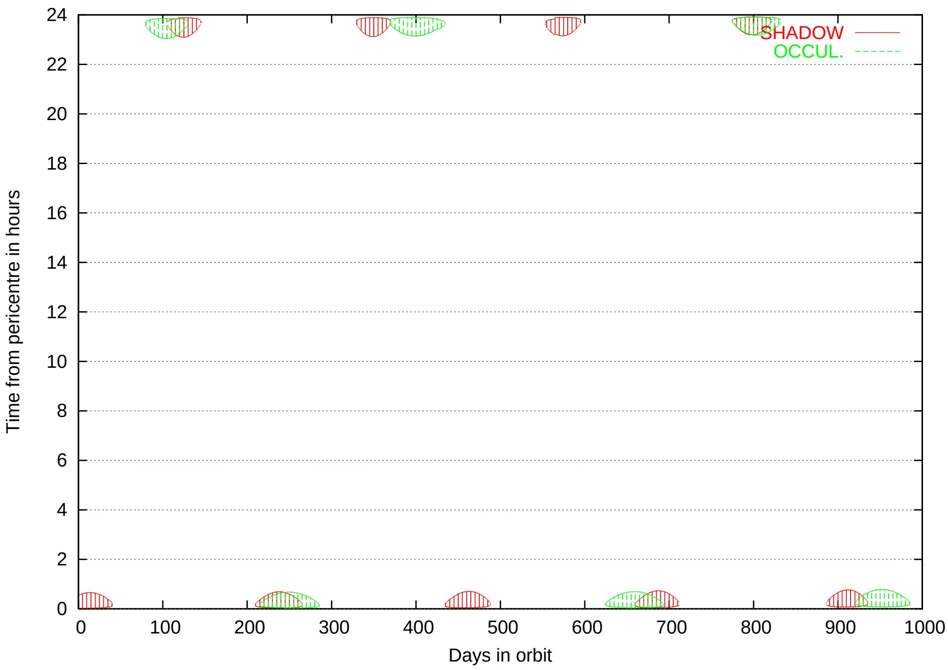

Figure 9-8: Eclipse and Earth occultation position along the orbit.

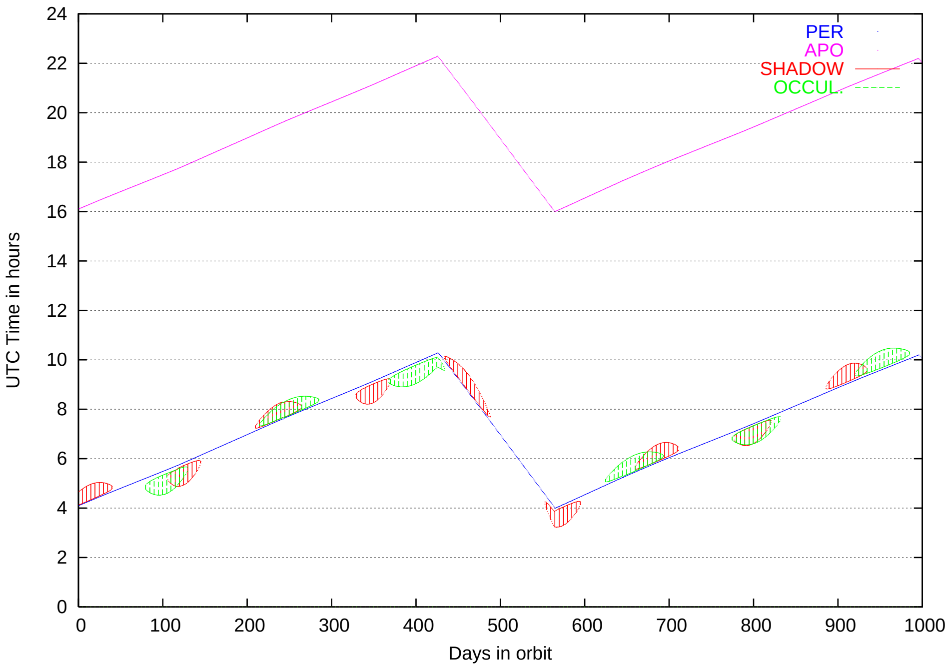

Figure 9-9: Eclipse and Earth occultation position along the day with control of the visibility phasing from Cebreros. Time is UTC

Figure 9-6 shows the UTC time of the Earth occultations when control is applied to keep the visibility phasing from Cebreros. Figure 9-7,Figure 9-8, and Figure 9-9 present the combination of eclipses and Earth occultations.

10. GROUND STATION COVERAGE

Cebreros will be the main ground station for Venus Express with New Norcia used for the periods of Earth occultation. Data transmission to Earth will be when the spacecraft is near apocenter while the Earth occultation periods are when the spacecraft is near pericenter.

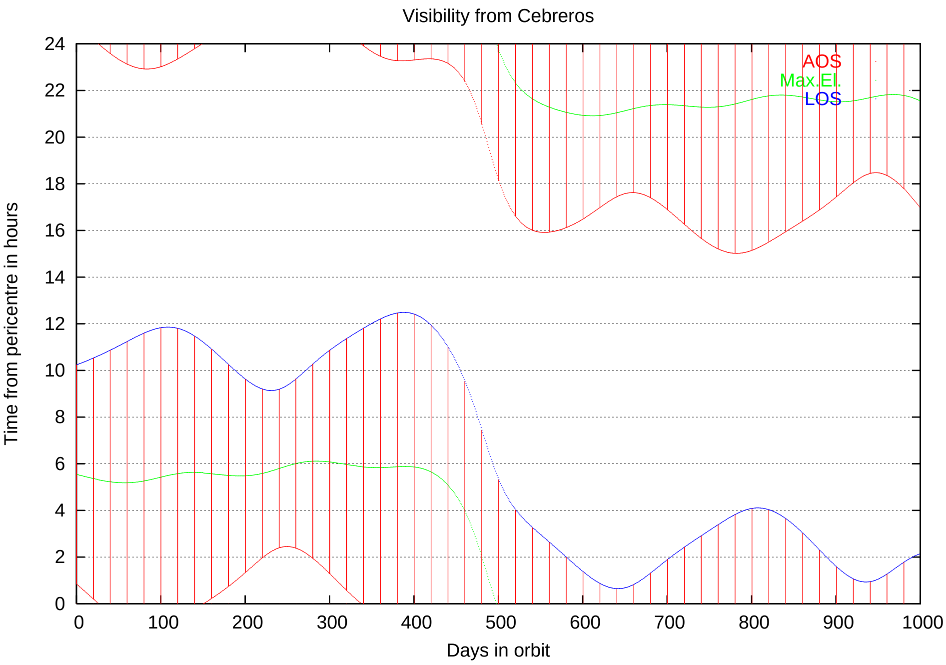

Figure 10-1: Visibility from Cebreros. No control of the visibility phasing. Time from pericenter

The orbital period of about 24 h, and the time of pericenter pass can be selected such that the contact with Cebreros ground station is when the satellite is near apocenter. Figure 10-1 shows the period of visibility as function of time from pericenter. For each day, the black region gives the time when the satellite is visible from Cebreros with an elevation of more than 5º. After about 14 months, the synodical motion of Venus induces a swap of the visibility period and the period of contact with Cebreros will be near pericenter. In the period of fast change of the time of contact, May to Aug. 2007, to maintain the Cebreros pass near the apocenter, the orbital period must be changed by about 150 s. This is equivalent to a change of the semi-major axis of the orbit of less than 50 km. The pericenter manoeuvre to do it is less than 0.5 m/s to initiate the correction and 0.5 m/s to stop it.

This can be easily controlled, and in fact, the influence of the AOCS wheel unloading manoeuvres will have a greater effect. However, it is possible to combine them with the pericenter maintenance manoeuvres, and with any manoeuvre needed to correct the phasing of the spacecraft in its orbit with respect to the period of visibility of the ground stations.

Figure 10-2 shows the same visibility information for New Norcia. The minimum elevation for this ground station is 10º. New Norcia will contact the spacecraft near pericenter, and, therefore, will be able to support the observation of VeRa during the periods of Earth occultation.

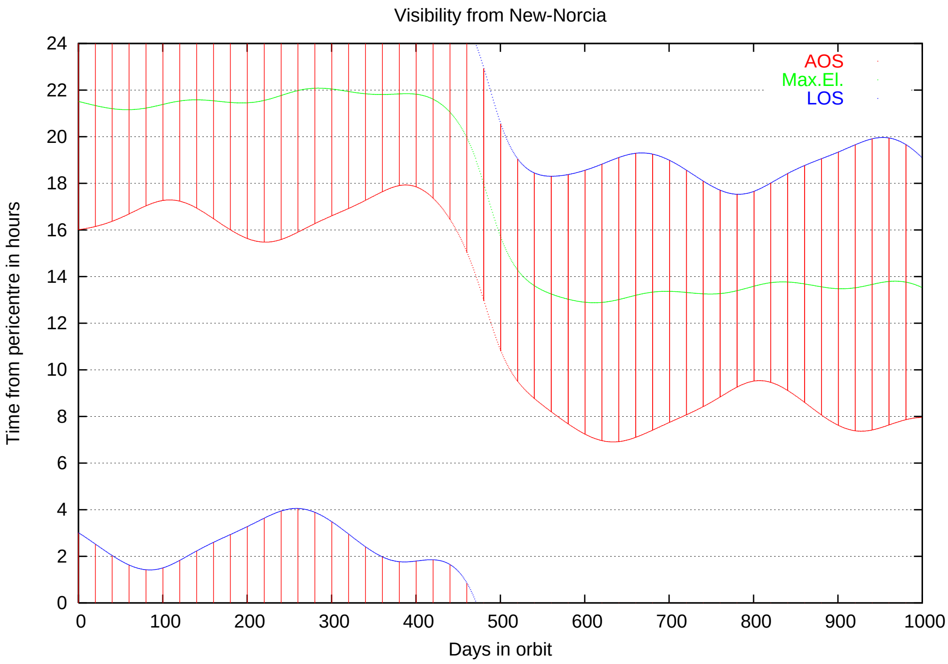

Figure 10-2: Visibility from New Norcia. No control of the visibility phasing. Time from pericenter

The orbital period of the spacecraft can be controlled to follow the natural drift of the visibility of Venus from Cebreros, see Section 6.3. The whole lifetime can be divided in different phases for which a nominal period of the operational orbit is defined. An extended lifetime of 1000 days will have at least three of these phases. In the first and third phases, there is only a small positive drift of about 50 seconds/day. For the second, the drift is negative and over 150 seconds/day. A fourth phase of negative drift would be short after the day 1000. Given the nominal period of a phase, the corresponding semi-major axis of the orbit can be obtained. The perturbation due to the third body effect of the Sun induces a variation of the orbital period from pericenter to pericenter of about 20 seconds; therefore this perturbation must be taken into account for the calculation of the nominal semi-major axis of each of the phases. The following table summarizes the phases, nominal periods and semi-major axis for a launch at 26/10/2005.

	Duration (days)	Orbital Period (seconds)	Semi-major axis (Km)
Phase 1	419	86452.4	39476.0
Phase 2	146	86240.4	39411.4
Phase 3	435	86454.2	39476.6

Table 16: Parameters of the phases defined to follow visibility phasing from Cebreros

Figure 10-3 shows the visibility periods from Cebreros when the AOS is constrained to occur within 2 hours after the pericenter pass. The instant of maximum visibility for each day has a smooth variation except for the phase of the negative natural drift and is contained between 4 and 6 hours after the pericenter. Figure 10-4 shows the same visibility periods but with respect to the UTC hour and Figure 10-5 shows the duration of each visibility. There are three seasons with a minimum duration of about 9 hours.

Figure 10-6, Figure 10-7 and Figure 10-8 show the visibility periods and their duration from New Norcia.

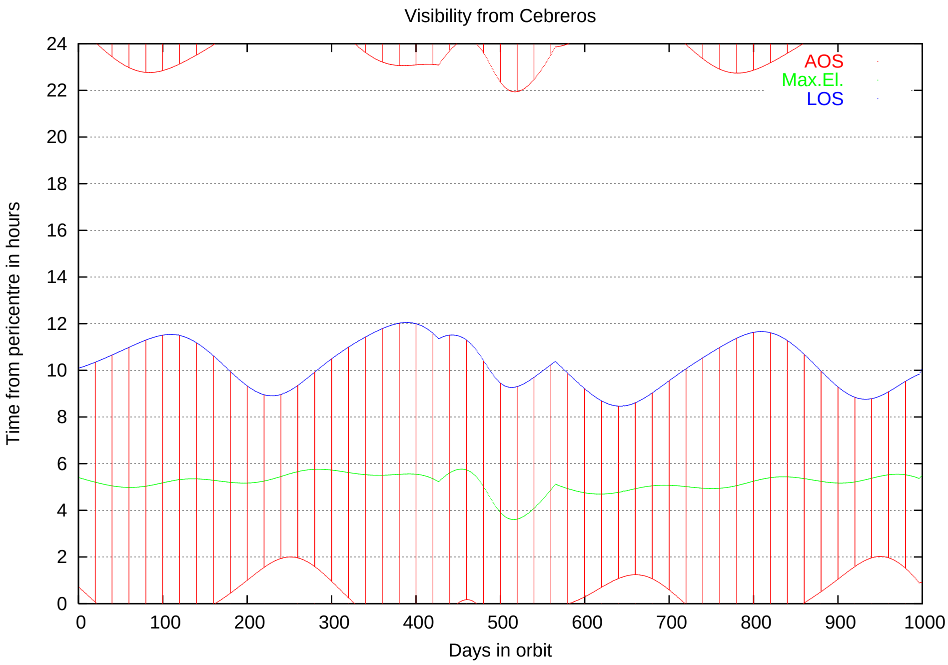

Figure 10-3: Visibility from Cebreros. Control of visibility phasing. Time measured from pericenter

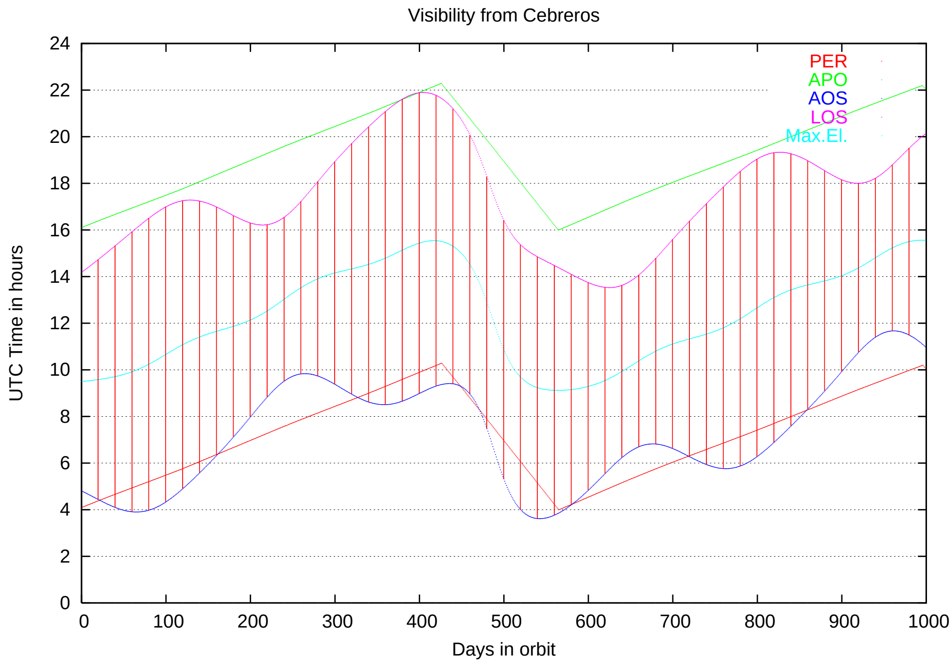

Figure 10-4: Visibility from Cebreros. Control of visibility phasing

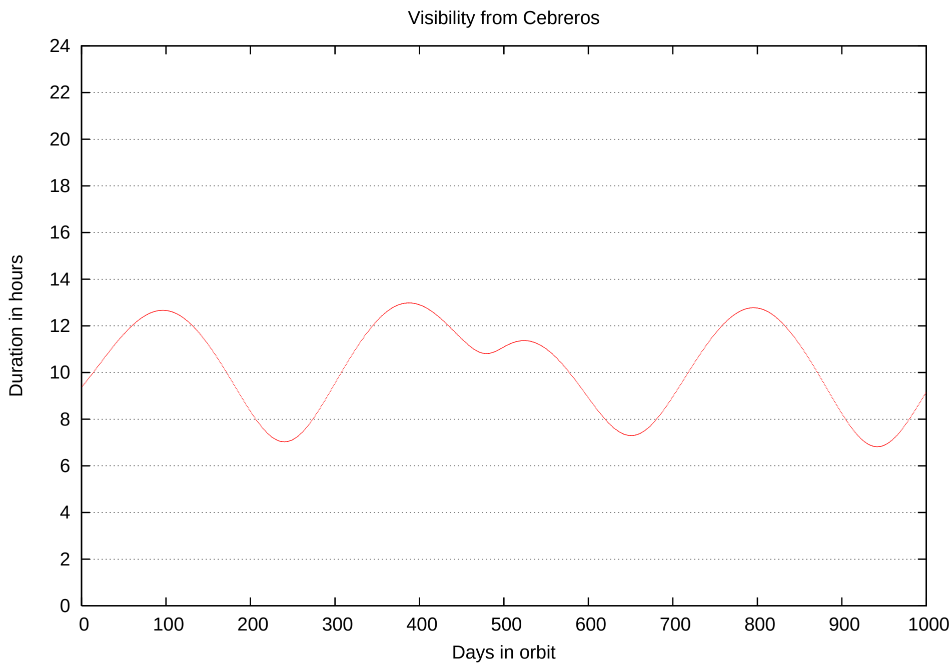

Figure 10-5: Duration of the visibility periods from Cebreros. Control of visibility phasing

