STABILITY OF THE ARIANE 5 ES GALILEO DISPOSAL ORBIT

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ABSTRACT

In 2001, European Union decided the principle of a European global navigation satellite system and signed later on with ESA the Galileo constellation agreement. The deployment strategy of the constellation is foreseen to be done partly with the Ariane 5 ES launcher jettisoning, two by two, four Galileo Satellites on the same orbit. Choice has been made to inject upper stage on a graveyard circular orbit located 300 km below the Galileo operational one. Thus, the risk of the operational Galileo orbit interference by the upper stage at long term has to be dealt with.

Indeed, the orbital parameters of the stage will be naturally modified by terrestrial potential, sun and lunar interaction and solar radiation pressure. Mainly, eccentricity's modification has to be tracked to prevent operational Galileo's orbit interference.

This paper will report the estimation of stability issues for the final orbit of the Ariane 5 upper stage in Galileo mission.

1 INTRODUCTION

1.1 General Galileo presentation

Galileo is the European global navigation satellite system (GNSS). This project is a program of the European Union (EU) and the European Space Agency (ESA). Under civilian control, the objective is to guarantee to everyone a real-time global positioning accuracy under metric range. The Galileo program has been officially agreed on 26th of May 2003.

The two firsts experimental satellites called respectively GIOVE-A and GIOVE-B have been launched in 2005 and 2008. These satellites tested critical Galileo technologies and secured the Galileo frequencies within the International Telecommunications Union.

On the 21st of October 2011, the two first operational satellites of the constellation have been orbited with a Soyuz launcher from the Kourou space port of French Guyana. These two satellites have been followed one year later, on the 12th of October 2012, by two other operational satellites. These four satellites, representing the In-Orbit Validation (IOV) constellation, have been designed to validate the Galileo concept in space as the related ground infrastructure on Earth. Four operational satellites are considered as a minimum for satellite navigation principle validation. On the 12th of March 2013, a first demonstration has been performed with these four satellites giving their first positioning.

The first services will be provided when the constellation will be completed by 14 additional satellites to reach Initial Operational Capability (IOC). This is foreseen to happen by mid-decade. The first four satellites of the in orbit validation phase are also deployed as part of the initial operational constellation.

Finally, the entire Galileo service will be available after deployment of 12 supplementary satellites reaching the Full Operational Capability (FOC) constellation by 2020. The full system will consist of 30 satellites, control centres in Europe and a network of sensor stations and uplink stations installed around the globe.

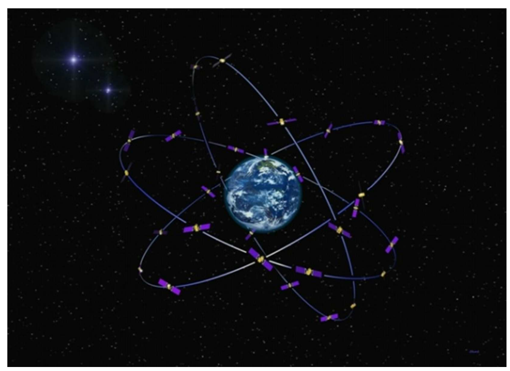

Figure 1. Galileo complete constellation illustration

The definition, the development and the In-Orbit Validation phases of the Galileo program are carried out by ESA and co-funded by ESA and the EU. The FOC phase will be funded by the EU and managed by the European Commission. The Commission and ESA have signed a delegation agreement by which ESA acts as design and procurement agent on behalf of the Commission. A contract for the provision of 14 satellites between 2012 and 2014 was signed with OHB (DE) on the 27th of January 2010 and an additional order of 8 satellites was placed with OHB on the 2nd of February 2012.

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The whole constellation will enclose thirty 700kg-satellites that have a lifetime of 12 years. They will be distributed on three orbital plans at an inclination of 56 degrees to the equator. Each orbital plan will be occupied by 10 satellites: 9 operational ones and the last as a spare. Orbits will be circular at 23 222 km altitude above the Earth. The Galileo's Walker Delta pattern will therefore be "56°:27/3/1".

The orbital period of each satellite will be a little more than 14 hours. A this altitude and with this period, the Galileo Constellation will be able to cover latitudes higher than 75° and from most locations, six to eight satellites will always be visible, allowing positions to be determined very accurately.

