A RECONFIGURABLE MARS CONSTELLATION FOR RADIO OCCULTATION MEASUREMENTS AND NAVIGATION

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ABSTRACT

A Martian constellation comprising several micro spacecraft can be exploited to perform radio occultation measurements between constellation satellites through the limb of the Martian atmosphere. This would allow the Martian atmosphere and climate cycle to be characterized with unprecedented spatio-temporal precision, leading to a significant increase in atmospheric knowledge and hence risk reduction for atmospheric entry probes. Additionally, this constellation could be reconfigured at relatively low cost using electric propulsion to provide navigation services to subsequent Mars missions. Thus, with the same radio payload, the constellation has the potential to substantially increase the accuracy of orbit insertion, aero-capture and possible landing of these missions. However, an assessment on the feasibility and potential benefit of such a concept has to be carried out. This paper describes the optimization of the orbital geometry of two constellations (comprising 4 and 8 satellites), which has been performed in order to obtain the maximum number of globally distributed radio occultation events between satellites in the constellation. The resulting constellation geometries have been explored from a navigation performance perspective, Assuming that the navigation aid to incoming spacecraft has to be given only every 26 months, i.e. when a minimum energy launch window occurs between the Earth and Mars, and some interesting constellations of 4 and 8 satellites are presented in the paper. Some reconfiguration strategies have then been proposed and assessed and the feasibility of the concept has been proven.

INTRODUCTION

With the current agencies' space policies on the exploration of the solar system Mars appears to be the ultimate goal for many near-future space missions. More and more missions are planned to fly to Mars as orbiters or landers; however, at present there is no existing and effective system that either totally describes the Martian atmosphere in space and time, or that would provide an efficient and precise aid for a spacecraft during final approach to the red planet.

Evidence on the possibility to perform useful measurements of some planet atmospheric properties by means of inter-satellite links was first given in 1995 when a LEO-GNSS occultation measurements campaign was performed and used to improve the existing numerical models of the Earth atmosphere. The European Space Agency founded, in the last decade, a number of pre-phase A studies (e.g. the Atmosphere and Climate Explorer Plus), but none of them was applied to Mars. On the other hand, the spacecraft going to Mars at present largely rely on the Deep Space Network, which provides sufficient navigation aid as long as the targeting does not require high precision: this would be the case during, for example, an aero-capture or targeted landing. Moreover, the DSN appears to render the spacecraft uncontrollable for the last 13 minutes before arrival at the periapsis due to the control signal time-of-flight to Mars.

Thus, this paper assesses the feasibility of a mission to Mars that has to perform two different tasks, i.e. it searches for optimal orbital configurations of constellations of 4 and 8 micro satellites tasked with alternately performing radio occultation measurements and navigation for incoming spacecrafts. In this paper, the radio-occultation technique is briefly reviewed and an objective function is derived. The sensitivity to the J_2 gravity harmonic for a chosen constellation is then analyzed, and the figures of merit that have to be evaluated to assess navigation performance are introduced. Finally, the numerical methods used to approach the problem are presented, together with the results.

RADIO OCCULTATION PERFORMANCE

Atmospheric measurements

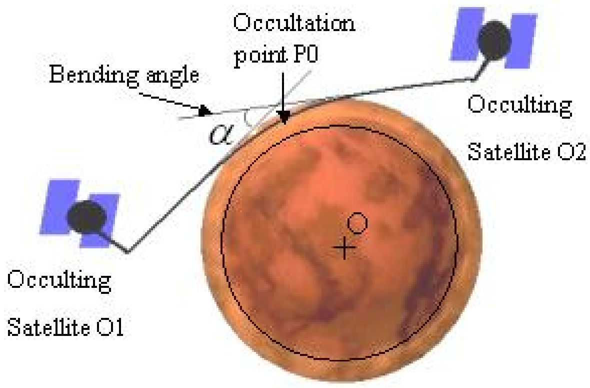

Figure 1: Radio Occultation Technique

The principle of the radio occultation technique (shown in Figure 1) is based on precise dual-frequency phase measurements of a satellite receiver in orbit tracking a setting or rising satellite in the same constellation through the limb of the Martian atmosphere. These measurements are combined with the Doppler shift due to the atmosphere, and the satellite state vector information can be used to obtain the atmospheric bending angles and furthermore vertical profiles of the atmosphere's refractive index. This leads to the characterisation of the atmosphere (pressure, temperature, water vapour, wind, etc) with high vertical resolution and accuracy in clear or cloudy conditions [Walker et al., 2004].