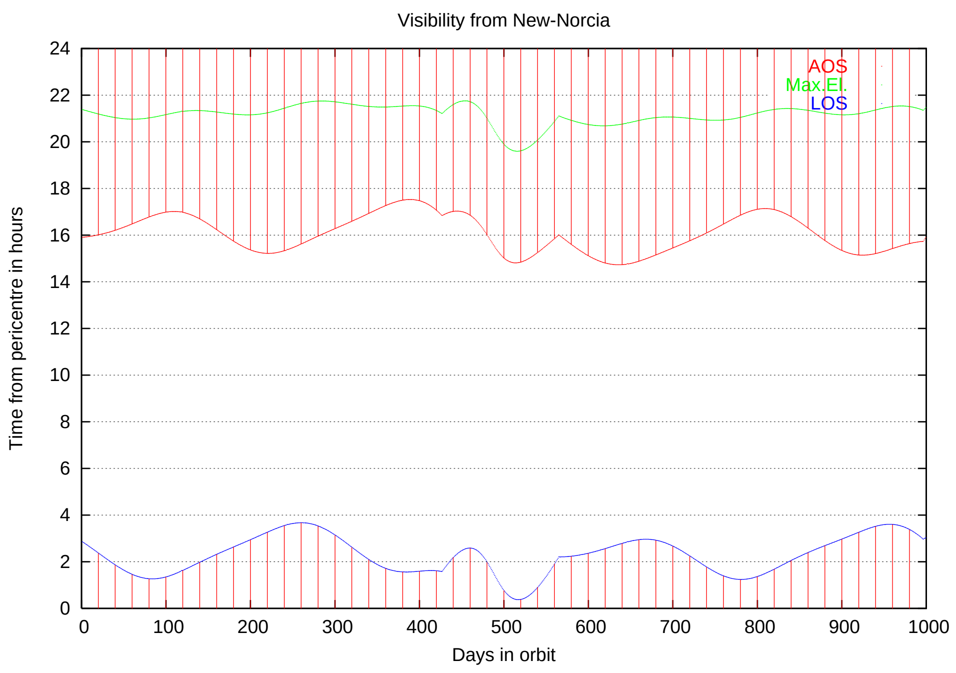

Figure 10-6: Visibility from New Norcia. Control of visibility phasing. Time measured from pericenter

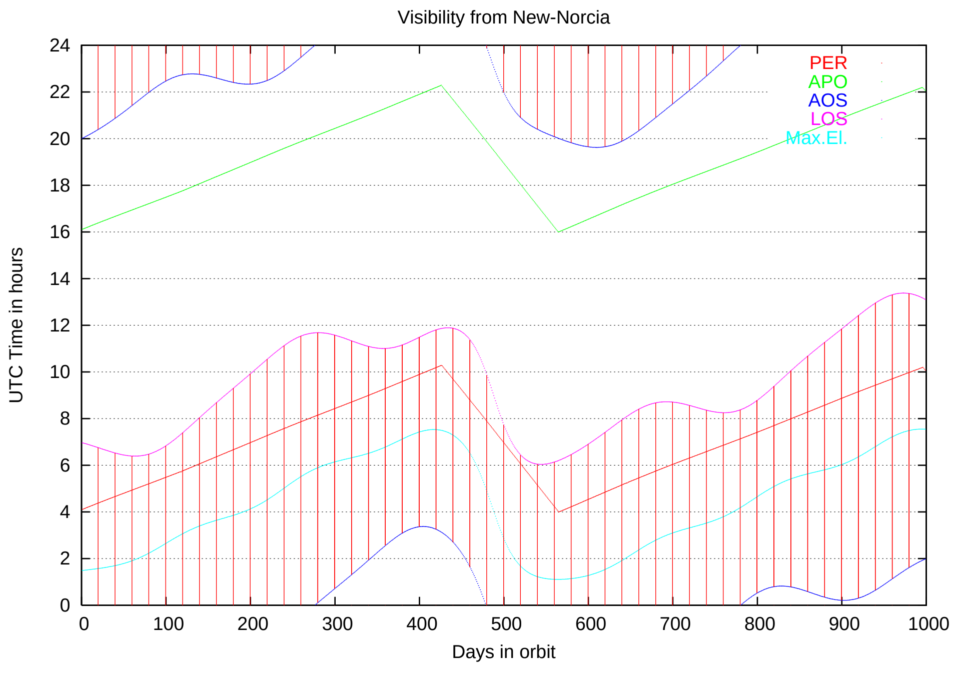

Figure 10-7: Visibility from New Norcia. Control of visibility phasing

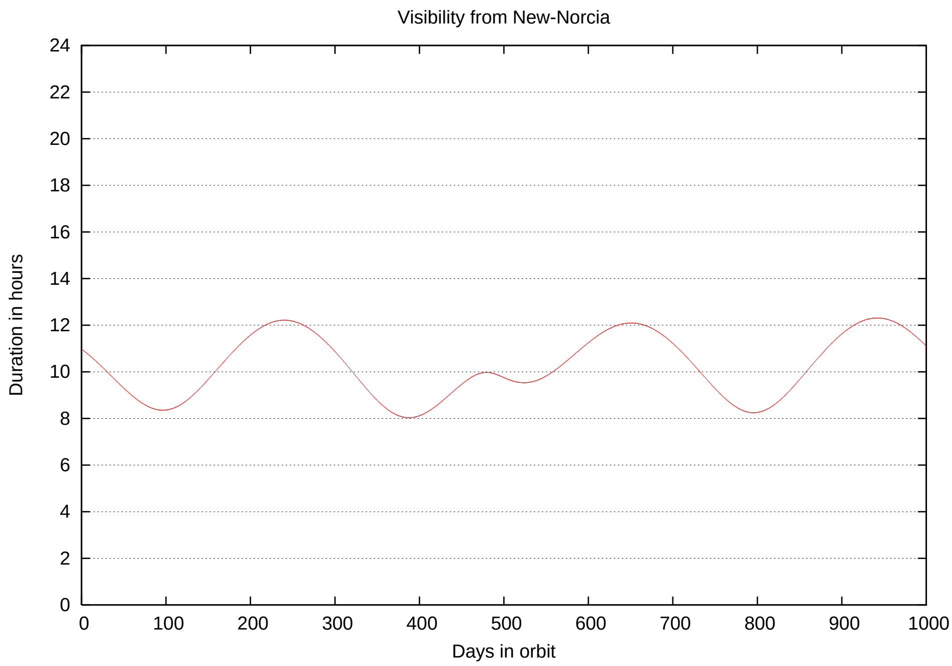

Figure 10-8: Duration of the visibility periods from New Norcia. Control of visibility phasing

The spacecraft will download the scientific data during the visibility periods from Cebreros. The maximum amount of power from the solar arrays is required for this task and the presence of eclipses is not desired. Figure 10-9 presents together the periods of visibility from Cebreros and the eclipses. For most of the days the transmission can start as soon as the observation has finished, for the three seasons of reduced visibility the transmission must wait until the AOS and only for two of the eclipse seasons after the pericenter the transmission must wait until the end of the eclipse.

For the VeRa experiment it is required that the line of sight of the spacecraft to the Earth disappears behind Venus limb while in visibility from a ground station. The pericenter is in visibility from New Norcia and this ground station can be used for this experiment during the Earth occultation seasons. Figure 10-10 presents together the periods of visibility from New Norcia and the Earth occultations.

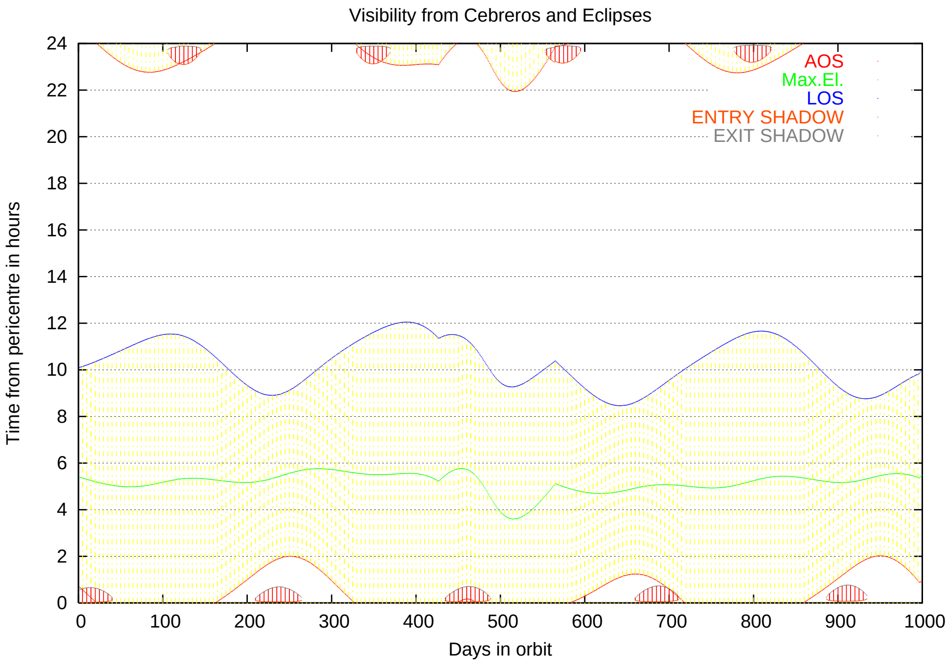

Figure 10-9: Visibility from Cebreros and periods of Eclipse

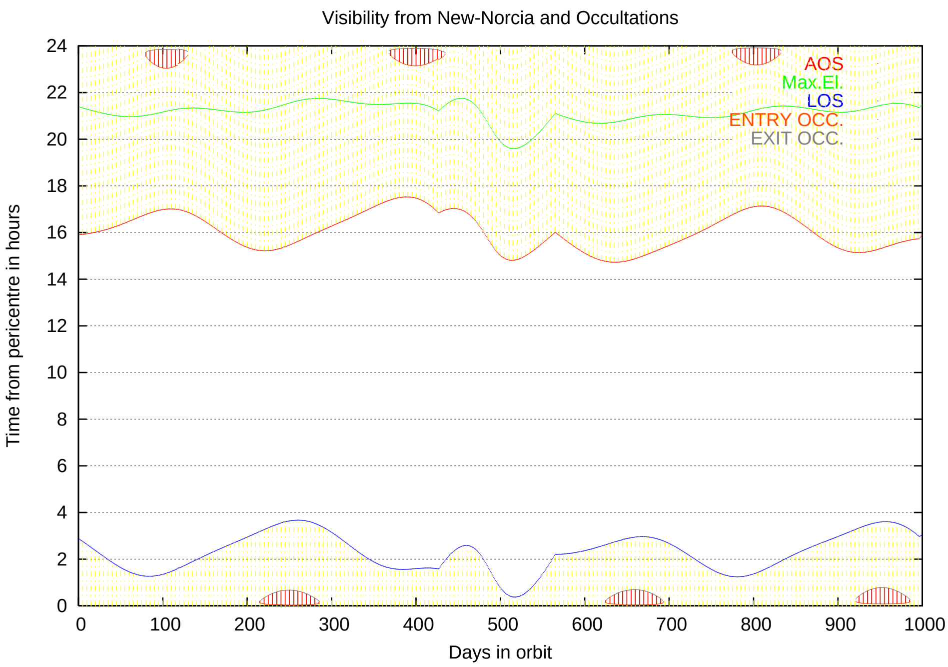

Figure 10-10: Visibility from New Norcia and periods of Earth Occultation

11. Orbit Determination and Navigation

11.1 Early Orbit Determination. First Correction Manoeuvre

Shortly after Venus Express is inserted into the escape trajectory the orbit determination campaign of the LEOP will start. The ground station of New Norcia will be the first to track the spacecraft about 30 minutes after the injection. Cebreros and Kourou will track the spacecraft afterwards and after a short coverage gap, the spacecraft will enter again in the visibility of New Norcia. The sequence of coverage will be repeated during the duration of the LEOP.

During the first visibility pass from New Norcia the observed relative velocity will decrease from about 5.0 Km/s to 4.0 Km/s. As the spacecraft escapes from the Earth, the relative velocity tends to the hyperbolic velocity. Figure 11-1 shows the evolution of the range rate of the spacecraft from the ground stations New Norcia and Cebreros.

DOPPLER MEASUREMENTS PLOT

VENUS EXPRESS LEOP

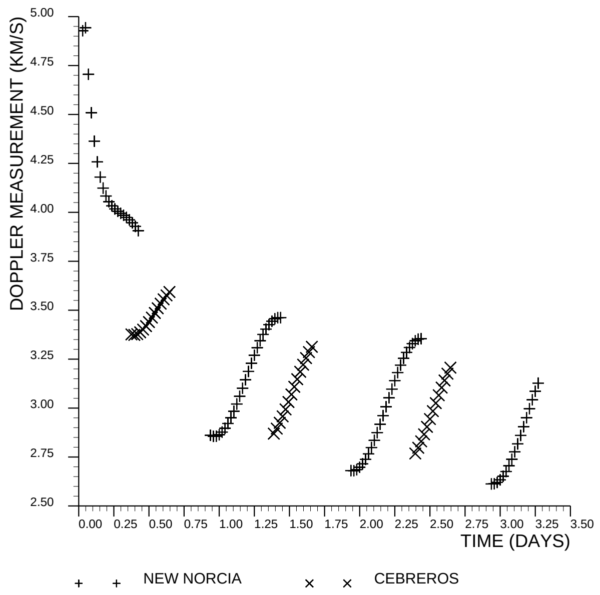

Figure 11-1: Early Orbit Trajectory. Range Rate Measurements from New Norcia and Cebreros

In the first acquisition of the spacecraft from New Norcia ground station, the injection dispersion errors will play a major role. A bad accurate injection could displace the trajectory from the beam of the antenna and force to search the spacecraft with the antenna spiral mode.

The instant of the beginning of the visibility above 10 degrees of elevation has been analysed. At this instant, the spacecraft is nominally at azimuth 32 degrees and 13100 Km from the station. Assuming the injection errors presented in section 3.5.4, the pointing errors of the antenna, the range and range rate errors due to the dispersion are presented in Table 17.

Azimuth (deg)	Elevation (deg)	Range (Km)	Range Rate (m/s)
0.08	0.04	5.0	6.8

Table 17: 1-σ Dispersion errors at New Norcia first acquisition

For the orbit determination during the LEOP, range and Doppler measurements from New Norcia and Cebreros have been assumed. The range measurements are taken every hour, while the Doppler measurements are obtained continuously and used in the orbit determination process as interpolated measurements every 10 minutes.

The error assumptions for the measurements are listed in Table 18.

Range random error	10 m
Range considered bias	10 m
Doppler random error	1 mm/s
Doppler consider bias	0
Ground Station coordinates bias in equator plane	1 m
Ground Station coordinates bias out of equator plane	3 m

Table 18: Measurement errors for the orbit determination (1-σ)

A bias of the solar radiation pressure is considered in addition to the measurements errors in the orbit determination process. The standard deviation of this bias is of 1% of the nominal solar radiation pressure coefficient.

The results of the orbit determination process during the LEOP are presented in the next figures. The 7 days range from the injection should be enough to prepare the launcher injection correction manoeuvre (LIC). The position error is quickly diminished to below 0.2 Km and then starts increasing as the distance to the spacecraft is growing. The velocity error decreases steadily to a level of less than 1 cm/s.