ESA entrusted the deployment of the 30 Galileo satellites to Arianespace. The two first flights have been made using the Soyuz launcher from French Guyana. The first flight that has placed two satellites on the first orbital plan was the also the first flight of the Russian launcher taking off from the European spaceport of Kourou. The second flight has placed two more satellites on a second orbital plan.

Arianespace is responsible for the 26 next Galileo satellite launches. The provision of 5 Soyuz launchers has been signed on the 27th of January 2010. Each of them will inject two more satellite. Another launch solution has been held: using Ariane 5 launcher to inject satellites four by four. The adaptation of Ariane 5 has been contracted to EADS Astrium on the 2nd of February 2012, co-funded by EU and ESA. The first Ariane 5 E/S Galileo version has been booked on the same day to be launched in 2014 and two launch options have been signed for 2015/2016. The complete deployment will therefore be a mixture of dual and multiple launches all taking off from Kourou. The next one will be a Soyuz adding two more Galileo-sat on the third orbital plane.

This paper concerns the Ariane 5 E/S version dedicated to Galileo mission and its strategy to inject four satellites each flight. CNES Launcher Directorate gives a technical support to ESA managing the launcher adaptation.

1.2 The Ariane 5 E/S Galileo mission

The Ariane 5 launcher is made of many variants: "ECA" has a cryogenic upper stage and is used for GTO classical dual launches; "ES" (meaning "Evolution Storable") is used to launch the ATV Vehicle to the International Space Station. This last variant is equipped with a re-ignitable upper stage called "EPS" (meaning "Étage à Propergol stockables" ≡ "Storable Propellant Stage"). This stage uses N2O4 and MMH propellants.

The Ariane 5 that will be used to launch Galileo satellites is the "ES" variant. The launcher is being adapted to carry away four satellites at a time to a Medium Earth Orbit (MEO). Main modifications concern the payload delivery system.

The Galileo mission for Ariane 5 is designed to jettison the satellites two by two on the same orbit. This orbit is defined as a circular one inclined from 54° to 58° on the equator. The effective inclination will be defined when the exact date of launch will be chosen depending on the inclination **modification** of the already operational constellation at the precise launch date **due to its natural oscillation**. The aimed orbit altitude is 22922 km. This means that the launcher orbit is 300 km below the operational Galileo constellation orbit. Thus, each Galileo satellite will have to increase its own altitude to ease its way to the operational traffic at an altitude of 23222 km. The Ariane 5 upper stage will be leaved on the injection orbit (circular one at 22922 km).

To realise this launch mission, Ariane 5 will lift-off from Kourou, French Guyana, and realise a first propelled phase where the two solid boosters will be jettison before the end of the first main stage propulsion cut-off. The second stage (the EPS one) will allow meeting an intermediate orbit defined as an elliptical one with an apogee altitude of 22922 km and a perigee altitude optimised for performance maximization (around 350 to 400 km depending on the exact inclination of the aimed orbit). Then, a long coasting phase will take place during approximately 3 hours. When the upper composite will reach its orbital apogee, the EPS will be re-ignited to circularize the orbit and increase the perigee up to 22922 km. This second boost will last a few minutes. After the last cut-off, orientation manoeuvres will be performed to jettison the four payloads two by two in proper attitudes. Then, a collision and contamination avoidance manoeuvre will be performed. Finally the upper stage and its attitude control system will be fully passivated and the upper composite will remain on the lower graveyard orbit plane.

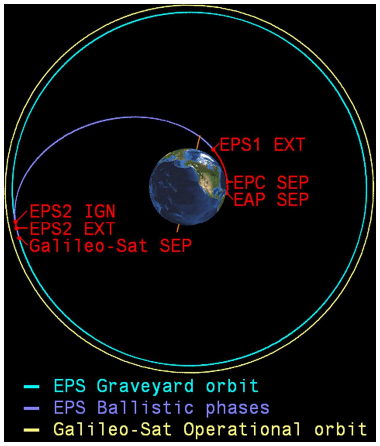

Figure 2. Ariane 5 E/S Galileo mission

1.3 Presentation of the study issue

As seen earlier, the upper stage of Ariane 5 will be injected on a graveyard orbit defined as circular and 300 kilometers below the Galileo constellation operational one. That means that rocket body debris will be placed close to Galileo-Sat and won't be any more controlled after the end-of-life disposition.

