OPERATIONAL IMPLEMENTATION OF SPACE DEBRIS MITIGATION PROCEDURES

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ABSTRACT

During the spacecraft lifetime, Astrium supports its customers to manage collision risks alerts from the Joint Space Operations Center (JSpOC). This was previously done with hot-line support and a manual operational procedure. Today, it is automated and integrated in QUARTZ, the Astrium Flight Dynamics operational tool. The algorithms and process details for this new 5-step functionality are provided in this paper. To improve this functionality, some R&D activities such as the study of dilution phenomenon and low relative velocity encounters are going on.

Regarding end of life disposal, recent operational experiences as well as studies results are presented.

1 INTRODUCTION

Astrium Satellites is developing satellite systems, payloads and ground infrastructures for telecommunications, Earth Observation and Science missions. For many customers, Astrium is in charge of the satellite in-orbit delivery. In addition, Astrium provides an in-orbit follow-on support during the entire operational life including Flight Dynamics activities. Astrium has developed, for that purpose, tools and operational procedures to limit the sources of debris both during the operational lifetime and for the end of life mission disposal.

2 DURING SPACECRAFT LIFETIME

2.1 Collisions Risk Management Context

Over the past years, several conjunction alerts have been raised by JSpOC towards Astrium customers. These alerts concern mainly LEO operators. The first alert has been received in November 2009 and their number has been increasing year after year. Since the beginning, Astrium has been supported operators in deciding whether an avoidance manoeuvre is necessary or not, and computing the manoeuvre.

2.2 From a Manual to an Automatic Process

This was previously done using a manual operational procedure based on JSpOC alert message. The first alerts contained very little information as compared to actual Collision Summary Messages (CSM). The operational procedure set-up was simple and adapted to the information contained in the alert: overall distance, 3-D components of separation between the two objects and uncertainties on radial axes in local orbital frame as illustrated in Fig. 1.

Predicted Distances

Primary Object	Secondary Object	TCA	Overall m	Radial (dU) m	In-Track (dV) m	Cross-Track (dW) m	Primary Radial Error (U) m	Secondary Radial Error (U) m
		031 (31 JAN) 12:30Z	772	-139	415	-637	22	20

Figure 1. Example of first JSPOC alerts format

The avoidance manoeuvre was thus decided based on a radial distance minimum criterion. If the radial distance was inferior to the computed minimum distance, an avoidance manoeuvre was recommended as illustrated in Fig. 2.

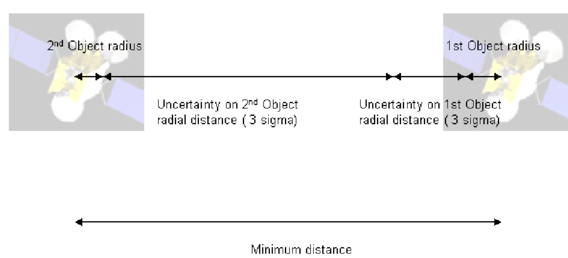

Figure 2. Minimum radial distance criterion

Once the avoidance manoeuvre had been computed, updated orbital ephemerides were sent back to JSpOC in order for them to re-assess the risk. The lessons learned from this operational experience showed that operators could make errors because of the very short time available. Indeed, alerts are sent by JSpOC only 72 hours before Time of Closest Approach (TCA). In the meantime, JSpOC alerts format evolved and the contents were improved a lot. They contain today a lot

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of very useful information such as the position/velocity of the two objects as well as their covariance matrices. Therefore, it was decided to set-up an automatic process both to make the most of the CSM content and to help operators managing smoothly a conjunction alert.

2.3 Automatic Conjunction Assessment and Collision avoidance software

This new functionality was implemented in QUARTZ, the Astrium Flight Dynamics Software. Indeed, Astrium customers are using QUARTZ as their Flight Dynamics Software for routine and orbit maintenance operations. The new Conjunction Assessment and Collision Avoidance functionality is operationally available since mid-2012. It has been delivered to two operators. This function is a 5-steps automatic procedure:

1- Automatic reading of CSM xml file and consistency check with the current best orbit estimate (as part of routine operations, operators update the current orbit on a daily basis)

2- Collision Risk probability computation based on position and covariance information

3- If the probability exceeds a predefined threshold (10^{-3}), an avoidance manoeuvre is proposed N+1/2 orbit before TCA.