One of the greatest difficulties in obtaining an optimal configuration is the extremely complicated relation between the constellation geometry, i.e. the number of satellites and the set of their osculating parameters, and the position, length and number of occultation events per Martian day (sol). The aim in this paper was to maximise both the number and uniform spatial distribution of occultation events per couple of sols.

One occultation is called an occultation event if it lasts more than 50 seconds and satisfies the constraints given by the following formulas: $(OP0) \perp (O1O2)$, $P0 \in [O1O2]$ and $0 < altitude(P0) < 300km$. In order to define some index describing the spatial distribution of the occultation events, the Martian surface was divided into N=40 sectors comprised between two latitudes λ by taking care that:

$$\sin\lambda_{i+1} - \sin\lambda_{i-1} = 2\sin\lambda_i$$

in order that the area of each sector is equal, i.e. S=3625300 km^2. For a given geometry the number of occultation events occurring in each sector was counted and the standard deviation σ was introduced as a measure of the distribution between sectors. Each geometrical configuration was therefore judged on the basis of both the total number of occultation events and of the value of this standard deviation. Though this problem should ideally be tackled with a multi-objective approach, it was decided to solve the problem with a unique formula combining the two objective functions in a suitably weighted fashion, hence yielding a scalar objective function. The chosen objective function has the form:

$$J(n,\sigma) = \frac{n}{\sigma^p}$$

with p =3 or 2 for respectively a 4 or 8-satellite constellations and n the total number of occultation events for two Martian days. This was found to be a very good compromise between a high number of occultation events and the spatial distribution.

Mars-oblateness effects

The effects of Mars oblateness were studied in order to define how the above objective function evolves with them, and whether they would necessitate station-keeping maneuvers. To allow us to concentrate on oblateness effects, the altitudes of the satellites were constrained above 300 km, such that drag effects are negligible. Two oblateness effects are distinguished, the first one being a regression of the nodes. The second perturbation effect is an advance of the argument of periapsis: this is however not treated here since the orbital eccentricity of constellation members was constrained to be close to zero. It was shown that the occultation performance is insensitive to the J_2 parameter and that therefore no station-keeping maneuver is necessary.

NAVIGATION PERFORMANCE

The performance of a navigation system is generally evaluated by the three following parameters: availability, accuracy and reliability. Only the first two are assessed in this paper.

The availability has been defined as the number of satellites in view of the incoming spacecraft at each observation step, i.e. every three minutes. In order to compute the position of the incoming spacecraft, it is necessary to have at least four satellites in view.

The accuracy of a navigation system is usually represented by two quantities: the User Equivalent Range Error (UERE) and the Geometric Dilution of Precision (GDOP). The UERE is found by mapping all of the system and user errors into a single error in one user measured range. It is mapped into the computed position by a geometrical factor called DOP. The GDOP is given in meters; the lower its value the 'stronger' the geometry of the observation model, i.e. the spacecrafts spread out in angle with respect to the user. The GDOP value is only available if at least four satellites are in view: thus it can also be used as a measure of the availability of the system. The DOP by itself can represent the accuracy of the system if it is assumed that all the measurements have the same UERE, which will be the case in this paper.