It is proven that all orbital objects' motions are influenced by many natural phenomena that induce perturbations with respect to (*wrt*) the "keplerian orbit". For MEO orbits, these phenomena can be lunar or solar attractions, solar radiation pressure or terrestrial gravity irregularities. Their influence could be seen on each orbital parameter of the object.

Every Galileo-Sat will have to deal with these effects during all their operational lifetime using their own control systems. The upper stage of Ariane 5 that will be left on the graveyard orbit will passively be submitted to these effects. Its initial orbit won't necessarily be constant neither stable.

One requirement for the launcher is to ensure that the passivated upper composite will not cross the Galileo operational orbit within 100 years. That means that the graveyard orbit reached at the mission's end should be stable enough to guarantee the protection of the constellation operational altitude within 100 years.

Hereafter, a short recall of different studies will synthesize the main factors that impact the orbital stability close to Galileo orbits. This will conclude that one of the main issues is the accuracy reached at the injection of the object. Then, it will be presented what are the different contributors to the injection accuracy for Ariane 5.

The study presented after has used the semi-analytic orbit propagator tool developed by the CNES called STELA (see [19]). This tool has been used to realize several Monte Carlo studies to analyze the impact of different orbit and accuracy budgets of the Ariane 5 launcher on the requirement to protect the constellation operational altitude within 100 years. These Monte Carlo will also be reported.

2 Theoretical analysis of natural phenomena on MEO

The main objective of this paper is not to describe precisely the perturbation theory on Medium Earth Orbit. This has been done previously in many publications as in [1] to [6]. Based on these publications, main phenomena are recalled hereafter to understand the general context of the present study.

Many kind of orbital perturbations can be listed. Unfortunately, these perturbations are coupled and cannot be independently considered. Hereafter are listed the main long-period or secular perturbations because this paper is dealing with MEO long term evolution.

2.1 RAAN and inclination perturbations

Right Ascension of Ascending Node (RAAN ≡ 'Ω') and inclination ('i') are linked to each other. This can be easily observed in the simplified expression of the terrestrial potential (reduced to the J_2 harmonic) producing a secular drift of 'Ω':

$$\dot{\Omega}_{J_2} = -\frac{3}{2} J_2 \left(\frac{a_e}{a}\right)^2 \sqrt{\frac{\mu}{a^3}} \cos i \qquad (1)$$

Thus, the drift of 'Ω' is directly linked to 'i' and 'a' (which is the semi-major axis).

To this perturbation, another drift of 'Ω' has to be added because of the lunar and solar potentials. This additional drift is coordinated to the previous one (cumulative effect), approximately two times bigger for the moon than for the sun and has a period of 37.5 years. Moreover, the moon-sun perturbations induce an oscillation of the orbital inclination with a large amplitude strongly linked to the RAAN value and a period of 37.5 years. The mean value of 'i' is not much impacted.

The solar radiation pressure has a negligible effect on 'Ω' and 'i' compared to the previous perturbations.

2.2 Semi-major axis and position in orbit perturbations

The semi-major axis is periodically perturbed by the terrestrial potential (zonal terms as well as tesseral ones). Moon-sun perturbations have short periodic effects only. Nevertheless, a coupling effect has been highlighted between these two perturbations. This coupling effect is strongly dependent on the initial value of 'a'.

The solar radiation pressure has a negligible effect on 'a' wrt the impact of this coupling effect.

A secular drift is induced by gravitational forces and at a lesser extent by the solar radiation pressure on the position in orbit (α). Periodical effects can be seen with a period of 37.5 years and amplitude of tens of degrees. These effects are also dependent of 'Ω'.

2.3 Eccentricity and argument of perigee perturbations

As well as the RAAN, the argument of perigee 'ω' is subject to a drift because of zonal terms of the terrestrial potential. The order of magnitude of this drift is approximately 360° in 80 years for the altitude and inclination of the Galileo constellation.

Concerning eccentricity, it has been shown that 'e' can be unstable and can significantly grow over several decades. These important variations are strongly dependent on the initial eccentricity, argument of perigee and RAAN. Moreover, the occurrence of these variations is directly linked to the inclination of the orbit. They can be attributed to a resonance effect between Sun-Moon perturbation and the secular component of the J2 Earth gravitational harmonic. The main solutions proposed in literature to minimize these coupling effects are a modification of the orbital inclination or a minimization of the initial eccentricity of the orbit.