4- The avoidance manoeuvre is then implemented (conversion of the impulsive manoeuvre in a start time and duration) and spacecraft platforms specific constraints related to the manoeuvre are checked.

5- The orbital ephemerides (taking the manoeuvre into account) are then generated in a specific format so that they can be processed by JSpOC in order for them to update the conjunction information.

2.3.1 Collision Probability Computation

Most of the conjunctions are high relative velocity cases. Therefore, the computation of the Collision Risk probability shall be at least valid for high relative velocity conjunctions (i.e. > ~20 m/s in LEO). The following assumptions are used:

- The encounter is very short (few ms).
- The relative motion close to the encounter is linear.
- The velocity errors are negligible.

The collision events can then be located in a conjunction plane which is defined perpendicular to the relative velocity of the two objects and conventionally centred on the primary object as presented in Fig. 3.

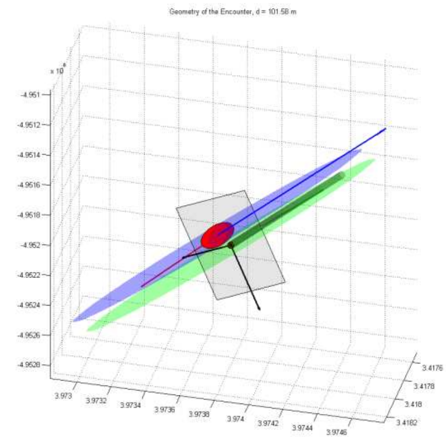

Figure 3. Example of 3-D uncertainties distribution projected in the conjunction plane

Using the steadiness property of the normal law, we can establish that the covariance of the relative position of the secondary object is the sum of primary and secondary covariance. This dispersion is projected into the conjunction plane, giving a 2D combined covariance called Σ_{conj}. The collision probability is then defined in Eq. 1. by the following integral:

$$P_c = \frac{1}{2\pi \det(\Sigma_{conj})} \iint_S \exp\left(-\frac{1}{2} M' \Sigma_{conj}^{-1} M\right) dS \quad (1)$$

Where $\begin{bmatrix} X_S \\ Y_S \end{bmatrix}$ is the mean relative position of the secondary object, $M = \begin{bmatrix} X - X_S \\ Y - Y_S \end{bmatrix}$ and S is the circular impact area whose radius is defined by the sum of R_p and R_s, respectively the first and second object radius. This integral is computed numerically using polar coordinates.

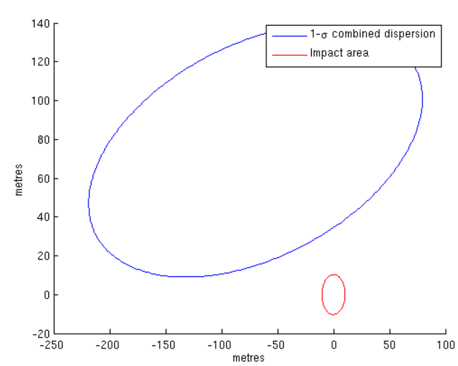

Figure 4. Example of relative dispersion into the conjunction plane and the impact area

2.3.2 Avoidance Manoeuvre Computation

The software aims at computing a tangential manoeuvre of the primary object N+½ orbits *before* TCA in order to increase the radial separation at the conjunction.

N can be chosen equal to 0, 1, 2, etc in order to comply with the operational constraints (mission, station visibilities, CSM updates).

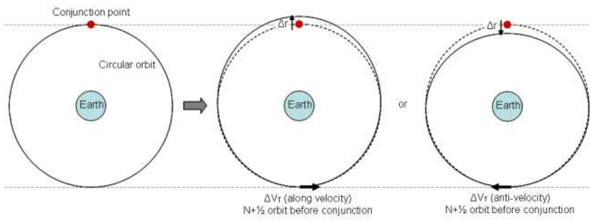

Figure 5. Avoidance Manoeuvre Computation

The manoeuvre minimal magnitude is computed by a dichotomous process, allowing lowering the collision risk probability under the threshold. The algorithm is the following:

1- Compute new primary orbit (monitored satellite) at TCA including the current avoidance manoeuvre.