The DOP values were derived for an incoming spacecraft on an approaching hyperbola with a perigee altitude target of 150 km (representative aero-capture trajectory) from four days before arrival at the periapsis to the periapsis passage. This was repeated for orbit inclinations starting from 6 to 150 degrees. The GDOP is defined by the covariance matrix of errors $(H^T H)^{-1}$ as follows:

$$GDOP = \sqrt{tr(H^T H)^{-1}} \text{, where } H = \begin{bmatrix} s_{1x} & s_{1y} & s_{1z} & 1 \\ s_{2x} & s_{2y} & s_{sz} & 1 \\ \cdots & \cdots & \cdots & \cdots \\ s_{mx} & s_{my} & s_{mz} & 1 \end{bmatrix}$$

H is called the sensitivity matrix, $\vec{s}_i$ is a unit vector expressed in the Mars Inertial Reference Frame and pointing from the incoming spacecraft towards the i^{th} navigation satellite in the constellation and m is the number of visible satellites from the incoming spacecraft perspective. The sum of the number of times that the GDOP is not defined was used as the performance index for the Monte Carlo simulation for a 4-day spacecraft approach at an inclination of 6 and 90 degrees.

As mentioned earlier, no reliability performance evaluation (through evaluating robustness in the face of loss of one or more constellation members) was performed.

NUMERICAL METHODS

The following methods were applied for constellations of 4, 6 and 8 satellites but the results will only be presented for 4 and 8 satellites, as these are the most interesting cases.

Radio occultation events

The problem was first tackled with a Monte Carlo approach of 50,000 iterations. At each iteration the constellation geometry, i.e. the keplerian elements of each satellite, was initialized randomly, with some constraints on the semi-major axis and the eccentricity ($1.2 * r\_mars < a < 10 * r\_mars$ and $e < 0.025$). The objective function was evaluated and according to the results the geometry information was either stored in a file representing a population of promising candidate geometries, or was discarded. Once the population of candidate geometries was sufficiently large, these constellation geometries were locally optimized with a local gradient descent (using the same objective function) and the best one was selected to go further in the process. The Monte Carlo simulation and the local optimization were performed over a period of 2 sols. The selected constellations were then evaluated for navigation services (using the availability and accuracy indices described above). However the constellation geometries optimized for occultation yielded poor performance, leading to the necessary definition of a reconfiguration strategy.

Reconfiguration strategy

The apparent incompatibility between occultation and navigation geometries necessitates constellation reconfiguration for navigation events. The use of electric propulsion for the deployment, orbital acquisition and control of the constellation satellites would efficiently support the feasibility of the concept, by providing sufficient maneuver delta-V for several reconfigurations of the constellation over the mission lifetime. Known figures on electric propulsion [Wells et al, 2004, Walker et al, 2004] show that a 120 kg micro satellite arriving at low Mars orbit has between 6 and 8 kg of Xenon left assuming a 30 kg total propellant mass and Isp of 4500 sec. This leads to the main reconfiguration constraint, which is the cost of propellant – this has been set to 1.6 kg per satellite. Assuming that the mission length is 4 to 6 years, this enables the constellation to be reconfigured at least four times (supposing that the primary geometry is for radio occultation measurements and navigation events correspond to minimum energy launch windows from Earth – i.e. once every 26 months). As the different eccentricities in the primary geometry are low and a change in inclination is too expensive, these parameters were fixed along with the argument of perigee. Since the desired time of perigee and right ascension of ascending node can be easily targeted using differential J_2 drift by choosing the right time when to apply the maneuver, the strategy was focused on the semi-major axis change, for which the maneuver cost is given by the following approximate formula for a low-thrust orbit-raising spiral:

$$M_{fuel} = M_{sc}(1 - \exp(\frac{\sqrt{\mu}}{g_0 I_{sp}}(\frac{1}{\sqrt{a_2}} - \frac{1}{\sqrt{a_1}})))$$

where M_{sc} is the mass of the spacecraft before the maneuver, μ is the gravitational constant of Mars, I_{sp} is the specific impulse, g_0=9.81 m/s^2, a_1 and a_2 are respectively the initial and target semi-major axis (km).

Navigation services

In order to find the "best" constellation, a Monte Carlo simulation of 100,000 iterations was performed. Only the time of perigee and the right ascension of the ascending node were randomly initialized. The semi-major axis was constrained to an interval defined by the maximum allowed cost for the maneuver. The other keplerian elements were kept constant. The population of possible geometries was selected with respect to the objective function. Then a finer selection was performed by analyzing both the total number of times that the GDOP is not defined for the following inclinations: 6, 10, 20, …, 120 and 150 degrees (availability) and the value of the GDOP (accuracy).