This important variation of eccentricity in time is the main reason of the orbital instability and the main factor leading to the Galileo operational orbit being perturbed by the graveyard orbit of the Ariane 5 upper stage.

2.4 IADC recommendation

All these perturbations and mainly the important variation of eccentricity have been identified and IADC edited a disposal solution to reduce the risk of perturbing the operational orbit:

- Raise the orbital altitude and reduce the eccentricity to below the maximum tolerable for the achieved altitude gain. A guide to the approximate relationship between these values is given by:

$$e \leq 0.000021.\Delta H - 0.0025 \qquad (2)$$

- Target the argument of perigee so that $2\omega + \Omega \approx 90°$, if required (certain combinations of the argument of perigee and the right ascension of ascending node are more stable than others).
- Passivate the spacecraft, so that all on-board sources of stored energy are depleted. As part of this process, manoeuvres could be performed to move the orbital inclination away from 56° (56° is the worst inclination wrt orbital stability for Galileo-type orbits).

3 ARIANE 5 ORBITAL INJECTION ACCURACY

3.1 General information

ARIANE 5, as well as any launch vehicle, cannot guarantee a perfect orbital accuracy injection for Payloads. Accuracy is driven by various phenomena such as: navigation (incl. sensors) performance, guidance and control performances, and upper stage engine tail-off dispersions.

Sensors performance is the main contributor to injection inaccuracy, as ARIANE 5 implements inertial navigation. Moreover, this system was initially designed for short duration missions, limiting inertial drift phenomena: re-use constraints for Galileo MEO longer missions, makes it more critical.

To limit the impact of inertial drift during long coasting phases, a specific strategy has been put in place, similar to the one applied for A5E/S-ATV missions:

- Use of accelerometers for navigation is inhibited during long ballistic phases, and LV orbit is only propagated for navigation using an on-board model of Earth gravity (J2 is included, as the other harmonics have only negligible effect) ;
- Barbecue motion is performed during ballistic phase, around launch vehicle longitudinal axis, with alternated rate.

Both strategies allow not integrating the accelerometric errors during long durations (were no acceleration apart gravity is supposed to be applied to launch vehicle) and to mean the effect of the gyrometric errors (thanks to barbecue alternation).

3.2 Mission requirements

Galileo mission requirements in terms of orbital inaccuracy at PL injection have been specified in [12] and are presented in Tab. 1: they partly account for the objective of limiting the risk of interference between Launch Vehicle upper stage on its graveyard orbit and Galileo operational satellites. One should nevertheless remind that some maneuvers are performed by the launch vehicle after payloads separation (Collision and Contamination Avoidance Maneuver for instance) and that all propellant tanks are depleted at the end of mission. All these events (and their associated uncertainties) impact the characteristics of the graveyard orbit on which the launch vehicle will remain.

	3σ max. inaccuracy at PL injection
a (km)	100
e (-)	0.001
i (°)	0.12
Ω (°)	0.12

Table 1. A5E/S Galileo injection accuracy requirements

The most stringent requirement concerns the eccentricity error, as it is understandable according to IADC recommendation presented above.

One can notice that these requirements are expressed as 3σ values, were σ stands for the standard deviation of a parameter, but:

- they do not account for the correlation that may exist between parameters ;
- for an aimed circular orbit, eccentricity errors are always positive, and thus cannot follow a Gaussian distribution. On the contrary, other parameters errors can be either positive or

negative. Standard deviations are thus not comparable between these parameters and covariance matrix should be defined considering near-circular parameters.

A proper modelization of all the launch vehicle missions and all phenomena impacting orbital accuracy up to the end of mission is thus necessary to quantify (using adapted set of parameters) the actual domain of reachable graveyard orbits of the upper stage, with a given reliability.

Such work is presented in chapter 4 hereafter.

3.3 Considerations on inertial sensors performances

ARIANE 5 implements class-1 Inertial Measurement Unit (IMU) for inertial navigation (see [13] for typical IMU characteristics). Two equipment are embedded for reliability purpose: one is used nominally and the other one used in back-up, in case of failure. These equipments are designed and produced by Thales Avionics. They include a 3-axis gyrolaser designed by Thales and 3 QA-3000 accelerometers from Honeywell (see [14]).