2- Compute new TCA with the secondary object (debris) and use it to propagate the primary orbit.

3- Compute the secondary orbit at the new TCA using the Clohessy-Wiltshire relative motion equations.

4- Update the primary covariance in position matrix due to the manoeuvre realization errors at the new TCA.

5- Compute the collision risk probability of the two objects at the new TCA.

This algorithm is valid under the main following assumptions:

- The magnitude of manoeuvre is low.
- The efficiency of manoeuvre and the direction error are Gaussian.
- The relative trajectory between old and new TCA is assumed to be circular for the 2 objects.

2.3.3 Iterative and Final Checks

Once the specific ephemerides are generated taking into account the avoidance manoeuvre, they are sent to JSpOC in order for them to re-assess the risk with the updated ephemeris. The avoidance manoeuvre is scheduled and uploaded to the satellite as late as possible before the TCA allowing thus the operators to process potential CSM updates from the JSpOC. If any, new CSMs are thus processed as described previously.

If the collision risk remains confirmed, the prepared TC plan is uploaded to the satellite during the last but one ground station visibility before TCA and the avoidance manoeuvre performed.

2.4 Improvements and Way Forward

The current QUARTZ functionality is able to handle most of the conjunction alerts in Low Earth Orbit. However, as the probability computation is based on the hypothesis of high relative velocity, the low relative velocity encounter cases are not managed yet. Astrium is currently working on that topic. The objective is to work out a systematic way to assess the risk in such situations. Then, the Conjunction Assessment and Collision Avoidance functionality could be extended to geostationary orbit. Indeed, in geostationary orbits, the conjunctions are more likely to be low relative velocity encounters than high relative velocity collision risks.

Another concern is that one shall not miss potential dangerous events because of a high uncertainty on the 2 objects position/velocity. A bad accuracy can artificially lead to underestimate the real risk: the collision probability is mathematically very low (Fig. 6) but the risk is real. This situation is called the dilution phenomenon. As a first step, it is necessary to determine if the computed probability is in the dilution region or not. The second step will be to work out an algorithm so that this specific dilution situation can be managed automatically by the Conjunction Assessment and Collision Avoidance functionality integrated in QUARTZ.

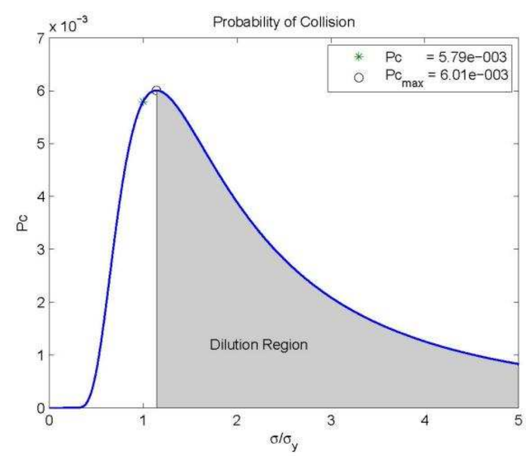

Figure 6. Example of 1-D Dilution Region

3 END OF LIFE DISPOSAL

3.1 Re-orbitation for geostationary satellites

Spacecraft that have terminated their mission should be maneuvered far enough away from GEO so as not to cause interference with space systems still operating in geostationary orbit. The IADC recommends a minimum increase in perigee altitude which takes into account all orbital perturbations:

$$\Delta Perigee = 235km + \left(1000 * CR * \frac{A}{m}\right) \qquad (1)$$

Where CR is the Solar radiation pressure coefficient, A/m the aspect area to dry mass ratio [m2/kg] and 235 km the sum of the upper altitude of the GEO protected region (200 km) and the maximum descent of the re-orbited space system due to moon, sun and geo-potential perturbations (35 km).
In the IADC recommendation there is no mention of the eccentricity of final orbit, but the eccentricity shall be minimized. A small eccentricity will minimise the deviation between the apogee and perigee altitudes which consequently permits a higher relative perigee altitude and will increase the stability of the orbit from moon and sun perturbations.
Astrium has recently re-orbited Nilesat 101 on behalf of Nilesat Company. This satellite is an Astrium E2000 platform launched in 1998. The re-orbitation operations have been performed in February 2013. During Mission Analysis phase, a manoeuvre plan was elaborated to re-orbit while exiting safely the GEO box and keeping sufficient inter-satellite separation relative to NIL102 and NIL201 (collocated at the same longitude). 3 pairs of manoeuvres separated by 12 hours to keep the eccentricity as close as possible to the natural eccentricity circle were proposed as described in Table 1. This corresponds to the minimal manoeuvre plan as re-orbitation shall be guaranteed considering the known uncertainties on remaining propellant.