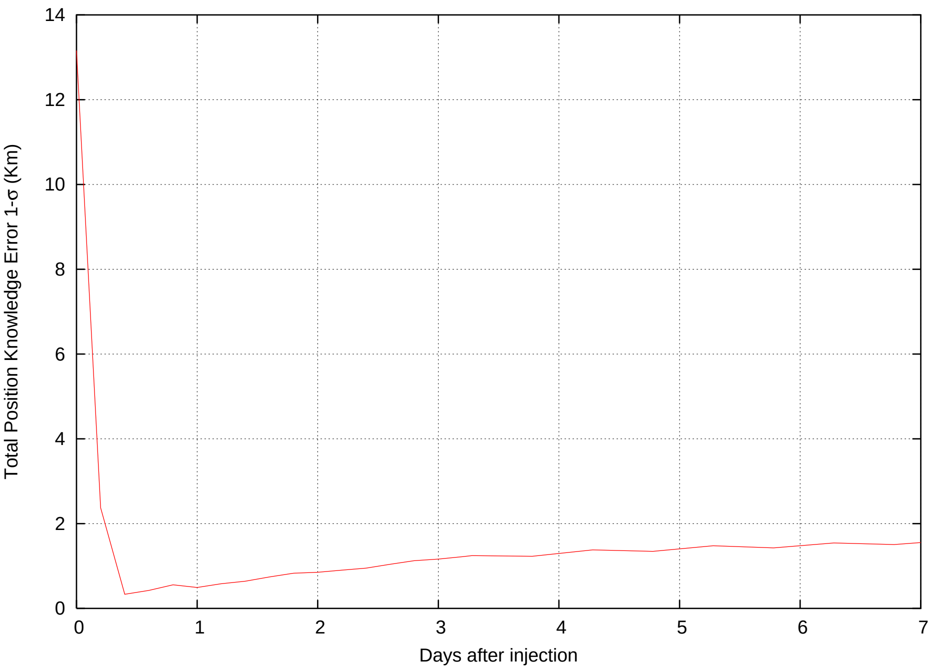

Figure 11-2: Early Orbit Determination Position Error

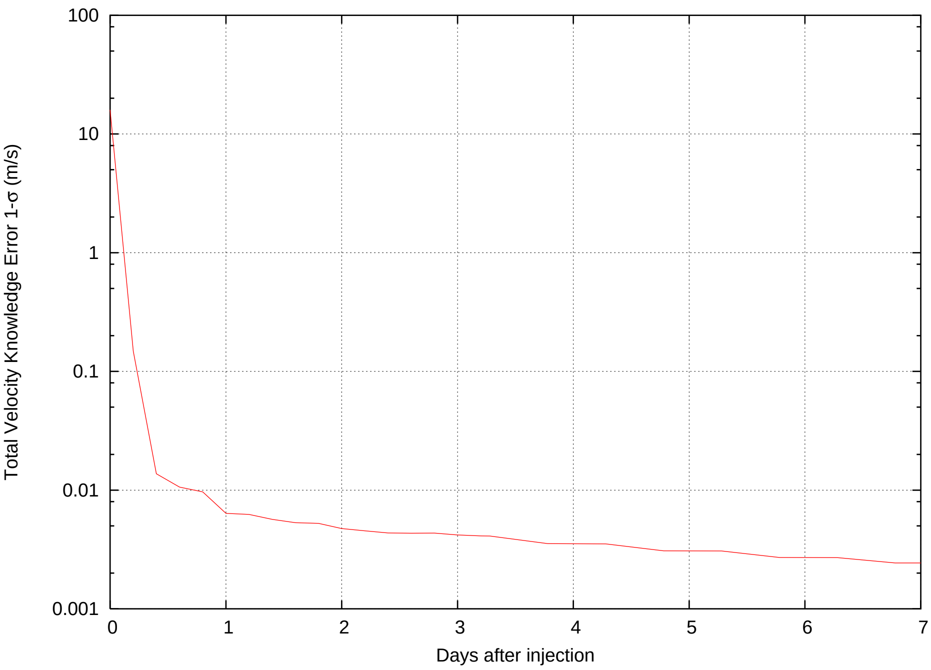

Figure 11-3: Early Orbit Determination Velocity Error

The large dispersion due to the launcher injection errors must be corrected by a first manoeuvre. The orbit determination process and first tests of the systems should get the spacecraft ready to perform such a correction manoeuvre in about 3 days.

The ΔV associated with this LIC has been computed by means of a Monte Carlo analysis (10000 simulations). The LIC is assumed to target the arrival conditions at Venus. The results presented in Table 19 are parametrized with the days elapsed since injection to the implementation of the manoeuvre. The mean value, standard deviation and maximum value of the ΔV increase with time following a parabolic law.

Days after Injection	LIC DELTA-V (M/S)			DISPERSION ERROR AFTER LIC		
	Mean	1-σ	Max	SMA (KM)	SMI (KM)	LTF (s)
4	5.09	2.87	21.90	294.65	50.82	102.55
7	5.34	3.03	23.09	288.16	56.23	98.44
10	5.65	3.23	24.52	292.79	60.57	97.89
13	6.03	3.47	26.26	305.42	64.44	100.04
16	6.51	3.77	28.44	324.28	68.70	104.23
19	7.13	4.16	31.26	349.84	74.13	110.48
22	7.98	4.70	35.14	385.00	81.68	119.52

Table 19: Statistic results of the First Correction Manoeuvre

The dispersion errors of the impact vector are also presented in the table. These dispersions include a mechanization error in the implementation of the manoeuvre of 1% in modulus and 0.5 degrees in angular error (both values are 1-σ). The dispersions are given in terms of the dimensions of the error ellipse in the plane perpendicular to the local velocity vector – SMA and SMI are the semi-major and semi-minor axes, respectively - and the error along the velocity as the Linearized Time of Flight LTF. The dispersions also increase with time after the day 7, which appears to have the advantage of the lowest semi-major axis. In the light of these results, the LIC should be performed as soon as possible after launch to save ΔV not only of this manoeuvre but also of the subsequent corrections. For the rest of the navigation analysis it is assumed that the manoeuvre is performed 7 days after injection.

11.2 Interplanetary Navigation and Venus Arrival

The interplanetary navigation will be based on radio tracking from the mission dedicated ground station at Cebreros. Range and Doppler measurements will be used to determine precisely the orbit of Venus Express and calculate the adequate corrections to guide the spacecraft to the nominal arrival conditions at Venus. The nominal trajectory selected for the navigation and guidance analysis is the one for a launch on 26/10/2005.

The assumed errors for the orbit determination during the transfer are the same as the ones presented in section 11.1. However, a smaller frequency of the measurements is assumed except for the arcs immediately before a correction manoeuvre. A long phase without any observation or orbit determination has also been considered.

The guidance strategy to reach the desired arrival conditions at Venus consists on the above-mentioned LIC and three additional Mid-Course correction manoeuvres MC. These manoeuvres are distributed along the interplanetary transfer as follows.

LIC	7 days after injection
MC1	20 days after LIC
MC2	40 days before nominal arrival
MC3	10 days before nominal arrival

Table 20: Correction manoeuvres during transfer orbit

The MC1 cleans up any residual error that the implementation errors of the LIC have left. MC2 and MC3 do the fine targeting of the capture conditions as the spacecraft arrives to Venus. A prudential time has been kept after the last correction manoeuvre and the capture manoeuvre at Venus.

In the previous issue of this document, two possible guidance strategies were studied. In the first strategy (named IIVV), both the LIC and the MC1 manoeuvres target the position at an intermediate point, which is exactly the MC2. Position errors at this point should then be small and allow reaching Venus with enough precision. The second strategy (named VVVV) consists simply on four manoeuvres targeting the arrival at Venus. It was verified that the strategy VVVV provides better performances in terms of ΔV required for the corrections and the dispersion errors at arrival to the pericentre. In the following sections only this strategy is considered.

Two possible scenarios have been studied. In the first one, the calibrated scenario, it is supposed that the measurements permit to calibrate the dynamical model. In particular, the solar radiation pressure bias can be estimated below a level of 1 %. In the second one, the perturbed scenario, it is assumed that the reaction wheel-off loading manoeuvres that have to be carried out regularly during the interplanetary trajectory introduce an uncertainty in the velocity that actually destroys the information about the solar radiation pressure.

11.2.1 Calibrated scenario

In a first step, a study by means of covariance analysis is made. Range and Doppler measurements are taken from Cebreros ground station particularly during the days before one of the correction manoeuvres. The covariance matrix that contains the errors of how the position and velocity of the spacecraft can be estimated is updated with these measurements by means of a square root information filter SRIF. The dynamic is linearized and the covariance matrix propagated with the transition matrix. The dispersion matrix, which contains the errors of how the position and velocity differ w.r.t. the nominal vales, is propagated in time. Whenever a correction manoeuvre must be implemented, the guidance matrix to the target point is computed assuming a fixed time guidance law. The covariance and dispersion matrices are updated by the effect of the manoeuvre and its implementation errors.

The main output from this study is an estimation of the ΔV necessary to guide the spacecraft and the delivery errors at the arrival to the planet. These are obtained as the dispersion errors at the B-plane of Venus arrival and the errors at the pericenter of the arrival hyperbola. A special meaning have the radial error and the osculating hyperbolic velocity error at pericenter, which are important for the capture manoeuvre in which a minimum altitude of 250 Km must be ensured.

Figure 11-4 presents the evolution of the knowledge and dispersion position errors mapped at the pericentre of the arrival hyperbola at Venus. In this case, the second manoeuvre targeting the arrival at Venus reduces the dispersion to about 100 Km. The next two manoeuvres reduce the dispersion to the level of km.

Figure 11-5 presents the dispersion error ellipses projected on the B-plane of the nominal arrival at Venus before and after the manoeuvre MC1. The effective collision section of Venus for the nominal arrival hyperbolic velocity has a radius of 14920 Km. Before the correction, the very big ellipse has almost a 40 % of its surface intersecting the section of Venus impact. After the MC1, the dispersion ellipse is greatly diminished to about 100 Km.

Figure 11-6 presents the error ellipses at the implementation of MC2. After this correction, both axis of the ellipse are reduced to a level of a few km with the semi-major axis forming an angle of about 18 degrees with the radial direction.

Figure 11-7 presents the dispersion errors ellipses before and after the last correction manoeuvre MC3. The ΔV of this manoeuvre almost vanishes for this strategy. The effect of the correction is only to remove the error perpendicular to the radial direction. The radial error is unaffected and the ellipse is turned in such a way that it is aligned with the radial direction.

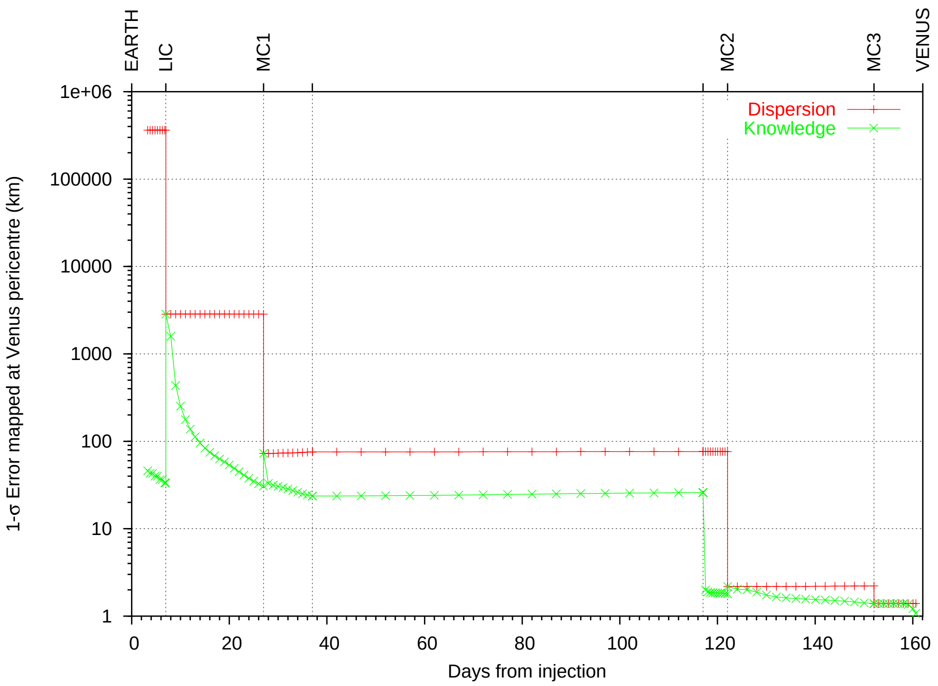

Figure 11-4: Evolution of mapped errors at hypothetical Venus pericentre (calibrated scenario)

The results of the linearized analysis have been crosschecked with a Monte Carlo method of 100000 cases. In this case, an initially perturbed state vector with the launcher injection errors is numerically propagated from injection to the pericentre of Venus at arrival. Whenever a correction manoeuvre must be implemented, the knowledge errors have been assumed to be the same as the resulting ones from the covariance analysis. The guidance matrix is computed by means of the numerically computed transition matrix for the nominal trajectory and assuming a fixed time guidance law. The implementation errors of the manoeuvre are simulated for each case.

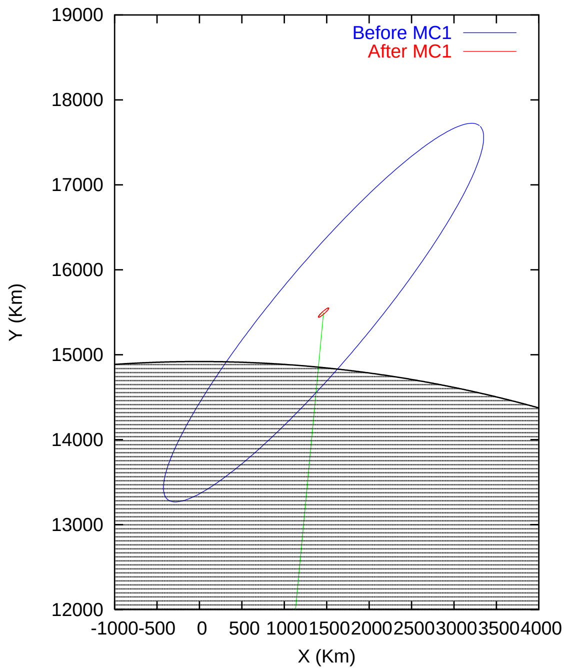

Figure 11-5: Dispersion 1-σ ellipses at Venus B-plane for MC1 (calibrated scenario)

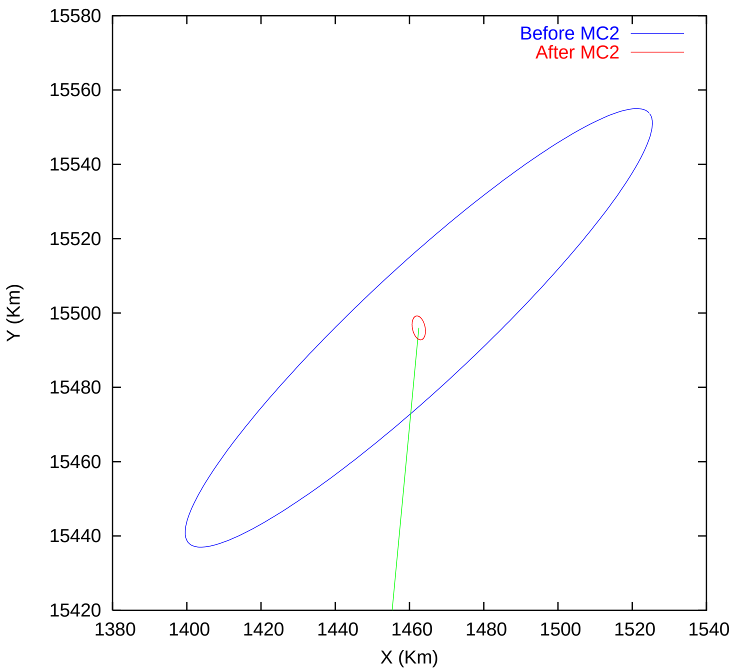

Figure 11-6: Dispersion 1-σ ellipses at Venus B-plane for MC2 (calibrated scenario)

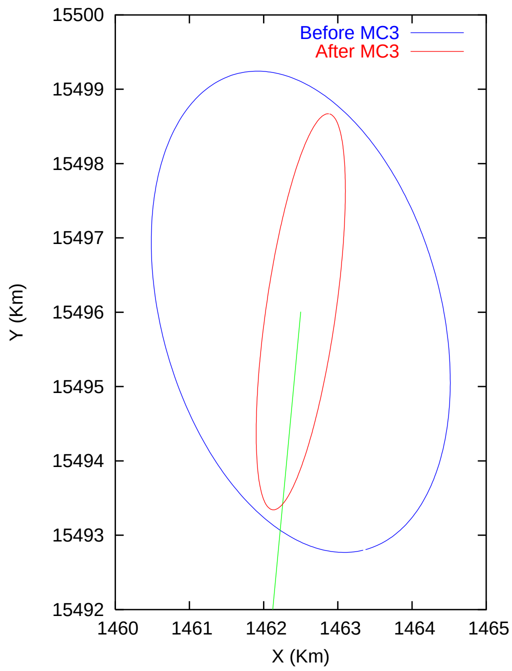

Figure 11-7: Dispersion 1-σ ellipses at Venus B-plane for MC3 (calibrated scenario)

Table 21 shows the ΔV obtained by means of the linearized method and the Monte Carlo method. Both methods agree very well on the mean values, while the 99% and maximum values of the Monte Carlo are slightly bigger.