Preliminary orbital accuracy computations have demonstrated that the main contributors to the eccentricity and semi-major axis errors at injection are the errors on the longitudinal accelerometer of the IMU, mainly bias and scale factor. These errors are indeed accumulated during the propelled phases of the mission and directly impact the estimated achieved velocity.

Gyro errors mostly impact the angular characteristics of the injection orbit, such as inclination and RAAN.

Reference performances of the IMU (defined by Thales and derived from QA-3000 accelerometers datasheets, see [14]) are considered for qualification purpose and orbital accuracy quantification.

Actual performances of the produced IMUs (and especially their accelerometers) are generally much better, as discussed in [15] and [16]. In particular, ageing of the accelerometers appears to have less impact than in reference documents.

Aforementioned information and also return of experiment of the ARIANE 5 GTO flights (and of the IMU measured performances), have been used to build a set of reduced dispersions of ARIANE 5 and assess a more realistic orbital accuracy budget, to be compared to the qualification reference one.

Even though planning constraints have imposed a re-use of ARIANE 5 navigation means as they currently are, improvements could be considered for the future, to improve the achieved orbital accuracy:

- Since two IMUs are embedded on ARIANE 5, improvement could be achieved (in case of no failure) using the mean of the measurements for navigation. If one considers IMU errors as random parameters, their impact on the mean of 2 IMU measurements could be reduced by a factor $\sqrt{2}$;
- Improvement could also be brought by hybridation of the IMU measurements with GNSS measurements. Various methods exist for hybridation that allow limiting the effect of inertial drift (see [17]). In the next chapter, pessimistic hypotheses of a loose hybridation have been considered for performance computations.

Four orbital accuracy budgets have thus been computed at the end of A5E/S mission, for the launcher stage remaining on graveyard orbit, to allow comparison of their impact on the long term interference with Galileo satellites operational orbit:

- reference budget considering IMU qualification data ;
- “realistic” budget considering reduced IMU errors, but still with margins wrt A5E flight return of experiment ;
- improved budget considering the possible use of the mean of 2 IMUs for ARIANE 5 navigation ;
- Improved budget considering the possible use of GNSS for IMU measurement hybridation. This budget has been computed considering pessimistic hypotheses for GNSS accuracy, and hybridation approach.

4 TOOLS AND METHODS USED

4.1 Orbital accuracy computations at the end of ARIANE 5 mission

All results presented in this paper have been obtained by CNES DLA in the frame of cross check of industrial activities, required by ESA to secure the AE/S development.

Methods and tools applied are thus fully independent of the ones used by ARIANE 5 Prime contractor, ASTRIUM-ST.

Launch vehicles trajectories for 3 different aimed inclinations (54, 56 and 58°), have been computed using OPTAX tool, which is CNES reference optimization tool, already validated in the frame of past developments (ARIANE 4, ARIANE 5 and VEGA) (see [18] for more details).

These 3 trajectories and the sensors performances associated to the 4 scenarios listed above have been used as input to OCEANIDES tool. This tool is CNES reference one for navigation accuracy computations already validated in the frame of past developments too.

OCEANIDES navigation accuracy computations are based on covariance matrix propagation method. Errors are propagated along the trajectory, making the hypothesis that all IMU sensors errors (including bias, scale factors, non-linearity, misalignments, etc.) are independent random variables, following normal distribution.

Results provided by OCEANIDES are covariance matrices at the upper stage injection. They have been post-treated to add other contributions to injection accuracy (such as upper stage tail-off impulsion dispersions) and to propagate them up to the end of mission, considering the impact of ballistic phase maneuvers and propellant depletion, and the associated dispersions. Obtained covariance matrices at the end of mission have then been used as to generate initial conditions for STELA-extrapolator.

Fig. 3 compares the standard deviations obtained at the end of mission for each orbital parameter, for a 58° inclination orbit, considering the 4 navigation scenarios. Value of 1 corresponds to the standard deviation obtained with reference scenario.