RESULTS

The same scheme will be followed for the presentation of the results derived for each type of constellation geometry.

The "best" orbits for radio occultation measurements (constellation R0) will first be depicted followed by the spatial distribution of the radio occultation events where the meshing of the planet is also distinguishable. The time period for the first two figures is 10 sols. The details of the orbits parameters and the results of the process will be presented in the two tables underneath. Then the reconfigured geometry (constellation R0*) for navigation services corresponding to the above constellation will be represented in the next figure along with trajectories of incoming spacecrafts on a hyperbola leg targeting a periapsis of 150 km and at 8 different inclinations. Each trajectory on the figure stops at the periapsis. Next will be shown the GDOP values for all the trajectories represented in the previous figure. The time span is 30 minutes before arrival of the spacecraft at the periapsis. When the GDOP falls below zero, it means that there are less than four satellites in view. The results in terms of cost and length of the maneuver, availability and accuracy of the navigation constellation for each satellite are presented in the last two tables.

4-satellite constellation

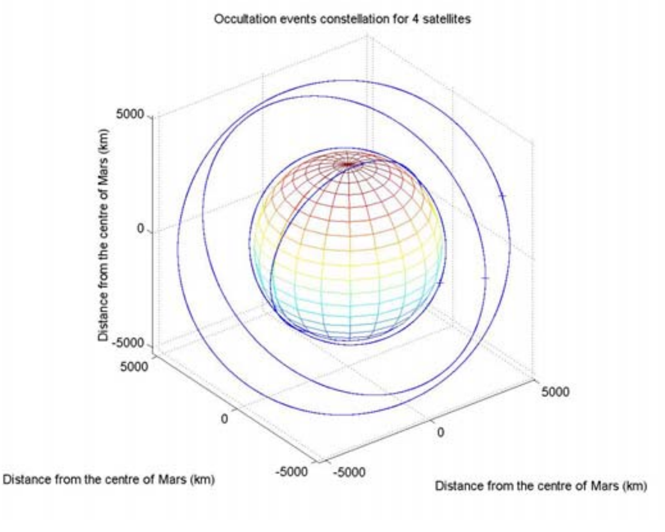

Figure 2: Constellation R0 of 4 satellites for radio occultation measurements

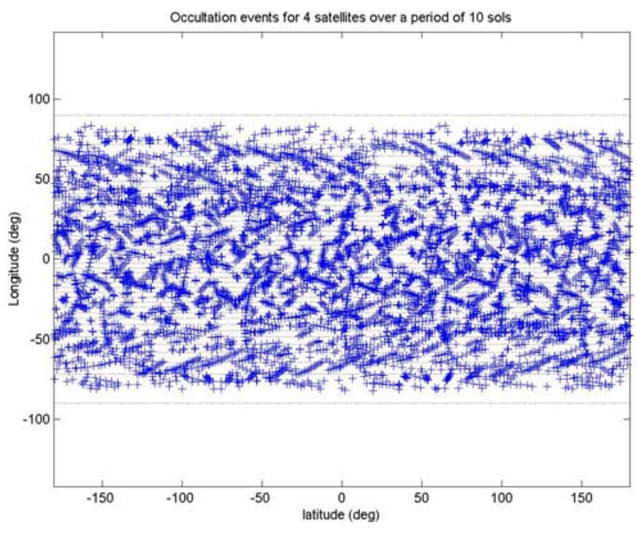

Figure 3: Spatial distribution of the radio occultation events (10 sols)

Constellation R0	a (km)	e	i (deg)	Ω (deg)	ω (deg)	τ (sec)
1st satellite	3791.49	0	129.49	154.16	207.80	7087.95
2nd satellite	3736.93	0	96.91	0.79	288.10	6935.51
3rd satellite	6372.05	0.025	54.52	314.42	21.97	15442.75
4th satellite	6103.86	0.025	94.60	114.43	177.89	14478.15

Table 1: Keplerian elements of the 4 orbits for radio-occultation measurements

Objective function	# occultation events	Mean value per slice	Standard deviation
3.26	1535	38.37	7.78

Table 2: Results for the radio occultation measurements for 4 satellites and 2 sols

The number of occultation events is very reasonable and all the latitudes are covered.