(m/s)	Linear Method				Monte Carlo			
	Mean	1-σ	99%	Max	Mean	1-σ	99%	Max
LIC	5.34	3.04	14.59	23.05	5.41	3.09	15.08	25.53
MC1	0.13	0.06	0.32	0.48	0.10	0.09	0.45	1.35
MC2	0.01	0.01	0.04	0.07	0.01	0.01	0.05	0.18
MC3	0.00	0.00	0.01	0.01	0.00	0.00	0.01	0.02
Total	5.49	3.04	14.95	23.62	5.52	3.15	15.37	26.02

Table 21: Delta-V Statistics of the Interplanetary Trajectory Guidance (calibrated scenario)

Table 22 presents the dispersion errors at the arrival at Venus for both the linear method and the Monte Carlo. First, the impact vector errors are presented as the dimensions of the error ellipse projected on the B-plane (SMA – semi-major axis; SMI – semi-minor axis), the angle θ between the semi-major of the ellipse and the radial direction and the linearized time of flight LTF. These results are only presented for the linear method. The next rows show the error of the arrival hyperbolic velocity osculating at pericentre, the position and the velocity errors at pericentre. Both methods provide very similar results, except for the error on the modulus of the arrival hyperbolic velocity. A radial position error of about 1 Km is achieved.

	Linear Method	Monte Carlo
1-σ	**Position Errors at Impact Vector**	
SMA (Km)	2.69	-
SMI (Km)	0.47	-
θ (deg)	1.9	-
LTF (s)	0.45	-
1-σ (m/s)	**Hyperbolic Arrival Velocity**	
Modulus	2.26	0.88
1-σ (Km)	**Position Error at Pericentre**	
A-T	1.01	1.01
C-T	0.11	0.11
R	0.96	0.95
Total	1.39	1.39
1-σ (m/s)	**Velocity Error at Pericentre**	
A-T	0.69	0.69
C-T	0.37	0.37
R	1.73	1.72
Total	1.90	1.89

Table 22: Statistics of the dispersion at Venus arrival (calibrated scenario)

11.2.2 Perturbed scenario

The reaction wheel off-loadings (WOL) can not be implemented without producing a residual delta-V. For the case of Mars Express, the WOL produced a delta-V approximately 20 mm/s and had a frecuency during the Mars approach phase of about 2 per week. In the case of Venus Express, the higher solar radiation torque will increase the frecuency of the WOL. It has been assumed that they have to be implemented at an average of 4 per week.

In order to model the effect of the WOL residual delta-V's, an exponentially correlated noise has been added to every component of the acceleration vector. The accumulated effect of this noise during a period between two WOL is equivalent to an uncertainty of 5 % on the delta-V, that is about 1 mm/s.

It has also been assumed that the solar radiation pressure can not be estimated with the same precision as for the calibrated scenario, because the uncertainty in the velocity introduced by the WOL and the high frecuency of them, does not permit a proper recovery of the information about the acceleration due to the solar radiation pressure. Therefore, a bias of 5 % of the nominal value of the ballistic coefficient has been considered for the whole interplanetary trajectory.

Figure 11-8 presents the evolution of the knowledge and dispersion position errors mapped at the pericentre of the arrival hyperbola at Venus. In this case, the perturbations due to the WOL and the solar radiation pressure bias produce an increase of the errors as the spacecraft approaches Venus. The MC1 reduces the dispersion to about 100 Km, but it grows up to about 700 Km just befor the MC2. The MC2 reduces the error to about 25 Km and the MC3 maintains this level of the dispersion when the spacecraft is reaching Venus.

Figure 11-9 presents the dispersion errors ellipses projected on the B-plane of the nominal arrival at Venus before and after the manoeuvre MC1. Before the correction, the very big ellipse has almost a 30 % of its surface intersecting the section of Venus impact. After the MC1, the dispersion ellipse is greatly diminished to about 100 Km.

Figure 11-10 presents the dispersion error ellipses at the implementation of MC2. After this correction, the semi-major axis of the ellipse is reduced to about 36 km forming an angle of about 50 degrees with the radial direction.

Figure 11-11 presents the dispersion errors ellipses before and after the last correction manoeuvre MC3. The effect of the correction is only to remove most of the error perpendicular to the radial direction. The radial error is unaffected and the ellipse is turned in such a way that it is almost aligned with the radial direction.

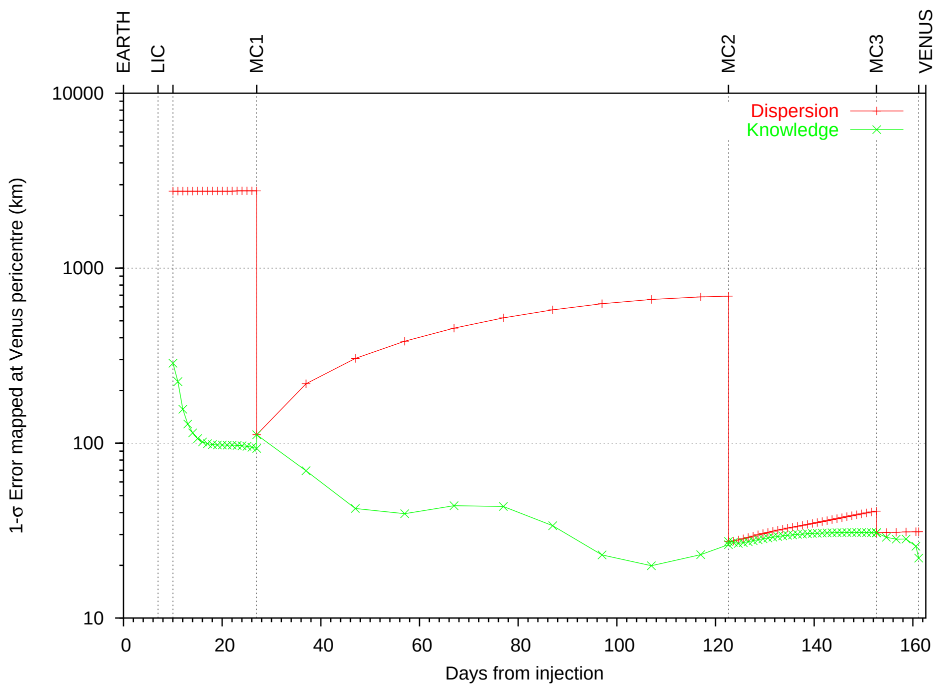

Figure 11-8: Evolution of mapped errors at hypothetical Venus pericentre (perturbed scenario)

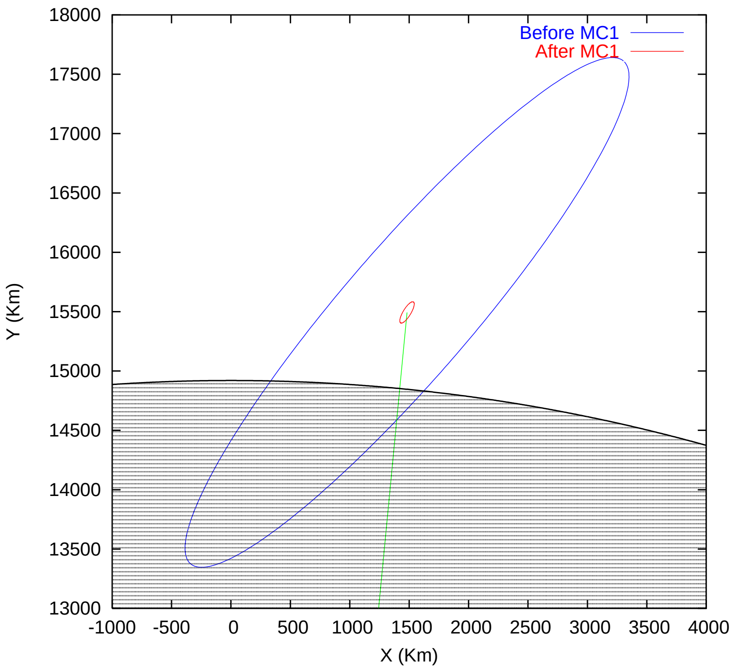

Figure 11-9: Dispersion 1-σ ellipses at Venus B-plane for MC1 (perturbed scenario)

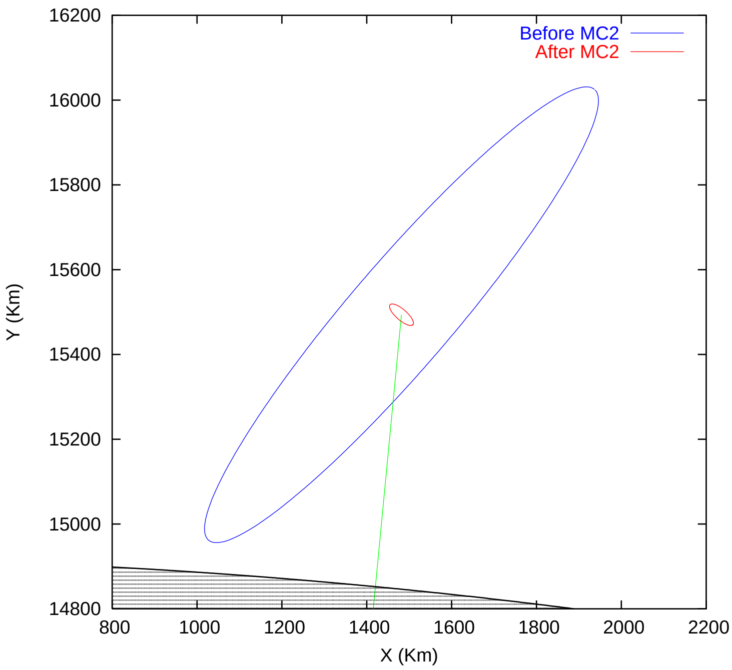

Figure 11-10: Dispersion 1-σ ellipses at Venus B-plane for MC2 (perturbed scenario)

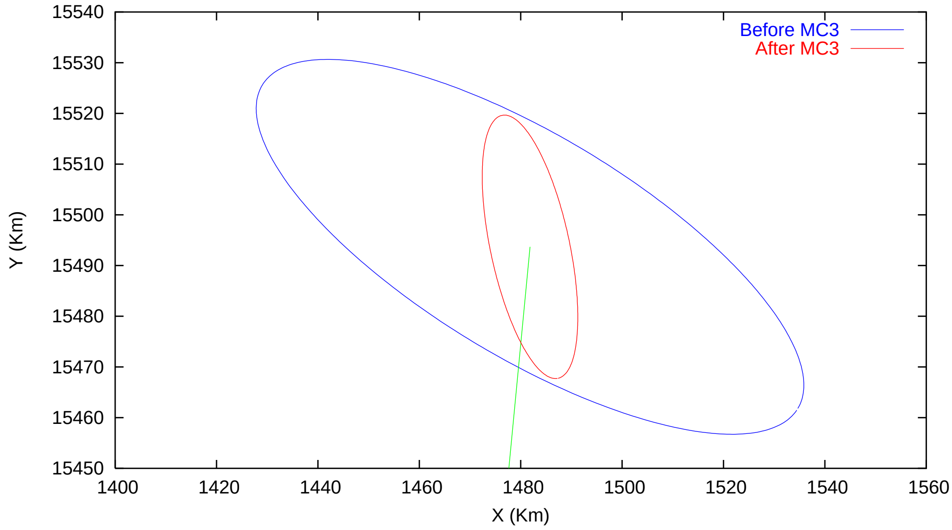

Figure 11-11: Dispersion 1-σ ellipses at Venus B-plane for MC3 (perturbed scenario)

Table 23 shows the ΔV obtained by means of the linearized method. In this case, the Monte Carlo simulation has not been carried out.

(m/s)	Linear Method			
	Mean	1-σ	99%	Max
LIC	5.15	3.07	14.58	23.04
MC1	0.12	0.06	0.3	0.45
MC2	0.11	0.08	0.35	0.55
MC3	0.05	0.04	0.16	0.26
Total	5.43	3.07	15.39	24.31

Table 23: Delta-V Statistics of the Interplanetary Trajectory Guidance (perturbed scenario)

Table 24 presents the dispersion errors at the arrival at Venus. A radial position error of about 16 Km can be achieved.

Comparing the results of both scenarios, while there is no much difference in the required ΔV, in fact the maximum expected for the corrections is only 1 m/s more in the perturbed scenario, however the dispersions at Venus arrival are one order of magnitude bigger in the perturbed scenario. The radial dispersion at pericentre of about 1 Km in the calibrated scenario becomes of about 16 Km in the perturbed scenario. The dispersion ellipses at the impact vector are also about 20 times bigger.

	Linear Method
1-σ	Position Error at Impact Vector
SMA (Km)	26.5
SMI (Km)	7.8
θ (deg)	17.5
LTF (s)	9.84
1-σ (m/s)	Hyperbolic Arrival Velocity
Modulus	0.49
1-σ (km)	Position Error at Pericentre
A-T	26.6
C-T	3.4
R	15.9
Total	31.2
1-σ (m/s)	Velocity Error at Pericentre
A-T	10.9
C-T	5.5
R	18.7
Total	22.3

Table 24: Statistics of the dispersion at Venus arrival (perturbed scenario)

11.2.3 Safe Mode during Venus Approach Phase

The effect of a safe mode when the spacecraft is approaching Venus is analysed in this section. When the spacecraft enter this mode it performs a sequence of manoeuvres to orientate the solar panels towards the Sun, which produce a residual delta-V. Typically this delta-V is about 15 cm/s and its direction is unknown a priori.

Obviously, if the spacecraft enters this mode during the Venus approach phase, the knowledge of the velocity components is destroyed and the dispersion is increased. This has the effect of increasing the Venus B-plane error ellipses and the dispersion at the pericentre. If the safe mode occurs very late, there is no time left to react but the effect on the B-plane target is small. If the safe mode occurs relatively early, there is still the possibility to determine the trajectory from the measurements of a few station passes and perform a trim manoeuvre to correct the dispersion on the B-plane.

Figure 11-12 presents the dispersion ellipses on the B-plane at Venus arrival mapped just after a safe mode occurs 2,4,6 or 8 days before the arrival. The dimensions of the ellipses increase as the propagated time is longer. The ellipse corresponding to the nominal arrival case without safe mode is clearly contained inside the others and is the only one whose semi-major axis deviates from the radial direction.

A correction manoeuvre 1.5 days before the arrival to Venus has been considered, when possible, to analyse the recovery of knowledge and the final dispersion at the pericentre of the arrival hyperbola that could be achieved if such a manoeuvre is possible. Normally, this is an optimistic case and due to operational reasons a correction manoeuvre should not be implemented during the last two days before the arrival.

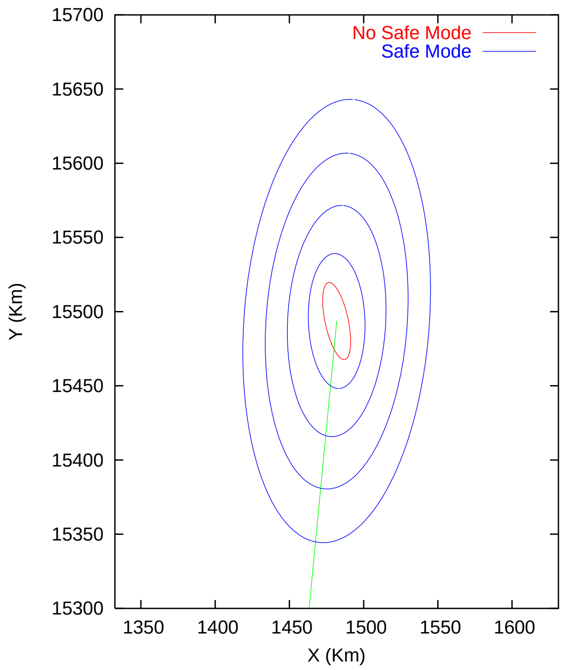

Figure 11-12: Dispersion 1-σ error ellipses on B-plane after safe mode during Venus approach

Table 25 presents the dispersions at B-plane and at the pericentre for the case of no safe mode and for the cases of a safe mode from 1 to 8 days before the arrival. A safe mode 1 day before arrival has little effect over the dispersion. For the other cases, if a trim manoeuvre is possible, the error in the pericentre altitude can be corrected up to a value slightly lower than in the nominal case. This is due to the fact that the very late orbit determination campaign and the additional correction manoeuvre allow a reduction of the radial error of about 2 Km. However, for the case of a safe mode 2 days before arrival the total position and velocity dispersions can not be recovered below the values of the nominal arrival.