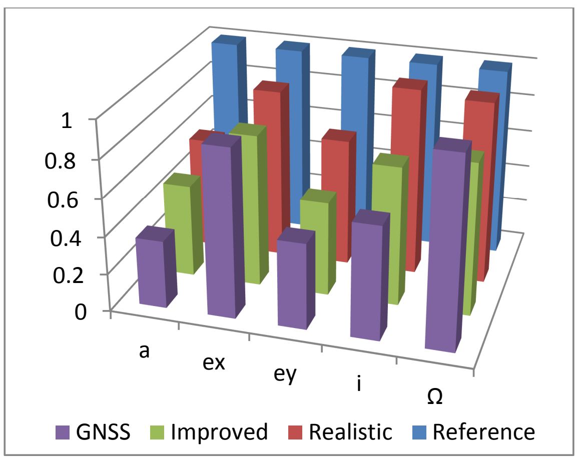

Figure 3. Orbital parameters standard deviations at the end of mission for the 4 considered scenarios

One can see that pessimistic hypotheses for GNSS hybridation make it favorable only for some of the orbital parameters. Improvement is nevertheless brought for semi-major axis, inclination and e_y parameter, which is more dispersed than e_x.

4.2 STELA

A reference tool called STELA implementing the dynamical models to extrapolate orbital parameters in time has been developed by CNES and has been presented in [7] and [8]. STELA is a reference tool in the frame of the French space act but is also usable in mission design and advanced studies. The STELA tool implements a semi-analytical approach that ensures short computation times (about one to two minutes for 100 years of propagation). Semi-analytical approach consists in numerical integration of equations of motion, where the short periodic terms have been removed by means of averaging. This allows the use of a very large integration step size (24h typically), reducing significantly the total amount of computation time.

The averaging approach follows methods developed in the theory of mean orbital motion by Deleflie and Metris presented in [9] and [10].

STELA has been validated by comparison with CNES reference numerical propagators (PSIMU, ZOOM: software used for Precise Orbit Determination and in spacecraft operations) which take into account a complete model of perturbations. See [11] for details on STELA and its validation process.

4.3 A Monte-Carlo approach

In the present analysis, a Monte-Carlo (MC) statistical study has been performed to evaluate the evolution of numerous orbits possibly reached by the Ariane 5 upper stage at the end of the Galileo mission. Each case has been extrapolated using STELA tool. The initial conditions represent possible orbits reached by the upper stage after the commercial mission and were generated through drawings of orbital conditions based on covariance matrices described above.

Three cases have been studied, each of them representing one trajectory on different inclinations. As specified in [12], the aimed orbit will have an inclination between 54° and 58°. Thus, three trajectories have been computed by CNES at 54°, 56° and 58°.

The operational Galileo orbit has an altitude of 23222 km. Thus, it is also specified in [12] that the aimed orbit of the Ariane 5 upper stage (which will also be its graveyard orbit) has an altitude 300 km below meaning 29222 km. This altitude is the one reached by the three nominal trajectories used in the present study.

The RAAN is fixed because the plan to reach is defined by the already launched satellites. Wrt this RAAN, the launch time is optimized on performance constraints on each trajectory.

Finally, all the orbital parameters have been dispersed wrt covariance matrices described above.

The initial date of the extrapolation has been dispersed uniformly over 1 year from the 15th of December 2014 (known as the aimed date for the first Galileo launch on Ariane 5). This allows demonstrating no real date-dependence of the extrapolation as it will be shown hereafter.

Uniform distribution has been applied to the initial Ariane 5 upper stage's mass (±20% around the nominal one estimated by ESA). This distribution covers a dispersion of the whole ballistic coefficient "S.Cx/M".

Twelve MC have been realized, each of them considering 1000 cases. Every 12000 apogee evolution has then been analyzed to check if it crosses the operational altitude of the Galileo Satellites over 100 years.

5 STUDY RESULTS

Main results of the MC study are presented in Tab.2. The four first lines of tab.2 sort the 12000 cases by their scenarios whatever the initial inclination. The three last lines sort the 12000 cases by their initial inclination whatever the navigation scenario. "Earliest date" means the shortest extrapolation duration wrt the concerned case that produce an apogee growth above the operational altitude of Galileo Satellites.

	Nb of crossing cases	**% of cases**	**Earliest date** [years]	**Mean date** [years]
Ref	419	14.0	43.0	84.2
Realistic	224	7.5	54.6	88.0
Improved	145	4.8	66.6	90.5
GNSS	147	4.9	66.9	90.4
54°	119	3.0	52.0	86.5
56°	540	13.5	43.0	86.0
58°	276	6.9	52.3	89.4

Table 2. Main MC results

5.1 Analysis by orbital accuracy injection

It can be seen that the reference accuracy injection produces 14% of cases that will encounter the critical altitude of 23222 km within 100 years. This percentage is almost divided by two when considering more realistic injection accuracy for A5E/S and divided by three when considering an improved accuracy. It can also be seen that "Improved accuracy" (using 2 IMUs) or "GNSS accuracy" are producing very similar results.