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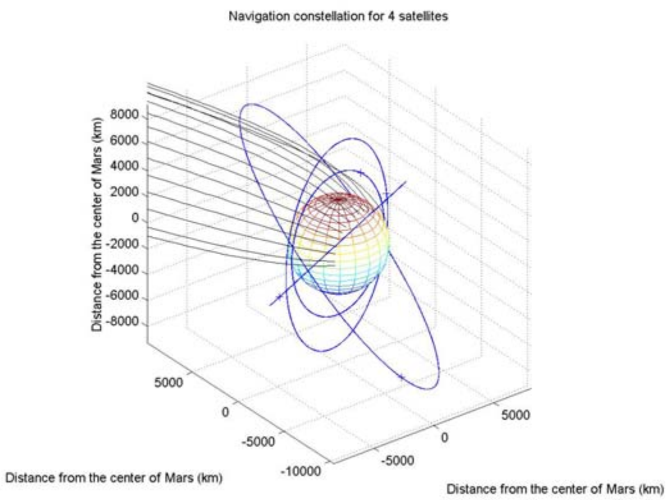

Figure 4: Constellation R0* of 4 satellites for navigation services

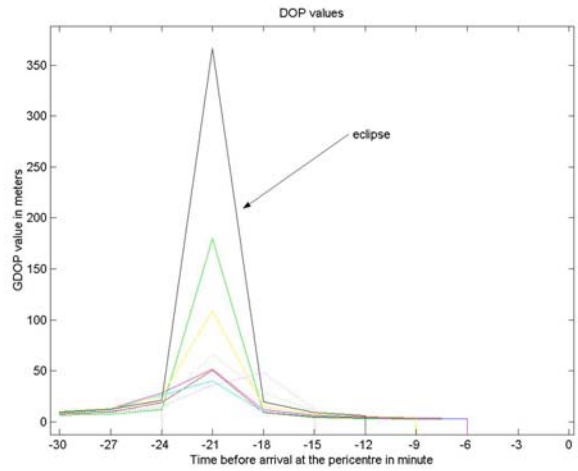

Figure 5: GDOP values from 30 minutes before the periapsis to the periapsis arrival

Constellation R0*	Semi-major axis change (km)	Cost (kg)	Time of maneuver (days)
1st satellite	+2349.42	1.55713	49.72
2nd satellite	+1270.01	0.99919	31.91
3rd satellite	+4922.82	1.39661	44.60
4th satellite	+1103.56	0.45939	14.67

Table 3: Reconfiguration maneuver details for 4 satellites

	Time of non availability summed through all the inclinations for 4 days	DOP value through all the inclinations in the last 30 minutes
R0	Never defined	Never defined
R0*	291 minutes	<350 if eclipse taken into account <14 if no eclipse

Table 4: Comparison of the Navigation services performances for R0 and R0* for 4 satellites

The reconfiguration maneuver improves considerably the navigation services, which could not be supported by the initial configuration, R0. The maximum reconfiguration maneuver cost per satellite is 1.557 kg of Xenon, which is within the constraints and takes less than 50 days to be performed. The GDOP is still too high and the availability too weak in order that this constellation could be considered for only navigation services. It could provide however a very good navigation aid for the incoming spacecraft.