EPOCH	Duration(s)	dvr(m/s)	dvt(m/s)	dvn(m/s)	direction
05/02/2013 14:00:00	234	1.25411	1.99584	0.00241	East
06/02/2013 02:00:00	235.2	1.26160	2.00775	0.00242	East
06/02/2013 14:00:00	234	1.25621	1.99917	0.00241	East
07/02/2013 02:00:00	234.6	1.26048	2.00597	0.00242	East
07/02/2013 14:00:00	175.2	0.94202	1.49916	0.00181	East
08/02/2013 02:00:00	175.8	0.94584	1.50524	0.00182	East

Table 1. Minimal Nilesat 101 re-orbitation Plan

Given the propellant amount left, 16 maneuvers have been performed. Nilesat 101 perigee altitude reached 718 km above the geostationary arc at the end of re-orbitation phase. Nilesat 101 will not come back in the GEO protected region within 100 years, whatever the hypotheses on spacecraft attitude (driving the perturbation effects on eccentricity)

3.2 Controlled re-entry whenever possible

For missions crossing the LEO region, de-orbitation is the preferred end of life disposal approach; it can be either an uncontrolled or controlled re-entry.
During the past 10 years, Astrium performed 2 controlled re-entries of Telecom satellites following a Proton launcher failure. In addition to this operational experience, Astrium has conducted a R&D study for CNES in 2012 to assess the feasibility of controlled re-entry for different types of orbits and satellite platforms and to identify the key show stoppers.

3.2.1 From an eccentric orbit

In both operational cases, the failed orbit delivered by the launcher was very inclined (~50 degrees) with an apogee altitude of [15000 km, 20000 km], far below geostationary altitude. Several analyses using exotic transfers by the moon where conducted but concluded that it was not possible to save the mission given the amount of propellant available on-board. It was thus decided to perform a controlled re-entry in agreement with the customers. Chemical Telecom satellites on an eccentric orbit after a launcher failure have more than enough propellant to make a controlled re-entry. Moreover the liquid apogee engine allows targeting fictitious perigee altitudes such that the re-entry footprint is quite small. The operational implementation is made taking into account possible AOCS constraints and the actual orientation of the apogee-perigee line. Usually these two constraints are balanced by the huge delta V capacity of geostationary satellites.

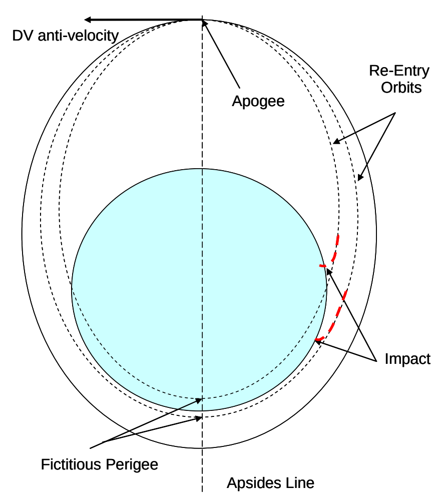

Figure 7. Controlled Re-Entry from an eccentric orbit

To protect populations, only low density zones with almost no air and maritime routes are eligible as impact zones. Such zones are illustrated in Fig. 8. To phase the impact with the selected zone, 2 options shall be considered. Either one can wait until the phasing conditions are met to perform the re-entry manoeuvre or one can perform intermediate manoeuvres to achieve the longitude Rendez-vous. During both Astrium operational experiences, a single manoeuvre was commanded once the phasing conditions were met. The manoeuvre size was driven by the choice of the impact point as illustrated in Fig. 7. According to the respective orientations of apsides line, the first controlled re-entry targeted the South pacific zone whereas the second one was performed in the North Pacific zone with a manoeuvre size precisely tuned to prevent from an impact on the United States whatever the dispersions on the manoeuvre.