Furthermore, these results show that the earliest increase of apogee altitude to 23222 km is seen after 43 years of extrapolation for the worst draw in the reference case. This can be postponed to 66.9 years improving the navigation system. As these cases represent extreme initial conditions, the mean date should be taken into account: this date represent the average date of the first apogee crossing the critical altitude of 23222 km among all the cases crossing this altitude (it excludes all the cases that remain below 23222 km within 100 years). Considering this mean date, it can be said that using an improved navigation system postponed the critical date after 90 years of extrapolation.

To synthesize these results, it can be noticed that the injection accuracy directly conditions the duration that the extrapolation spend below the critical altitude.

5.2 Analysis by aimed inclination

The analysis of the results sorted wrt the initial aimed inclination shows that 56° is the worst inclination wrt the time spent below the critical altitude. 13.5% of cases have an extrapolated apogee altitude that grows above the operational Galileo Satellite altitude.

Comparing cases at 54° to those at 56°, 4.5 times less cases are increasing over 23222 km within 100 years. When aimed inclination is 58°, the number of critical cases is between the two other configurations.

It can also be reported that the shortest extrapolation duration before the apogee altitude equals the Galileo-Sat operational one is 9 years worst at 56° than 54° or 58°. On the other hand, considering the mean date before the critical altitude is reached, values are very close for the 3 cases (but this average is made on 2 to 4.5 times less values at 58° or 54° than at 56°).

These observations are coherent with [1] to [6]: an orbital instability is captured when inclination is very close to 56°. 54° seems to be the most comfortable inclination wrt the Galileo Satellite altitude protection.

5.3 Complementary analysis

If the same exercise is done over 200 years, it appears that around 40% of cases are critical for inclination close to 54° or 58° and 69% for inclination close to 56°. Mean extrapolation duration before crossing the critical altitude is around 140 years for 54° and 58° inclination orbits and around 128 years for 56°.

5.4 Date of launch impact

The date of launch has been dispersed uniformly among one year in the 12 MC draws. In Fig.4, it can be seen that the maximal apogee altitude reached over 100 years does not depend on the initial date between the 15th dec 2014 and the 15th dec 2015.

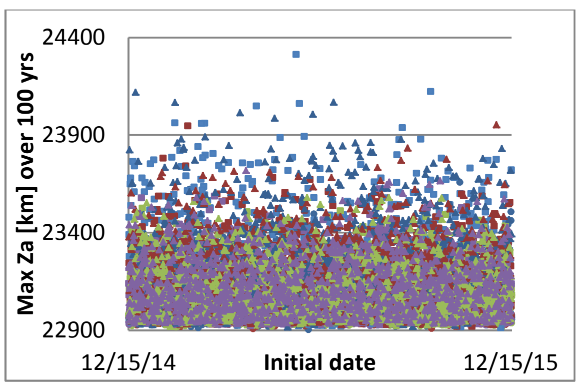

Figure 4. Maximal apogee altitude reached wrt to initial date of extrapolation

5.5 Initial "2ω+Ω" impact

As the nominal aimed orbit is circular, the argument of perigee is not defined. In concrete terms, the reached orbit is not perfectly circular. Thus, ω is defined and randomly distributed. As explained previously, the RAAN is fixed by the already launched satellites.

The IADC recommendation presented above can be found again representing the maximum apogee altitude reached for the 12000 cases wrt the initial angle "2ω+Ω". This representation is seeable on Fig.5.

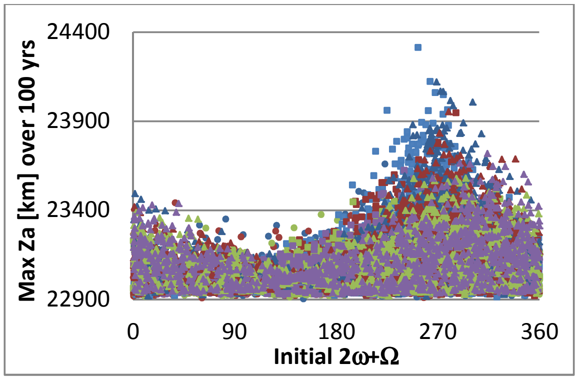

Figure 5. Maximal apogee altitude reached over 100 years wrt to initial angle "2ω+Ω"

With this representation, it can be seen that the worst case is reached when the initial "2ω+Ω" is around 270° and the most stable orbits are concentrated around initial "2ω+Ω" equals ~90°. This important result is coherent with the IADC recommendation and previous studies.