		NO SAFE MODE	SAFE MODE: Days before Venus arrival				
			1	2	4	6	8
1-σ Dispersion at B-Plane							
SMA	Km	26.5	-	44.9	77.8	113.3	149.7
SMI	Km	7.8	-	18.9	32.9	47.5	62.4
1-σ Dispersion at Pericentre – No correction							
RADIAL	Km	15.9	16.9	19.5	27.3	36.6	46.4
TOTAL POSITION	Km	31.2	36.9	51.2	88.7	129.5	171.4
TOTAL VELOCITY	m/s	22.3	26.9	38.0	66.2	96.8	128.1
1-σ Dispersion at Pericentre – Correction 1.5 days before arrival							
RADIAL	Km	-	-	13.7	13.5	13.2	13.2
TOTAL POSITION	Km	-	-	34.2	24.8	25.5	26.6
TOTAL VELOCITY	m/s	-	-	25.5	20.3	20.9	21.9

Table 25: 1-σ Dispersions at Venus arrival with safe modes (perturbed scenario)

The recommended strategy to follow is:

- Before 4 days to the arrival at Venus: if a safe mode occurs, then the orbit can be recovered from the measurements of the following passes and a very late correction can be implemented 2-4 days before Venus arrival to obtain similar dispersions as in the nominal case.
- After 4 days to the arrival at Venus: the spacecraft safe mode system should be disabled.

11.2.4 Capture Manoeuvre

The capture manoeuvre is mainly affected by the error in the arrival hyperbola altitude and the error in the arrival hyperbolic velocity. The minimum altitude of 250 km must be respected in any case. A Monte Carlo method has been used to analyse this minimum altitude. Conservative errors taken from the perturbed scenario are assumed for the pericentre altitude and hyperbolic velocity of the arrival hyperbola:

1-σ Altitude Error: 15.9 Km 1-σ Hyperbolic Velocity Error: 0.49 m/s

The trajectory during the thrusting phase has been numerically integrated in each simulation and the minimum altitude during this trajectory stored. The thrust is taken opposite to the local velocity, in the assumption that the real attitude profile programmed on-board for the manoeuvre will not present significant differences.

With the new capture strategy, the spacecraft does not reach the constrain of 250 Km during the VOI for almost every day of the launch window, except for launches on 02/11 and 03/11/2005. For a launch on 26/10/2005 the minimum altitude during the thrust arc is 278.2 Km. Table 26 presents the statistics obtained by Monte Carlo simulation of the minimum altitude reached during the thrust arc. Two cases are presented: for launch on 26/10, the minimum altitude was 222.7 Km and a 99% of the cases has minimum altitude above 258.4 Km and for launch on 02/11, the minimum is 191.8 Km and the 99 % is at 231.5 Km.

	Minimum Altitude during Thrust (Km)				
Launch day	**Mean**	**1-σ**	**Min**	**99 %**	**Max**
26/10/2005	278.4	14.4	222.7	258.4	338.0
02/11/2005	250.0	13.9	191.8	231.5	308.8

Table 26: Statistic of the minimum altitude during thrust phase of the Capture Manoeuvre

Therefore, the stochastic minimum value of the altitude during the thrust arc is about 60 Km lower than the nominal value. For the cases in which the minimum is below the constraint there are two possibilities. The first possibility is to increase the pericentre altitude of the arrival hyperbola which costs about 1 m/s every 10 Km, so less than 6 m/s are needed. The second possibility is to delay the ignition of the main engine starting the VOI manoeuvre at a higher true anomaly. As the cases with lower pericenter altitude are at the beginning of the launch window, the first apocenter lowering with the main engine can be used to correct the deviation of the achieved argument of pericenter at beginning of the operational orbit. This could be performed with very small delta-V cost.

12. LAUNCH WINDOW AND FLIGHT PROGRAMS

The Launch Window for Venus Express depends on the Soyuz-Fregat performances, on the Venus regions that the scientists want to observe at low altitude, and on the capabilities of the spacecraft (available propellant, thermal, etc.). At the end of the Preliminary Mission Analysis to be performed by STARSEM, the performances of Soyuz-Fregat will be known with enough accuracy to precisely define the launch period. In particular, the effect of a reduced number of Flight Programs on the performance must be taken into account. However, due to programmatic reasons, the Launch Period of Venus Express has been fixed to the period from 26/10/2005 to 24/11/2005.

The optimum launch conditions for every launch day have been assessed in Section 4 - Transfer to Venus in 2005. These launch conditions would be selected in the case that one dedicated flight program for the Fregat upper stage were available for every launch day. The Fregat guidance system will reproduce a fixed trajectory in an inertial reference frame whose orientation is defined by the lift-off time. Therefore, a given flight program provides a unique combination of escape velocity V_∞ and declination of the escape hyperbola δ_∞ relative to the Earth equator, while the right ascension can be freely chosen by adjusting the lift-off time.

In order to reduce the number of flight programs needed to cover the whole launch window, some launch days can be grouped in one single flight program. This is normally possible with a ΔV penalty with respect to the optimum launch conditions. The diagrams of departure hyperbolic velocity vs. declination $V_\infty - \delta_\infty$ are useful to see which days could be grouped, although they provide little information about the performance loss. The criterion to group the flight programs is to reduce the ΔV penalty below about 30 m/s and to obtain the best performances.

A correction manoeuvre is considered when using a non-optimum flight program for a launch day. The correction strategy consists on a manoeuvre at Earth departure 2 days after the injection into the escape orbit. At this time, the spacecraft is yet in the sphere of influence of the Earth and with the manoeuvre, it is intended to change the escape velocity trying to approximate the interplanetary arc to the optimum one.

A detailed analysis of the launch window is presented in R 8.

12.1 Launch Strategy: Free arrival date.

A launch strategy consisting on 10 flight programs has been obtained by ESOC MAO and crosschecked and agreed by STARSEM. The strategy is defined as follows:

- The optimum launch conditions for 28/10 are used from 26/10 to 04/11.
- The optimum launch conditions for 09/11 are used from 04/11 to 11/11.
- The optimum launch conditions for 13/11 are used also for 12/11 and 14/11.
- From 15/11 to 20/11 3 couples of two consecutive launch days are grouped: 15-16, 17-18 and 19-20. The launch conditions (V_∞ , δ_∞) for these couples are selected in order to achieve about the same ΔV penalty for both launch days.
- From 21/11 to 24/11 one flight program per day.

In order to define the interface with STARSEM for the launch of Venus Express it was decided to compute the targets at the B-plane of Venus. These targets shall not include any further deep space manoeuvre performed by the spacecraft itself. Therefore, the state vector at injection has been propagated taking into account all meaningful perturbations until reaching the Venus pericentre. From the pericentre state vector, the arrival hyperbolic velocity and the position of the impact vector in the B-plane can be derived.

Table 27 presents the launch conditions and the arrival parameters for every launch day with the 10-flight programs strategy. Differences in the launch conditions (V_∞ , δ_∞) with the same flight program are due only to rounding errors. The lift-off hours appearing in the table are the definitive times obtained by STARSEM.

Earth Departure				Venus Arrival					FP
Date	Lift Off	V_∞ Km/s	δ_∞ deg	Date	Hour	V_∞ Km/s	ξ Km	η Km	
26.10.05	04:43:38.7	2.7855	-25.614	06.04.06	21:16:27	4.6215	8815.3	12826.5	1
27.10.05	04:37:42.4	2.7855	-25.614	07.04.06	02:12:56	4.6192	8824.2	12828.4	
28.10.05	04:31:46.4	2.7855	-25.614	07.04.06	07:02:54	4.6171	8832.2	12829.9	
29.10.05	04:25:36.0	2.7855	-25.613	07.04.06	11:46:26	4.6153	8839.3	12830.9	
30.10.05	04:19:25.9	2.7855	-25.613	07.04.06	16:24:56	4.6139	8845.4	12831.5	
31.10.05	04:13:10.7	2.7855	-25.613	07.04.06	20:57:02	4.6128	8850.7	12831.5	
01.11.05	04:06:50.1	2.7855	-25.613	08.04.06	01:22:50	4.6121	8854.9	12830.9	
02.11.05	04:00:23.6	2.7855	-25.613	08.04.06	05:41:07	4.6119	8858.2	12829.6	
03.11.05	03:53:50.4	2.7855	-25.612	08.04.06	09:52:23	4.6120	8860.5	12827.5	
04.11.05	03:47:09.4	2.7855	-25.612	08.04.06	13:53:36	4.6127	8861.9	12824.2	
05.11.05	04:03:21.1	2.7904	-21.052	10.04.06	17:27:06	4.6059	8769.9	12910.2	2
06.11.05	03:57:04.2	2.7904	-21.051	10.04.06	18:29:06	4.6036	8767.5	12919.5	
07.11.05	03:44:32.1	2.7904	-21.051	10.04.06	12:10:22	4.6033	8790.0	12905.4	
08.11.05	03:39:30.3	2.7904	-21.051	11.04.06	04:26:18	4.5999	8733.8	12955.0	
09.11.05	03:33:34.5	2.7904	-21.050	11.04.06	08:16:25	4.5990	8715.3	12970.4	
10.11.05	03:26:40.7	2.7904	-21.050	11.04.06	11:26:04	4.5986	8697.4	12983.7	
11.11.05	03:19:19.0	2.7904	-21.050	11.04.06	14:27:44	4.5987	8677.8	12996.4	
12.11.05	03:19:32.8	2.8560	-19.502	12.04.06	09:12:37	4.5983	8582.6	13061.0	3
13.11.05	03:12:43.3	2.8560	-19.502	12.04.06	11:55:55	4.5984	8552.8	13080.1	
14.11.05	03:04:40.2	2.8560	-19.502	12.04.06	14:12:26	4.5990	8523.5	13097.1	
15.11.05	03:02:33.2	2.9248	-18.474	13.04.06	03:09:15	4.5998	8426.0	13157.4	4
16.11.05	02:55:14.4	2.9248	-18.474	13.04.06	05:11:16	4.6010	8385.8	13179.3	
17.11.05	02:51:57.2	2.9945	-17.688	13.04.06	15:15:14	4.6026	8285.2	13237.5	5
18.11.05	02:44:46.0	2.9945	-17.688	13.04.06	16:50:14	4.6044	8235.4	13262.5	
19.11.05	02:41:23.8	3.0708	-17.013	14.04.06	01:21:59	4.6067	8124.3	13323.4	6
20.11.05	02:34:20.2	3.0708	-17.013	14.04.06	02:28:47	4.6093	8063.9	13351.6	
21.11.05	02:30:20.9	3.1388	-16.423	14.04.06	09:16:48	4.6122	7945.3	13413.0	7
22.11.05	02:25:27.3	3.1886	-16.031	14.04.06	13:49:55	4.6155	7834.5	13467.4	8
23.11.05	02:20:39.8	3.2415	-15.637	14.04.06	18:04:16	4.6191	7714.0	13525.2	9
24.11.05	02:15:58.5	3.2972	-15.240	14.04.06	21:59:19	4.6231	7583.1	13586.5	10

Table 27: Departure and Arrival conditions for first strategy with free arrival date

The B-plane is defined as the plane perpendicular to the arrival hyperbolic velocity with origin in the centre of Venus. In order to determine the coordinates of the target point, two reference axes ξ and η are used. η is perpendicular to the arrival hyperbolic velocity vector and the Sun-Venus radius vector at the arrival time. ξ is perpendicular to η and the arrival hyperbolic velocity vector. The exact definitions of the axes unitary vectors are given by the following formulas:

$$\bar{\eta} = \frac{\bar{r}_v \times \bar{v}_\infty}{\left|\bar{r}_v \times \bar{v}_\infty\right|} \qquad \bar{\xi} = \frac{\bar{\eta} \times \bar{v}_\infty}{\left|\bar{v}_\infty\right|}$$

Where:

$\bar{r}_v$ - radius vector from Sun to Venus at pericentre passage.

$\bar{V}_\infty$ - asymptotic velocity vector of incoming hyperbola.

The information provided to STARSEM consists on the arrival time at pericentre, the coordinates (ξ,η) of the impact vector in the Venus B-plane and the arrival hyperbolic velocity. This information is summarized in Table 28.

Launch Date	Venus Arrival Time		Coordinates in B-Plane		Arrival Velocity	LP
dd.mm.yy	Date dd.mm.yy	Time hh:mm:ss	ξ Km	η Km	km/s	
26.10.05	07.04.06	14:34:46	-402910.0	477604.3	4.71583	1
27.10.05	07.04.06	10:56:58	-179039.8	214926.6	4.65878	
28.10.05	07.04.06	06:59:55	8832.0	12830.2	4.61705	
29.10.05	07.04.06	06:55:18	165801.1	-171656.8	4.57756	
30.10.05	07.04.06	06:58:29	285493.5	-301861.3	4.55099	
31.10.05	07.04.06	07:57:06	370171.2	-393537.7	4.53235	
01.11.05	07.04.06	10:04:09	419980.4	-448933.4	4.52078	
02.11.05	07.04.06	13:24:45	434085.0	-468926.3	4.51585	
03.11.05	07.04.06	18:00:19	409870.8	-452562.2	4.51759	
04.11.05	07.04.06	23:48:43	340359.3	-395374.8	4.52670	
05.11.05	10.04.06	01:25:22	383543.5	-417174.2	4.50165	2
06.11.05	10.04.06	05:38:13	209841.4	-287053.2	4.52376	
07.11.05	10.04.06	16:08:37	334180.5	-119417.7	4.59751	
08.11.05	11.04.06	00:32:10	380680.0	-265882.7	4.54561	
09.11.05	11.04.06	08:06:25	8686.7	13002.4	4.59897	
10.11.05	12.04.06	03:35:03	-370311.0	416949.3	4.69805	
11.11.05	12.04.06	18:59:27	-807989.6	892297.2	4.81761	
12.11.05	11.04.06	10:11:51	640048.9	-591307.9	4.44890	3
13.11.05	12.04.06	11:46:38	8553.4	13079.9	4.59836	
14.11.05	13.04.06	14:05:14	-704323.2	721333.3	4.78266	
15.11.05	12.04.06	07:20:56	557407.9	-502122.6	4.46441	4
16.11.05	13.04.06	18:52:59	-384673.9	370528.5	4.69548	
17.11.05	12.04.06	12:47:35	735125.8	-626555.9	4.43069	5
18.11.05	14.04.06	07:42:16	-419536.0	396322.9	4.71019	
19.11.05	12.04.06	20:02:28	845623.9	-682557.0	4.41398	6
20.11.05	14.04.06	20:53:23	-506152.1	468031.0	4.74062	
21.11.05	14.04.06	09:15:41	7944.8	13413.4	4.61217	7
22.11.05	14.04.06	13:48:38	7833.9	13467.8	4.61548	8
23.11.05	14.04.06	18:03:18	7713.3	13525.6	4.61911	9
24.11.05	14.04.06	21:58:12	7582.3	13587.0	4.62305	10

Table 28: STARSEM targets for 10 launch programs strategy with free arrival date

Table 29 presents the ΔV performances for every launch day when using the 10 flight programs strategy. The table contains by columns: the launch day, the magnitude of the correction manoeuvre 2 days after injection, the impulsive manoeuvre for Venus capture, the total ΔV (sum of correction and capture manoeuvres) and an estimation of the total ΔV considering gravity losses (12.8 % of the capture ΔV). The worst case is estimated for launches on 26/10/2005 and 11/11/2005 with about 1317 m/s. The biggest correction manoeuvre for the launch program is for launch on 11/11/2005 with 26 m/s.