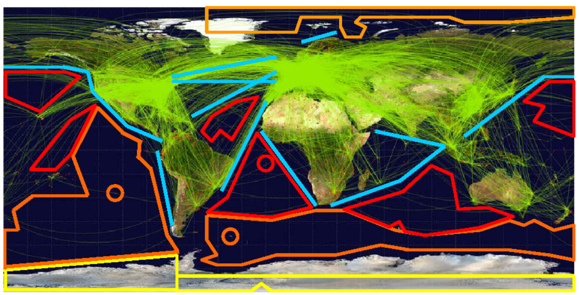

Figure 8. Potential Re-Entry Zones

3.2.2 From a Low Earth Orbit

The controlled re-entry from a Low Earth Orbit is today not achievable with the actual spacecraft platforms. Indeed, the propellant needed to perform the re-entry itself is very often far larger than the propellant need for the mission. For example, a micro-satellite flying at an altitude of 700 km will need at least 160 m/s to perform controlled re-entry whereas its overall capacity is about 70 m/s. Furthermore, the maximum delta V size shall be large enough to enable a large last manoeuvre from the minimum altitude where the AOCS is able to control the satellite platform to the target perigee altitude (it shall be below 50km to ensure the controlled re-entry). Should controlled re-entry become mandatory, spacecraft design should be drastically reviewed and alternative propulsion systems considered. Another alternative to controlled re-entry is semi controlled re-entry. In this case, re-entry footprint is spread on a small number of orbits. However, it is not straight forward to find a phasing such that the ground tracks of 2 to 3 orbits cross only low density regions. This becomes possible if the semi controlled re-entry footprint is limited to one orbit. This is possible for an initial circular orbit of 130-140 km. This option could be interesting in case of electric propulsion in LEO. The level of risk will be higher than a controlled re-entry but should remain smaller than an uncontrolled re-entry.

3.3 Uncontrolled re-entry

To ensure proper end of life disposal, Astrium performs systematically a de-orbitation analysis as part of mission analysis. In the former mission analysis process, a few years ago, the target orbit for natural re-entry was worked out using numerical propagation tools. The limit of this method was that the result was highly depending on the considered hypothesis on solar activity. Now, a new tool, STELA, is available from CNES. STELA is the reference tool in the frame of French Space Law Technical Regulation. Therefore, Astrium now performs its de-orbitation analyses using STELA.

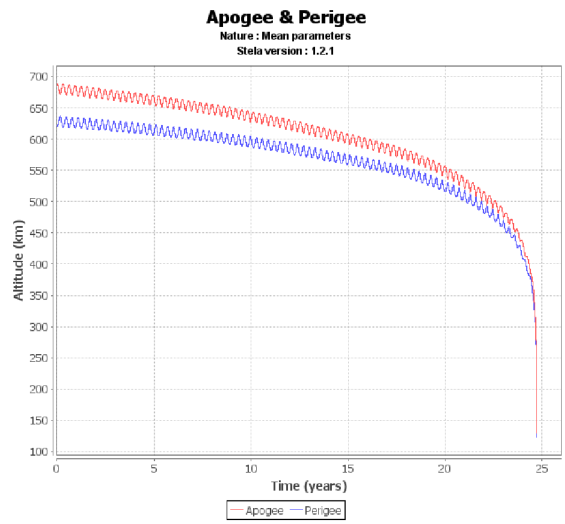

Figure 9. Example of de-orbitation plot from STELA

4 WAY FORWARD

At Astrium, in parallel to mission analyses process and customer operational support already in place, activities are conducted to develop the necessary tools and expertise relevant for Space Debris Mitigation.

Regarding Collision Risks Management, it means to carry on with the work on dilution phenomenon and on low relative velocity encounters in the frame of internal R&D.

Regarding End-Of-Life Disposal, it is necessary to work out new spacecraft design so as to ensure compliance with international regulations both for uncontrolled re-entry (modification of materials) and controlled-re-entry (modification of propulsion system).

5 ACKNOWLEDGMENTS

Some results and figures presented in this paper were worked out in the frame of a 2012 CNES R&T study: “Optimisation of controlled re-entry