5.6 Initial eccentricity impact

As explained earlier, the initial eccentricity is the main factor to control the evolution of apogee altitude in time: minimizing initial eccentricity induces a minimization of the apogee variation. This can be figured out representing the maximal apogee altitude reached over 100 years wrt the initial eccentricity of the orbit:

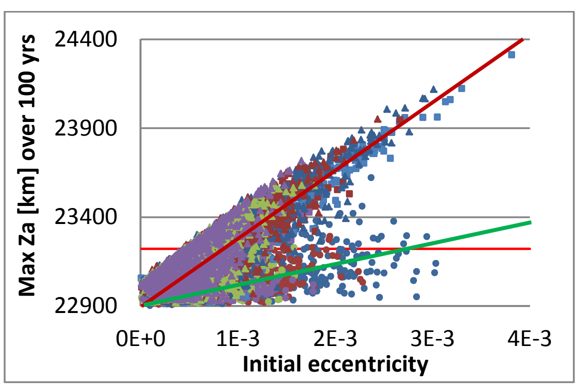

Figure 6. Maximal apogee altitude reached over 100 years wrt to initial eccentricity

Two main results can be seen on Fig.6:

- The lowest the initial eccentricity is, the lowest the apogee growth will be. It is also seen that limiting the initial eccentricity to the critical value of 10^{-3} is a good manner to constrain the growth of perigee below the operational altitude of Galileo Satellites.
- Two trends can be seen: all the drawn points can be divided in two groups each of them drawing a line with two different slopes. Analyzing more precisely the division of points in two groups, it appears that the red line on Fig.6 gathers the results of extrapolations on 56° and 58° inclination while the green line gathers results on 54° inclination.

This last observation can be intensified considering extrapolations over 200 years:

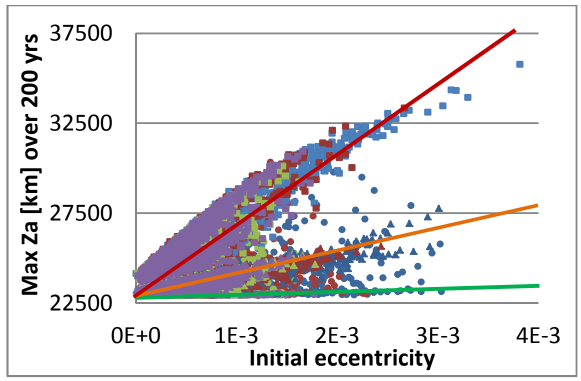

Figure 7. Maximal apogee altitude reached over 200 years wrt to initial eccentricity

On Fig.7, it is seen that results can be divided in three groups, represented by the three lines. It has also been checked that these groups are differentiated by the aimed inclination: the green line groups the 54° aimed inclination; the orange line groups the 58° aimed inclination; the red line groups the 56° aimed inclination.

As a conclusion, the aimed inclination defines the slope of the apogee altitude growth over the extrapolation and the initial eccentricity defines the maximum apogee altitude reachable during the extrapolation.

6 CONCLUSION

This study has shown that the long term stability of the Ariane 5 Upper Stage graveyard orbit in the Galileo mission is directly linked to the eccentricity reached after the passivation and the aimed inclination decided wrt the launch date.

The three IADC recommendations have been checked:

- $e \leq 0.0038$ whatever the navigation system is;

- $2\omega+\Omega = 90°$ is preferred angle to reach;
- A passivation process is planned at the commercial mission end.

It has also been noted that all the orbital inclinations are not equivalent with a fixed initial eccentricity superior to 10^{-3} wrt to apogee altitude evolution.

Finally, the real injection accuracy is expected to be better than the reference case and allow being confident on the orbital stability of the graveyard orbit. Post-flight analysis will allow realizing new studies based on effective reached orbit.

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