Launch Date	ΔV_C m/s	ΔV_A m/s	ΔV_T m/s	ΔV_{T*} m/s	FP
05.10.26	15.6	1153.7	1169.3	1317.0	1
05.10.27	7.6	1152.7	1160.3	1307.9	
05.10.28	0.9	1151.9	1152.8	1300.2	
05.10.29	5.0	1151.1	1156.1	1303.4	
05.10.30	9.0	1150.5	1159.5	1306.7	
05.10.31	11.7	1150.1	1161.8	1309.0	
05.11.01	13.3	1149.8	1163.1	1310.3	
05.11.02	13.7	1149.7	1163.4	1310.5	
05.11.03	13.1	1149.7	1162.8	1310.0	
05.11.04	12.2	1150.0	1162.2	1309.4	
05.11.05	9.8	1147.2	1157.0	1303.8	2
05.11.06	3.9	1146.2	1150.1	1296.8	
05.11.07	17.7	1146.0	1163.8	1310.5	
05.11.08	12.8	1144.6	1157.4	1304.0	
05.11.09	1.5	1144.3	1145.8	1292.3	
05.11.10	11.9	1144.1	1156.0	1302.5	
05.11.11	26.0	1144.1	1170.2	1316.6	
05.11.12	19.7	1144.0	1163.6	1310.0	3
05.11.13	1.7	1144.0	1145.7	1292.1	
05.11.14	23.1	1144.3	1167.4	1313.9	
05.11.15	18.5	1144.6	1163.1	1309.6	4
05.11.16	12.5	1145.1	1157.6	1304.1	
05.11.17	23.4	1145.8	1169.2	1315.8	5
05.11.18	13.5	1146.5	1160.1	1306.8	
05.11.19	21.7	1147.5	1169.2	1316.1	6
05.11.20	20.7	1148.6	1169.2	1316.3	
05.11.21	0.0	1149.8	1149.8	1297.0	7
05.11.22	0.1	1151.2	1151.3	1298.6	8
05.11.23	0.1	1152.7	1152.8	1300.4	9
05.11.24	0.1	1154.4	1154.5	1302.2	10

Table 29: Manoeuvre balance for first strategy with free arrival date

Figure 12-1 shows the $V_\infty - \delta_\infty$ diagram for the 10 flight programs strategy. The line with diamond symbols represents the selected flight programs. The small dots represent the optimum launch conditions, which in this case are very near the launch conditions of the flight program.

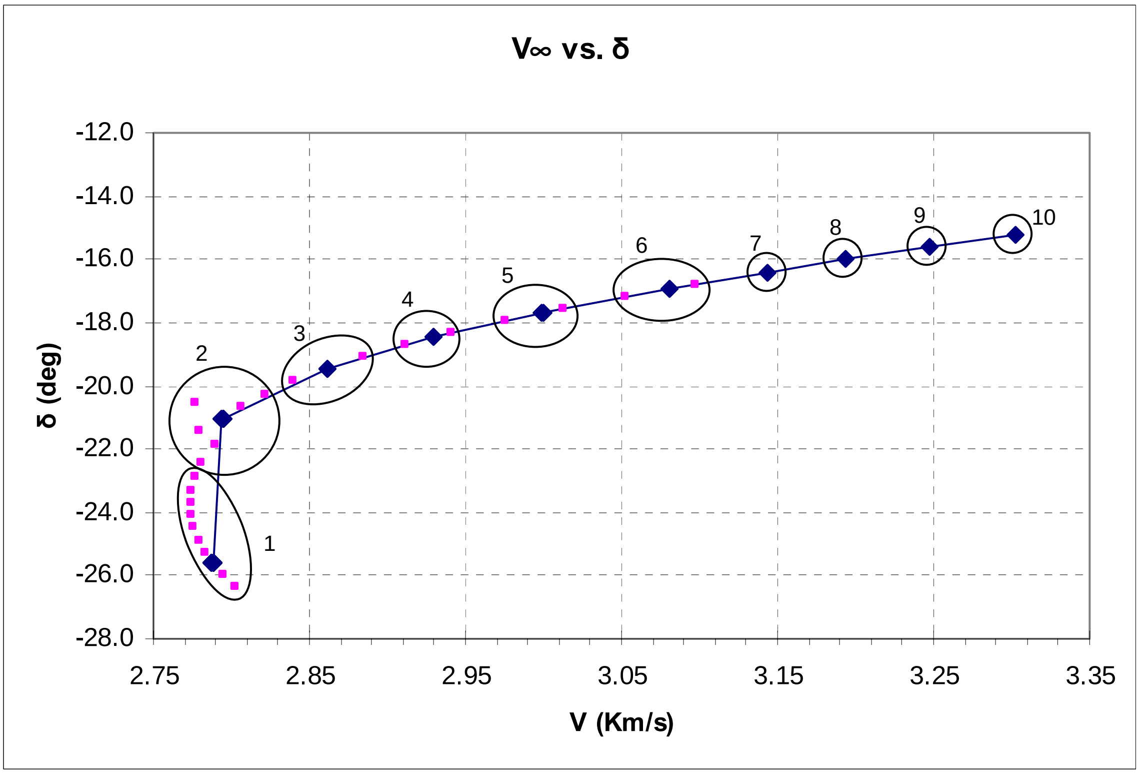

Figure 12-1: 10 Flight Programs Strategy: departure hyperbolic velocity vs. declination

12.2 Modification of the arrival hour

For operational reasons it is desirable that the VOI occurs during the visibility from one of the ESA ground stations, Cebreros or New Norcia, in order to have telemetry of the operations before and after the thrust arc. For the arrival period from 6/4/2006 to 14/4/2006 there is a gap of visibility of about 6 hours between the LOS of Cebreros at approximately 10 UTC and the AOS of New Norcia at approximately 16 UTC. Whenever it is possible, the thrust arc should be out of a band covering from 2 hours before Cebreros LOS and half hour after New Norcia AOS.

The deterministic correction manoeuvre due to the Launch Program 2 days after injection can be modified in order to shift the arrival hour out of this band. The delta-V required for this obviously grows with the number of hours to be shifted, but also depends strongly on the launch day. This delta-V can be small in some cases but can reach more than 20 m/s for other cases. The decision on whether to modify the arrival hour can be taken when the stochastic manoeuvre to correct the launcher disperrsion errors is known. If this manoeuvre is small, then enough delta-V can be available to perform the change of the arrival hour.

Figure 12-2 presents a diagram of the visibility from the ESA ground stations at the arrival to Venus for any day of the launch window. The thrust arc of the VOI are represented with vertical lines and have a duration approximately of 52 minutes. For 16 launch days, the arrival hour provides good observability conditions. For the rest of the launch days that are in the visibility gap band, the diagram also shows the delta-V penalty to shift the arrival hour in both directions in order to reach good observability conditions. 7 launch days can be shifted with us much as 5 m/s.

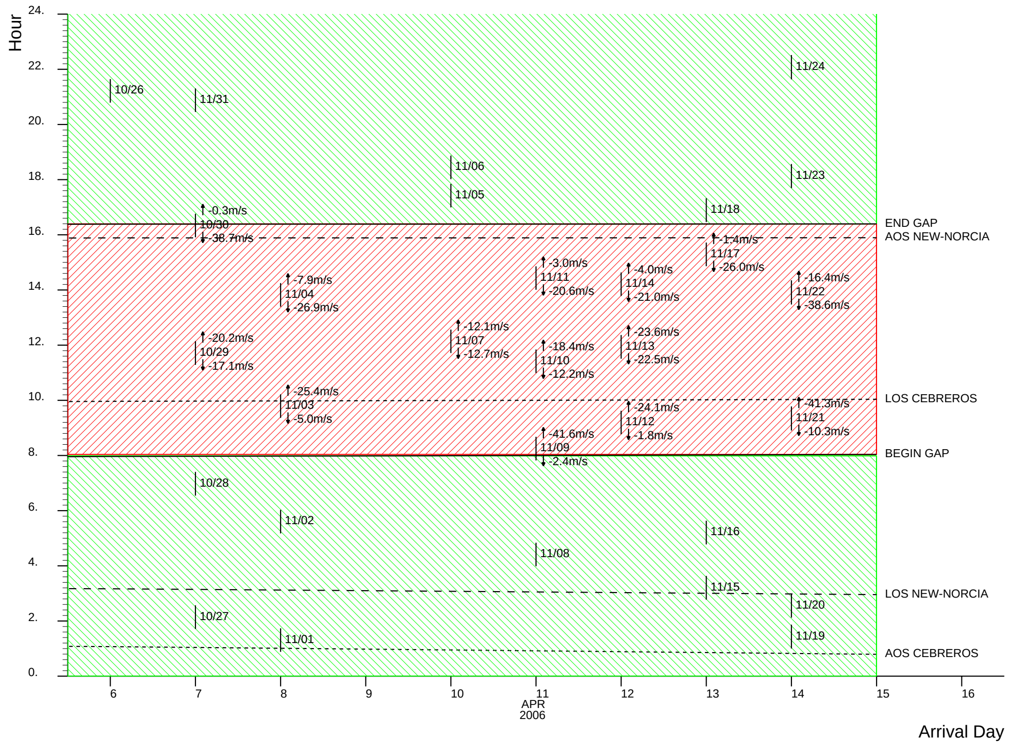

Figure 12-2: Diagram of arrival hours and visibility from ESA ground stations.

The following table presents the hours shifted and the ΔV penalty for the mission in case of modifying the arrival hour for any day of the launch window for which the optimum arrival hour is out of the observed region from the ESA ground stations.

Launch Day	Arrival Day	Hour Shift (h)		ΔV Penalty (m/s)	
		Earlier	Later	Earlier	Later
29/10/2005	7/4/2006	-4.17	5.10	17.1	20.2
30/10/2005	7/4/2006	-8.79	0.48	38.7	0.3
03/11/2005	7/4/2006	-2.24	7.03	5.0	25.4
04/11/2005	8/4/2006	-6.27	2.99	26.9	7.9
07/11/2005	8/4/2006	-4.57	4.67	12.7	12.1
09/11/2005	8/4/2006	-0.68	8.57	2.4	41.6
10/11/2005	8/4/2006	-3.84	5.40	12.2	18.4
11/11/2005	10/4/2006	-6.86	2.38	20.6	3.0
12/11/2005	10/4/2006	-1.61	7.62	1.8	24.1
13/11/2005	10/4/2006	-4.36	4.88	22.5	23.6
14/11/2005	11/4/2006	-6.62	2.61	21.0	4.0
17/11/2005	11/4/2006	-7.70	1.53	26.0	1.4
21/11/2005	11/4/2006	-1.75	7.49	10.3	41.3
22/11/2005	11/4/2006	-6.32	2.92	38.6	16.4

Table 30: Delta-V penalty to modify the arrival hour as a function of the launch day.

13. DELTA-V BUDGET

The following table describes the assumptions made for the calculations of the propellant budget of the mission from the launch up to the end of the extended mission lifetime.

Launch mass (Kg)	1270.0
Launch vehicle adaptor mass (Kg)	38.3
Spacecraft dry mass (Kg)	670.9
ME Propellant cut-off (Kg)	60.1
ΔV for launcher injection correction (m/s)	15.0
ΔV for interplanetary navigation (m/s)	5.0
ΔV for RCT check-out manoeuvre (m/s)	1.0
ΔV for ME check-out manoeuvre (m/s)	3.0
Propellant for WOL during interplanetary arc (Kg)	1.0
Propellant for WOL during operational orbit (Kg)	7.0
Propellant for safe modes (Kg)	3.0
Residual propellant (Kg)	7.7

Table 31: Assumptions for the propellant budget

For every launch day, the delta-V required for the mission can be obtained computing the deterministic manoeuvre for the launch program correction, the magnitudes of the manoeuvres for the capture and the apocentre lowering up to the operational orbit and for the maintenance of the operational orbit during the extended mission lifetime. These results are presented in Table 32, which presents as well the remaining delta-V at EOL, that is, the delta-V that is left after 1000 days of operations. The total delta-V also includes the 20 m/s for launcher injection correction and interplanetary navigation, 1 m/s of the RCT check-out and 3 m/s of the ME check-out.

The worst case is for launch on 24/11/2005 requiring a total of 1737 m/s to complete the mission. A margin of 17 m/s will still be available at the operational orbit, which would allow a mission extension of about 240 days.

The sensitivity of the fuel budget to the spacecraft dry mass can be expressed in terms of a loss of the delta-V at EOL. An increase of the dry mass produces a diminution of the available propellant for the ME during the first apocentre lowering. Therefore, the orbit reached after the two ME thrusts is higher and the RCT has to provide an additional delta-V to compensate this. Eventually, the efficiency of the capture and transfer to the operational orbit will be reduced.

For a launch on 26/10/2005, if the spacecraft dry mass is increased by one extra Kg, that is 671.9 Kg, the delta-V at EOL will be reduced by 10 m/s, which reduces about 20 days the possible extension of the mission. For a launch on 24/11/2005, the reduction of the margin is of about 12.4 m/s, that is, the delta-V at EOL will be 5 m/s and the possible mission extension would reduced from 240 to 160 days.

Launch Day	ΔV LP (m/s)	ΔV VOI (m/s)	ΔV 1st APO. LOW. w. ME (m/s)	ΔV APO. LOW. w. RCT (m/s)	ΔV OP.ORBIT CONTROL (m/s)	ΔV TOTAL (m/s)	ΔV MARGIN (m/s)
2005/10/26	15.6	1286.4	268.9	27.6	40.0	1662.5	87.8
2005/10/27	7.6	1284.5	279.4	17.1	47.9	1660.5	90.3
2005/10/28	0.9	1284.1	287.4	8.1	44.4	1648.9	101.8
2005/10/29	5.0	1283.9	283.4	22.8	40.9	1659.9	101.1
2005/10/30	9.0	1281.6	280.8	15.6	48.7	1659.5	90.7
2005/10/31	11.8	1281.1	278.2	18.6	47.5	1661.0	89.3
2005/11/01	13.3	1280.6	276.9	19.9	46.3	1660.9	89.1
2005/11/02	13.7	1280.0	276.7	20.5	56.5	1671.5	78.7
2005/11/03	13.1	1280.2	277.3	19.9	56.9	1671.3	78.9
2005/11/04	12.2	1281.0	277.7	19.5	47.2	1661.5	88.9
2005/11/05	9.8	1281.4	279.4	25.6	69.0	1689.2	68.9
2005/11/06	3.9	1279.5	287.8	9.0	72.2	1676.4	74.2
2005/11/07	17.7	1279.1	273.2	26.5	51.5	1672.0	77.8
2005/11/08	12.8	1278.7	278.9	19.9	67.4	1681.6	68.7
2005/11/09	1.5	1279.4	290.7	19.4	62.2	1677.2	85.4
2005/11/10	11.9	1279.4	279.1	18.9	68.5	1681.8	68.6
2005/11/11	26.1	1276.5	266.1	38.3	62.3	1693.2	56.3
2005/11/12	19.7	1279.5	270.1	29.4	82.4	1705.0	44.7
2005/11/13	1.7	1285.3	285.0	13.8	71.2	1680.8	70.5
2005/11/14	23.1	1282.5	263.4	37.0	72.3	1702.4	47.4
2005/11/15	18.5	1296.6	255.7	44.4	59.7	1698.8	52.3
2005/11/16	12.5	1288.6	269.6	31.1	72.4	1698.1	52.7
2005/11/17	23.4	1293.6	252.5	49.4	72.5	1715.3	35.4
2005/11/18	13.5	1301.6	256.3	45.1	69.7	1710.2	41.5
2005/11/19	21.7	1306.3	242.2	63.6	71.4	1729.3	22.7
2005/11/20	20.7	1310.9	238.5	66.6	69.2	1729.8	22.4
2005/11/21	0.0	1309.3	263.6	39.9	82.4	1719.2	33.8
2005/11/22	0.1	1317.5	254.8	54.1	72.4	1722.8	31.0
2005/11/23	0.1	1311.7	260.6	43.2	92.7	1732.3	20.8
2005/11/24	0.1	1324.9	247.6	57.4	82.8	1736.9	17.2

Table 32: Delta-V required for every launch day.

The delta-V needed for the mission, operational orbit during 1000 days, is 1737 m/s (launch on 24/11/2005).

The delta-V left at EOL will disappear if the spacecraft dry mass is increased by more than 1 Kg.

14. ABBREVIATIONS

AN	Right Ascension of the Ascending Node
AOCMS	Attitude and Orbit Control Monitor System.
AOS	Acquisition Of Signal.
AU	Astronomical Unit (149597870.6 km).
BOM	Beginning of Mission.
CRMA	Consolidated Report on Mission Analysis.
ΔDOR	Delta Differential One-way Range.
DEG	Degrees.
DSM	Deep Space Manoeuvre.
DSN	Deep Space Network.
EOM	End of Mission.
HGA	High Gain Antenna.
ICD	Interface Control Document.
LEOP	Launch and Early Orbit Phase.
LGA	Low Gain Antenna.
LIC	Launcher Injection Correction.
LOS	Loss Of Signal.
LP	Launch Program.
LTF	Linearized time of flight.
LW	Launch Window.
MAO	Mission Analysis Office.
MCC	Mid-Course Correction.
MEEQ2000	Mean Earth Equator 2000.0 reference frame.
MGA	Medium Gain Antenna.
MJD2000	Modified Julian Day 2000.
OBDH	On Board Data Handling.
PDR	Preliminary Design Review.
PE	Argument of Pericenter.
RAAN	Right ascension of ascending node.
RF	Radio Frequency.
SA	Solar Array.
SMA	Semi-major Axis.
SMI	Semi-minor Axis
TA	True Anomaly
TBC	To Be Confirmed.
TBP	To Be Provided.
VOI	Venus Orbit Insertion.
w.r.t.	with respect to.
WP	Working Paper

15. APPENDIX A. DETAILED INTERPLANETARY TRAJECTORY

NOTE: If not explicitly stated, reference system is Mean Earth Equator of 2000.

For electronic form data, please, contact: jose.rodriguez-canabal@esa.int

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```
                DEPARTURE FROM: EARTH
                ----------------------------


MISSION TYPE             :DEEP SPACE MISSION
TYPE OF PROBLEM          :BOUNDARY VALUE PROBLEM
NAME OF CENTRAL BODY     :EARTH
COORDINATE SYSTEM        :MEAN EARTH EQUATOR OF DATE

LAUNCH LIFT-OFF : 2005:10:26   4:44:14.8  ( 2125.1974)

LAUNCH INJECTION: 2005:10:26   6:22:43.2  ( 2125.2658)

INITIAL MASS (KG)      :         1231.700

STATE VECTOR AT INJECTION :
X-COORD (KM)    :        -3839.489
Y-COORD (KM)    :        -2478.046
Z-COORD (KM)    :         5756.069
VX-COORD (KM/S) :       3.18315300
VY-COORD (KM/S) :     -10.11540330
VZ-COORD (KM/S) :       1.94315635

ORBITAL ELEMENTS AT INJECTION :
SEMIMAJOR AXIS (KM)    :       -51371.482
ECCENTRICITY           :      1.130663000
INCLINATION (DEG)      :      51.76682266
R. AS. OF A. NODE (DEG):     115.76860451
ARG. OF PERIC. (DEG)   :      61.17610630
TRUE ANOMALY (DEG)     :      33.21300000

OSCULATING HYPERBOLIC VELOCITY :

MEAN EARTH EQ. 2000
HYP. DEPARTURE VEL. (KM/S):   1.862235 -1.685577 -1.204196
HYPERB. VELOCITY (KM/S):            2.786
EQ.A. DECLINATION (DEG):    -25.613881908
RIGHT ASCENSION (DEG):         -42.149409


MEAN EARTH EQUATOR OF DATE
HYP. DEPARTURE VEL. (KM/S):   1.865109 -1.683151 -1.203142
HYPERB. VELOCITY (KM/S):            2.786
EQ.A. DECLINATION (DEG):    -25.589831131
RIGHT ASCENSION (DEG):         -42.064389


CORRECTION MAN. : 2005:10:28   6:22:43.2  ( 2127.2658)

STATE VECTOR BEFORE CORRECTION MANOEUVRE :
X-COORD (KM)    :       391051.844
Y-COORD (KM)    :      -377457.816
Z-COORD (KM)    :      -238918.090
VX-COORD (KM/S) :       2.02373219
VY-COORD (KM/S) :      -1.83205938
VZ-COORD (KM/S) :      -1.30128221

CORRECTION MAN. TO COMPENSATE FOR DLW.  :
DVX-COORD (M/S) :      12.29043820
DVY-COORD (M/S) :      -6.50293045
DVZ-COORD (M/S) :      -7.11972514
DV MODULUS (M/S):      15.62157040
```

```
MASS AFTER CORRECTION MAN. (KG)  :    1225.52821193

ORBITAL ELEMENTS AFTER CORRECTION MAN.:
SEMIMAJOR AXIS (KM)    :       -50488.536
ECCENTRICITY           :      1.140885696
INCLINATION (DEG)      :      50.52726709
R. AS. OF A. NODE (DEG):     114.78921213
ARG. OF PERIC. (DEG)   :      62.76835125
TRUE ANOMALY (DEG)     :     148.65267960

STATE VECTOR AFTER CORRECTION MAN.:
X-COORD (KM)    :       391051.844
Y-COORD (KM)    :      -377457.816
Z-COORD (KM)    :      -238918.090
VX-COORD (KM/S) :       2.03601012
VY-COORD (KM/S) :      -1.83857830
VZ-COORD (KM/S) :      -1.30840888

OSCULATING HYPERBOLIC VELOCITY :

MEAN EARTH EQ. 2000
HYP. DEPARTURE VEL. (KM/S):   1.880495 -1.698689 -1.213698
HYPERB. VELOCITY (KM/S):            2.810
EQ.A. DECLINATION (DEG):    -25.591663472
RIGHT ASCENSION (DEG):         -42.092139


MEAN EARTH EQUATOR OF DATE
HYP. DEPARTURE VEL. (KM/S):   1.883392 -1.696239 -1.212633
HYPERB. VELOCITY (KM/S):            2.810
EQ.A. DECLINATION (DEG):    -25.567590984
RIGHT ASCENSION (DEG):         -42.007141




               PHASE NUMBER:  1  LEG NUMBER:  1
               -----------------------------------


MISSION TYPE             :DEEP SPACE MISSION
TYPE OF PROBLEM          :BOUNDARY VALUE PROBLEM
NAME OF CENTRAL BODY     :EARTH
COORDINATE SYSTEM        :MEAN EARTH EQUATOR OF DATE

INITIAL EPOCH:  2005:10:28   6:22:43.2  ( 2127.2658)

INITIAL MASS (KG)    :         1225.528

PERTURBATIONS CONSIDERED:
ATMOSPHERIC DRAG:  NO
SOLAR RADIATION :  YES
   REFFLECTIVE COEFF.  :           1.958
   CROSS SECTIONAL AREA:          10.210
THIRD BODY       :  YES
   THIRD BODIES       :  SUN         MOON
NON-SPHERICITY  :  YES -    4 ZONALS &     0TESSERALS

STOP CONDITION:  STOP AT EXIT OF SPHERE OF INFLUENCE OF EARTH
```

```
INITIAL STATE:
X-COORD (KM)    :       391051.844
Y-COORD (KM)    :      -377457.816
Z-COORD (KM)    :      -238918.090
VX-COORD (KM/S) :       2.03601012
VY-COORD (KM/S) :      -1.83857830
VZ-COORD (KM/S) :      -1.30840888

INITIAL ORBITAL ELEMENTS:
SEMIMAJOR AXIS (KM)    :       -50488.536
ECCENTRICITY           :      1.140885696
INCLINATION (DEG)      :      50.52726709
R. AS. OF A. NODE (DEG):     114.78921213
ARG. OF PERIC. (DEG)   :      62.76835125
TRUE ANOMALY (DEG)     :     148.65267960

FINAL EPOCH:  2005:11: 5   6:45:36.9  ( 2135.2817)

FINAL STATE:
X-COORD (KM)    :      1742421.132
Y-COORD (KM)    :     -1584260.839
Z-COORD (KM)    :     -1101855.795
VX-COORD (KM/S) :       1.91015937
VY-COORD (KM/S) :      -1.68739904
VZ-COORD (KM/S) :      -1.20996252

FINAL ORBITAL ELEMENTS:
SEMIMAJOR AXIS (KM)    :       -52081.365
ECCENTRICITY           :      1.231702313
INCLINATION (DEG)      :      33.86526240
R. AS. OF A. NODE (DEG):      93.51936570
ARG. OF PERIC. (DEG)   :      86.04638046
TRUE ANOMALY (DEG)     :     143.46323284


DEPARTURE PARAMETERS ( REFERENCE IS PERICENTRE )
COORDINATE SYSTEM      :  MEAN EARTH EQUATOR 2000

CENTRAL BODY STATE VECTOR W.R.T. SUN
                    POSITION (KM)  :  125012750.066
                                       73893326.146
                                       32035057.648
                    VELOCITY (KM/S):        -16.611
                                             22.881
                                              9.921

DEPARTURE HYPERBOLIC VELOCITY (KM/S)          1.670
                                             -0.447
                                             -1.666
                   MODULE   (KM/S):           2.401

PERICE. ALTITUDE (KM):                      984.447



               PHASE NUMBER:  2  LEG NUMBER:  2
               -----------------------------------


MISSION TYPE             :DEEP SPACE MISSION
```

```
TYPE OF PROBLEM          :BOUNDARY VALUE PROBLEM
NAME OF CENTRAL BODY     :SUN
COORDINATE SYSTEM        :MEAN EARTH EQUATOR 2000

INITIAL EPOCH:  2005:11: 5   6:45:36.9  ( 2135.2817)

INITIAL MASS (KG)    :         1225.528

PERTURBATIONS CONSIDERED:
ATMOSPHERIC DRAG:  NO
SOLAR RADIATION :  YES
   REFFLECTIVE COEFF.  :           1.958
   CROSS SECTIONAL AREA:          10.210
THIRD BODY      :  YES
   THIRD BODIES        :  EARTH
                          VENUS
                          MOON
                          JUPITER
                          SATURN
NON-SPHERICITY  :  NO

STOP CONDITION:  STOP AFTER A GIVEN TIME

INITIAL STATE:
X-COORD (KM)    :    110428738.565
Y-COORD (KM)    :     91009428.722
Z-COORD (KM)    :     39041359.331
VX-COORD (KM/S) :     -18.85966834
VY-COORD (KM/S) :      18.23830152
VZ-COORD (KM/S) :       7.42898338

INITIAL ORBITAL ELEMENTS:
SEMIMAJOR AXIS (KM)    :    126885517.739
ECCENTRICITY           :      0.172066128
INCLINATION (DEG)      :      22.65578028
R. AS. OF A. NODE (DEG):     358.67739320
ARG. OF PERIC. (DEG)   :     213.98701176
TRUE ANOMALY (DEG)     :     189.11608584

FINAL EPOCH:  2005:12:30   2:58:36.1  ( 2190.1240)

FINAL STATE:
X-COORD (KM)    :    -14138421.822
Y-COORD (KM)    :    122647326.732
Z-COORD (KM)    :     51070874.439
VX-COORD (KM/S) :     -29.60739361
VY-COORD (KM/S) :      -7.42367851
VZ-COORD (KM/S) :      -3.37725632

FINAL ORBITAL ELEMENTS:
SEMIMAJOR AXIS (KM)    :    127178511.394
ECCENTRICITY           :      0.169427083
INCLINATION (DEG)      :      22.66503163
R. AS. OF A. NODE (DEG):     358.71173600
ARG. OF PERIC. (DEG)   :     213.70032967
TRUE ANOMALY (DEG)     :     243.56322983
```

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VEX-ESC-RP-5500
Mission Analysis Office
ESA-OPS-GA

```
                 PHASE NUMBER:  2  LEG NUMBER:  3
                ----------------------------------


MISSION TYPE             :DEEP SPACE MISSION
TYPE OF PROBLEM          :BOUNDARY VALUE PROBLEM
NAME OF CENTRAL BODY     :SUN
COORDINATE SYSTEM        :MEAN EARTH EQUATOR 2000

INITIAL EPOCH:  2005:12:30   2:58:36.1  ( 2190.1240)

INITIAL MASS (KG)    :         1225.528

PERTURBATIONS CONSIDERED:
ATMOSPHERIC DRAG:  NO
SOLAR RADIATION :  YES
   REFFLECTIVE COEFF.  :            1.958
   CROSS SECTIONAL AREA:           10.210
THIRD BODY      :  YES
   THIRD BODIES        :  VENUS
                          EARTH
                          MOON
                          JUPITER
                          SATURN
NON-SPHERICITY  :  NO

STOP CONDITION:  STOP AT ENTRY OF SPHERE OF INFLUENCE OF VENUS

MANOEUVRE APPLAYED (M/S)
DV X-COORDINATE:       0.00000115
DV Y-COORDINATE:       0.00000001
DV Z-COORDINATE:       0.00000062
DV MODULE      :            0.000

INITIAL STATE:
X-COORD (KM)    :    -14138421.820
Y-COORD (KM)    :    122647326.717
Z-COORD (KM)    :     51070874.422
VX-COORD (KM/S) :     -29.60739361
VY-COORD (KM/S) :      -7.42367851
VZ-COORD (KM/S) :      -3.37725632

INITIAL ORBITAL ELEMENTS:
SEMIMAJOR AXIS (KM)    :    127178511.347
ECCENTRICITY           :      0.169427083
INCLINATION (DEG)      :      22.66503163
R. AS. OF A. NODE (DEG):     358.71173600
ARG. OF PERIC. (DEG)   :     213.70032974
TRUE ANOMALY (DEG)     :     243.56322976

FINAL EPOCH:  2006: 4: 5  11: 3:28.7  ( 2286.4607)

FINAL STATE:
X-COORD (KM)    :    -45804176.509
Y-COORD (KM)    :    -90223429.776
Z-COORD (KM)    :    -38070232.757
VX-COORD (KM/S) :      32.69040634
VY-COORD (KM/S) :     -17.35750912
VZ-COORD (KM/S) :      -6.87929750

FINAL ORBITAL ELEMENTS:
SEMIMAJOR AXIS (KM)    :    127868609.015
ECCENTRICITY           :      0.174124199
INCLINATION (DEG)      :      22.61915090
R. AS. OF A. NODE (DEG):     358.52605826
ARG. OF PERIC. (DEG)   :     213.82588333
TRUE ANOMALY (DEG)     :      32.46604081



                 PHASE NUMBER:  3  LEG NUMBER:  4
                ----------------------------------


MISSION TYPE             :DEEP SPACE MISSION
TYPE OF PROBLEM          :BOUNDARY VALUE PROBLEM
NAME OF CENTRAL BODY     :VENUS
COORDINATE SYSTEM        :NON-ROTAT. EQ. OF VENUS

INITIAL EPOCH:  2006: 4: 5  11: 3:28.7  ( 2286.4607)

INITIAL MASS (KG)    :         1225.528

PERTURBATIONS CONSIDERED:
ATMOSPHERIC DRAG:  NO
SOLAR RADIATION :  YES
   REFFLECTIVE COEFF.  :            1.958
   CROSS SECTIONAL AREA:           10.210
THIRD BODY      :  YES
   THIRD BODIES        :  SUN
NON-SPHERICITY  :  NO

STOP CONDITION:  STOP AT NEXT PERICENTRE

INITIAL STATE:
X-COORD (KM)    :      -125867.572
Y-COORD (KM)    :       508115.521
Z-COORD (KM)    :      -325177.287
VX-COORD (KM/S) :       0.94892940
VY-COORD (KM/S) :      -3.84529543
VZ-COORD (KM/S) :       2.59523439

INITIAL ORBITAL ELEMENTS:
SEMIMAJOR AXIS (KM)    :       -15203.265
ECCENTRICITY           :      1.418014093
INCLINATION (DEG)      :      88.51401684
R. AS. OF A. NODE (DEG):     104.83625759
ARG. OF PERIC. (DEG)   :     101.58231031
TRUE ANOMALY (DEG)     :     226.55749438

FINAL EPOCH:  2006: 4: 6  21:16:27.5  ( 2287.8864)

FINAL STATE:
X-COORD (KM)    :          302.471
Y-COORD (KM)    :        -1218.867
Z-COORD (KM)    :         6430.329
VX-COORD (KM/S) :       2.59513821
VY-COORD (KM/S) :     -10.45761460
VZ-COORD (KM/S) :      -2.10430825

FINAL ORBITAL ELEMENTS:
SEMIMAJOR AXIS (KM)    :       -15209.818
ECCENTRICITY           :      1.430762083
```

```
INCLINATION (DEG)      :      90.00000000
R. AS. OF A. NODE (DEG):     103.93685685
ARG. OF PERIC. (DEG)   :     101.05070890
TRUE ANOMALY (DEG)     :       0.00000000


MAJOR BODY FLY-BY PARAMETERS ( REFERENCE IS PERICENTRE )
COORDINATE SYSTEM      :  MEAN EARTH EQUATOR 2000

BIDIMENSIONAL IMPACT VECTOR (KM):          1493.244
                                          15512.545

TRIDIMENSIONAL IMPACT VECTOR (KM):        -2176.991
                                           2494.479
                                          15207.508

IMPACT VECTOR:
VAMEE2      1.1377932181711
           -4.3913434917065
            0.88318731859865
VA          4.6215247385687
IMPACT DISTANCE     15563.740679857
3-D IMPACT VECTOR    -2176.9909448808
                      2494.4787418308
                     15207.508348914

CHI     8815.3180292545
ETA     12826.542480066


CENTRAL BODY STATE VECTOR W.R.T. SUN
                    POSITION (KM)  :  -41731901.782
X-COORD (KM)    :          302.471
Y-COORD (KM)    :        -1218.867
Z-COORD (KM)    :         6430.329
VX-COORD (KM/S) :       2.32241418
VY-COORD (KM/S) :      -9.35862002
VZ-COORD (KM/S) :      -1.88316572

ARRIVAL MANOEVURE (KM/S)
                                             -0.273
                                              1.099
                                              0.221

ARRIVAL MAN. SIZE (KM/S)  :      1.15372071

FINAL INJECTED MASS (KG)  :    845.66939304
TOTAL DELTA-V      (M/S)  :        1169.342
```

```
                                                  -92322967.258
                                                  -38895747.983
                            VELOCITY (KM/S):             32.088
                                                        -11.684
                                                         -7.287

ARRIVAL HYPERBOLIC VELOCITY (KM/S):                       1.138
                                                         -4.391
                                                          0.883
                          MODULE   (KM/S):                4.622

PERICE. ALTITUDE (KM):                                  500.000




               ARRIVAL TO: VENUS
               ------------------------


ARRIVAL ORBITAL ELEMENTS:
SEMIMAJOR AXIS (KM)    :       122941.813
ECCENTRICITY           :      0.946708017
INCLINATION (DEG)      :      90.00000000
R. AS. OF A. NODE (DEG):     103.93685685
ARG. OF PERIC. (DEG)   :     101.05070890
TRUE ANOMALY (DEG)     :       0.00000000

ARRIVAL STATE VECTOR:
```

15. APPENDIX B. VISIBILITY EVENTS DURING LAUNCH AND EARLY ORBIT PHASE

DATE	HOUR (UTC)	TIME (HOURS)	EVENT			
05-11-05	05:39:51.5	0	BEGIN			
05-11-05	06:07:44.6	0.46	EXIT	NOV	DURATION	00:27:53
05-11-05	06:07:44.6	0.46	AOS	NEWNOR		
05-11-05	13:41:06.5	8.02	AOS	CEBRER		
05-11-05	15:18:58.0	9.65	LOS	NEWNOR	DURATION	09:11:13
05-11-05	15:55:12.7	10.26	AOS	KOUROU		
05-11-05	21:44:43.4	16.08	LOS	CEBRER	DURATION	08:03:36
05-11-06	02:12:35.1	20.55	LOS	KOUROU	DURATION	10:17:22
05-11-06	02:12:35.1	20.55	ENTRY	NOV		
05-11-06	04:02:35.3	22.38	EXIT	NOV	DURATION	01:50:00
05-11-06	04:02:35.3	22.38	AOS	NEWNOR		
05-11-06	14:06:21.2	32.44	AOS	CEBRER		
05-11-06	15:46:24.7	34.11	LOS	NEWNOR	DURATION	11:43:49
05-11-06	16:01:27.3	34.36	AOS	KOUROU		
05-11-06	21:42:49.1	40.05	LOS	CEBRER	DURATION	07:36:27
05-11-07	02:16:09.0	44.6	LOS	KOUROU	DURATION	10:14:41
05-11-07	02:16:09.0	44.6	ENTRY	NOV		
05-11-07	03:57:44.4	46.3	EXIT	NOV	DURATION	01:41:35
05-11-07	03:57:44.4	46.3	AOS	NEWNOR		
05-11-07	14:08:36.9	56.48	AOS	CEBRER		
05-11-07	15:49:14.7	58.16	LOS	NEWNOR	DURATION	11:51:30
05-11-07	15:59:55.3	58.33	AOS	KOUROU		
05-11-07	21:39:11.4	63.99	LOS	CEBRER	DURATION	07:30:34
05-11-08	02:14:44.1	68.58	LOS	KOUROU	DURATION	10:14:48
05-11-08	02:14:44.1	68.58	ENTRY	NOV		
05-11-08	03:53:22.9	70.23	EXIT	NOV	DURATION	01:38:38
05-11-08	03:53:22.9	70.23	AOS	NEWNOR		
05-11-08	14:07:28.5	80.46	AOS	CEBRER		
05-11-08	15:48:25.6	82.14	LOS	NEWNOR	DURATION	11:55:02
05-11-08	15:57:05.3	82.29	AOS	KOUROU		
05-11-08	21:35:19.7	87.92	LOS	CEBRER	DURATION	07:27:51
05-11-09	02:12:05.6	92.54	LOS	KOUROU	DURATION	10:15:00
05-11-09	02:12:05.6	92.54	ENTRY	NOV		
05-11-09	03:49:13.7	94.16	EXIT	NOV	DURATION	01:37:08
05-11-09	03:49:13.7	94.16	AOS	NEWNOR		
05-11-09	14:05:06.8	104.42	AOS	CEBRER		
05-11-09	15:46:16.0	106.11	LOS	NEWNOR	DURATION	11:57:02
05-11-09	15:53:48.1	106.23	AOS	KOUROU		
05-11-09	21:31:22.8	111.86	LOS	CEBRER	DURATION	07:26:15
05-11-10	02:08:56.0	116.48	LOS	KOUROU	DURATION	10:15:07
05-11-10	02:08:56.0	116.48	ENTRY	NOV		
05-11-10	03:45:07.9	118.09	EXIT	NOV	DURATION	01:36:11
05-11-10	03:45:07.9	118.09	AOS	NEWNOR		
05-11-10	05:39:51.5	120	END			

Table 26 Sequence of visibility events for the first 5 days after launch on 05/11/2005

DATE	HOUR (UTC)	TIME (HOURS)	EVENT			
05-11-15	04:38:32.8	0	BEGIN			
05-11-15	05:07:19.0	0.48	EXIT	NOV	DURATION	00:28:46
05-11-15	05:07:19.0	0.48	AOS	NEWNOR		
05-11-15	12:21:26.7	7.71	AOS	CEBRER		
05-11-15	14:04:25.4	9.43	LOS	NEWNOR	DURATION	08:57:06
05-11-15	14:44:52.3	10.11	AOS	KOUROU		
05-11-15	20:46:42.7	16.14	LOS	CEBRER	DURATION	08:25:15
05-11-16	01:05:02.3	20.44	LOS	KOUROU	DURATION	10:20:10
05-11-16	01:05:02.3	20.44	ENTRY	NOV		
05-11-16	02:58:52.3	22.34	EXIT	NOV	DURATION	01:53:49
05-11-16	02:58:52.3	22.34	AOS	NEWNOR		
05-11-16	12:45:11.9	32.11	AOS	CEBRER		
05-11-16	14:31:17.6	33.88	LOS	NEWNOR	DURATION	11:32:25
05-11-16	14:50:21.4	34.2	AOS	KOUROU		
05-11-16	20:45:16.7	40.11	LOS	CEBRER	DURATION	08:00:04
05-11-17	01:08:24.4	44.5	LOS	KOUROU	DURATION	10:18:02
05-11-17	01:08:24.4	44.5	ENTRY	NOV		
05-11-17	02:54:01.6	46.26	EXIT	NOV	DURATION	01:45:37
05-11-17	02:54:01.6	46.26	AOS	NEWNOR		
05-11-17	12:46:56.9	56.14	AOS	CEBRER		
05-11-17	14:33:53.8	57.92	LOS	NEWNOR	DURATION	11:39:52
05-11-17	14:48:42.4	58.17	AOS	KOUROU		
05-11-17	20:41:56.4	64.06	LOS	CEBRER	DURATION	07:54:59
05-11-18	01:06:59.4	68.47	LOS	KOUROU	DURATION	10:18:17
05-11-18	01:06:59.4	68.47	ENTRY	NOV		
05-11-18	02:49:49.0	70.19	EXIT	NOV	DURATION	01:42:49
05-11-18	02:49:49.0	70.19	AOS	NEWNOR		
05-11-18	12:45:31.9	80.12	AOS	CEBRER		
05-11-18	14:32:56.9	81.91	LOS	NEWNOR	DURATION	11:43:07
05-11-18	14:45:49.3	82.12	AOS	KOUROU		
05-11-18	20:38:07.9	87.99	LOS	CEBRER	DURATION	07:52:36
05-11-19	01:04:17.1	92.43	LOS	KOUROU	DURATION	10:18:27
05-11-19	01:04:17.1	92.43	ENTRY	NOV		
05-11-19	02:45:38.1	94.12	EXIT	NOV	DURATION	01:41:21
05-11-19	02:45:38.1	94.12	AOS	NEWNOR		
05-11-19	12:43:01.4	104.07	AOS	CEBRER		
05-11-19	14:30:38.9	105.87	LOS	NEWNOR	DURATION	11:45:00
05-11-19	14:42:25.4	106.06	AOS	KOUROU		
05-11-19	20:34:13.9	111.93	LOS	CEBRER	DURATION	07:51:12
05-11-20	01:01:01.0	116.37	LOS	KOUROU	DURATION	10:18:35
05-11-20	01:01:01.0	116.37	ENTRY	NOV		
05-11-20	02:41:25.7	118.05	EXIT	NOV	DURATION	01:40:24
05-11-20	02:41:25.7	118.05	AOS	NEWNOR		
05-11-20	04:38:32.8	120	END			

Table 27 Sequence of visibility events for the first 5 days after launch on 15/11/2005

DATE	HOUR (UTC)	TIME (HOURS)	EVENT			
05-11-24	03:51:48.8	0	BEGIN			
05-11-24	04:20:46.3	0.48	EXIT	NOV	DURATION	00:28:57
05-11-24	04:20:46.3	0.48	AOS	NEWNOR		
05-11-24	11:15:35.9	7.4	AOS	CEBRER		
05-11-24	13:03:44.3	9.2	LOS	NEWNOR	DURATION	08:42:58
05-11-24	13:48:09.5	9.94	AOS	KOUROU		
05-11-24	20:03:06.3	16.19	LOS	CEBRER	DURATION	08:47:30
05-11-25	00:11:05.6	20.32	LOS	KOUROU	DURATION	10:22:56
05-11-25	00:11:05.6	20.32	ENTRY	NOV		
05-11-25	02:09:03.0	22.29	EXIT	NOV	DURATION	01:57:57
05-11-25	02:09:03.0	22.29	AOS	NEWNOR		
05-11-25	11:35:55.2	31.74	AOS	CEBRER		
05-11-25	13:28:26.9	33.61	LOS	NEWNOR	DURATION	11:19:23
05-11-25	13:52:02.5	34	AOS	KOUROU		
05-11-25	20:01:58.4	40.17	LOS	CEBRER	DURATION	08:26:03
05-11-26	00:13:47.6	44.37	LOS	KOUROU	DURATION	10:21:45
05-11-26	00:13:47.6	44.37	ENTRY	NOV		
05-11-26	02:04:06.0	46.2	EXIT	NOV	DURATION	01:50:18
05-11-26	02:04:06.0	46.2	AOS	NEWNOR		
05-11-26	11:36:31.2	55.75	AOS	CEBRER		
05-11-26	13:30:17.1	57.64	LOS	NEWNOR	DURATION	11:26:11
05-11-26	13:49:56.0	57.97	AOS	KOUROU		
05-11-26	19:58:47.7	64.12	LOS	CEBRER	DURATION	08:22:16
05-11-27	00:12:04.5	68.34	LOS	KOUROU	DURATION	10:22:08
05-11-27	00:12:04.5	68.34	ENTRY	NOV		
05-11-27	01:59:49.0	70.13	EXIT	NOV	DURATION	01:47:44
05-11-27	01:59:49.0	70.13	AOS	NEWNOR		
05-11-27	11:34:35.2	79.71	AOS	CEBRER		
05-11-27	13:28:53.7	81.62	LOS	NEWNOR	DURATION	11:29:04
05-11-27	13:46:48.5	81.92	AOS	KOUROU		
05-11-27	19:55:09.7	88.06	LOS	CEBRER	DURATION	08:20:34
05-11-28	00:09:13.8	92.29	LOS	KOUROU	DURATION	10:22:25
05-11-28	00:09:13.8	92.29	ENTRY	NOV		
05-11-28	01:55:39.4	94.06	EXIT	NOV	DURATION	01:46:25
05-11-28	01:55:39.4	94.06	AOS	NEWNOR		
05-11-28	11:31:47.9	103.67	AOS	CEBRER		
05-11-28	13:26:22.6	105.58	LOS	NEWNOR	DURATION	11:30:43
05-11-28	13:43:18.2	105.86	AOS	KOUROU		
05-11-28	19:51:19.8	111.99	LOS	CEBRER	DURATION	08:19:31
05-11-29	00:05:54.7	116.23	LOS	KOUROU	DURATION	10:22:36
05-11-29	00:05:54.7	116.23	ENTRY	NOV		
05-11-29	01:51:31.7	118	EXIT	NOV	DURATION	01:45:37
05-11-29	01:51:31.7	118	AOS	NEWNOR		
05-11-29	03:51:48.8	120	END			

Table 28 Sequence of visibility events for the first 5 days after launch on 24/11/2005

16. APPENDIX C. DETAILED RESULTS OF THE CAPTURE AND TRANSFER INTO THE OPERATIONAL ORBIT

	Launch Day	26/10	05/11	15/11	24/11
CAPTURE	ΔV (m/s)	1286.4	1281.4	1296.6	1324.9
	TAN (Beg./End deg)	-87.9/82.7	-86.3/85.9	-80.0/95.4	-77.3/99.9
	Hmin (Km)	278.2	303.7	373.4	395.4
	ΔV next apo. (m/s)	0.2	1.1	0.0	0.4
	Duration (sec)	3079.7	3076.4	3096.1	3171.3
	Eclipse (sec)	915.0	1085.0	1110.0	1165.0
	Occultation (sec)	665.0	695.0	655.0	695.0
	β Sun (deg)	39.0	36.4	35.0	33.8
	Period of orbit (days)	6.19	6.19	6.19	6.19
APO. LOW. WITH ME	ΔV (m/s)	268.9	279.4	255.7	247.6
	TAN (Beg./End deg)	-28.3/14.6	-18.9/26.6	-3.7/37.1	4.5/42.8
	Hmin (Km)	250.7	254.2	261.9	280.1
	ΔV next apo. (m/s)	0.1	0.2	0.2	0.3
	Duration (sec)	496.2	516.5	470.8	455.3
	Eclipse (sec)	51.2	231.5	355.8	455.3
	Occultation (sec)	0.0	0.0	0.0	170.0
	β Sun (deg)	28.8	26.1	24.7	23.4
	Period of orbit (days)	1.10	1.06	1.17	1.22
APO. LOW. 2	ΔV (m/s)	13.5	16.3	14.4	18.1
	TAN (Beg./End deg)	-36.3/36.7	-42.9/42.9	-38.7/38.7	-46.8/46.9
	Hmin (Km)	253.0	253.1	253.9	254.8
	ΔV next apo. (m/s)	0.3	-	-	-
	Duration (sec)	879.0	1060.9	936.8	1175.4
	Eclipse (sec)	334.0	460.9	391.8	545.4
	Occultation (sec)	0.0	0.0	0.0	0.0
	β Sun (deg)	25.3	22.7	21.0	19.5
	Period of orbit (days)	1.05	1.00	1.11	1.14
APO. LOW. 3	ΔV (m/s)	13.4	-	14.4	18.1
	TAN (Beg./End deg)	-36.2/36.3	-	-38.4/38.4	-46.4/46.5
	Hmin (Km)	252.8	-	260.8	264.1
	ΔV next apo. (m/s)	-	-	0.8	1.1
	Duration (sec)	871.2	-	931.9	1167.7
	Eclipse (sec)	331.2	-	386.9	542.7
	Occultation (sec)	0.0	-	0.0	0.0
	β Sun (deg)	21.9	-	17.5	15.9
	Period of orbit (days)	1.00	-	1.05	1.07
APO. LOW. 4	ΔV (m/s)	-	-	14.5	19.5
	TAN (Beg./End deg)	-	-	-38.3/38.6	-49.3/49.3
	Hmin (Km)	-	-	253.3	253.8
	ΔV next apo. (m/s)	-	-	-	-
	Duration (sec)	-	-	933.2	1253.8
	Eclipse (sec)	-	-	398.2	588.8
	Occultation (sec)	-	-	0.0	0.0
	β Sun (deg)	-	-	14.1	12.5
	Period of orbit (days)	-	-	1.00	1.00
OPERATIONAL ORBIT	Total ΔV (m/s)	1582.9	1578.4	1596.7	1629.9
	Mass at O.O. (Kg)	723.7	726.6	719.3	716.0
	ω O.O. (deg)	101.1	103.6	103.0	105.7
	ΔV Control (m/s)	40.0	69.0	59.7	82.8
	ΔV Left (m/s)	127.8	137.9	112.0	100.0
	ΔV Margin (m/s)	87.8	68.9	52.3	17.2
	Control end (days)	2250.3	2288.4	2073.2	1239.7
	Orbit end (days)	2254.3	2300.4	2116.3	2154.3