8-satellite constellation

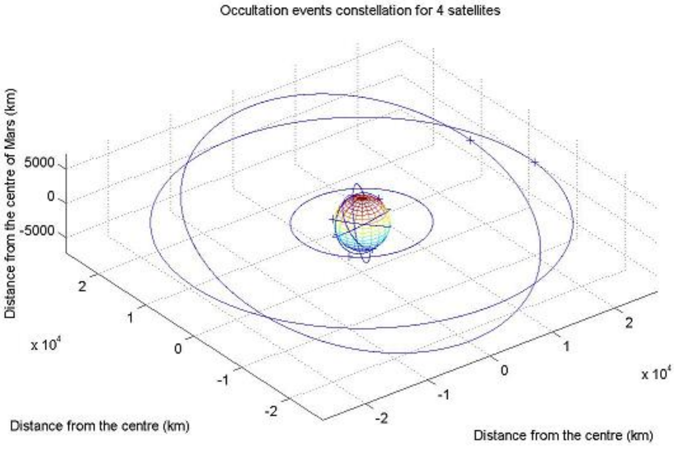

Figure 6: Constellation R0 of 8 satellites for radio occultation measurements

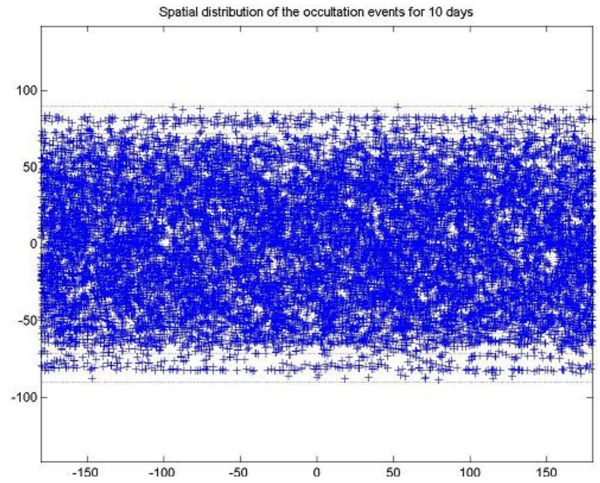

Figure 7: Spatial distribution of the radio occultation events (10 sols)

Constellation R0	a (km)	e	i (deg)	Ω (deg)	ω (deg)	τ (sec)
1st satellite	5094.68	0.024	101.55	226.63	165.70	11040.31
2nd satellite	3755.24	0	28.19	127.64	0.35	6986.55
3rd satellite	9078.74	0	178.44	4.42	142.29	26263.00
4th satellite	23430.76	0.009	17.96	5.18	11.25	108889.34
5th satellite	3745.82	0.025	54.31	34.52	244.86	6960.26
6th satellite	26884.20	0	180.00	0.00	2.51	133829.40
7th satellite	4097.32	0.025	144.29	11.67	1.59	7962.62
8th satellite	3922.21	0.023	73.34	357.06	36.56	7457.66

Table 5: Keplerian elements of the 4 orbits for radio-occultation measurements

Objective function	# occultation events	Mean value per slice	Standard deviation
14.76	4772	119.3	17.98

Table 6: Results for the radio occultation measurements for 8 satellites and 2 sols

The number of occultation events for 2 sols is very high and the spatial distribution is well balanced except from the poles due to the fact that there is no polar orbit.

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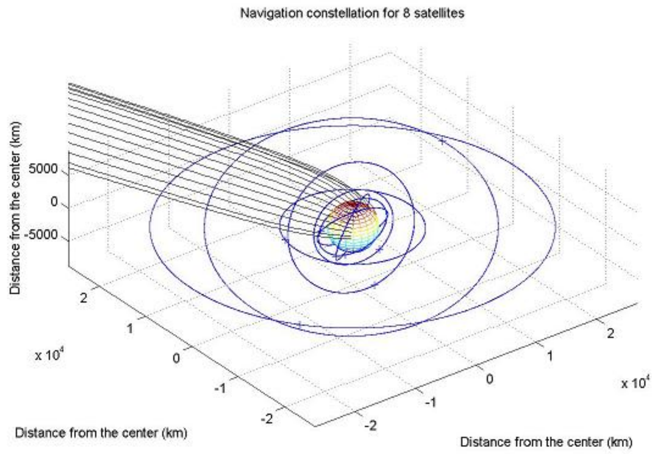

Figure 8: Constellation R0* of 8 satellites for navigation services

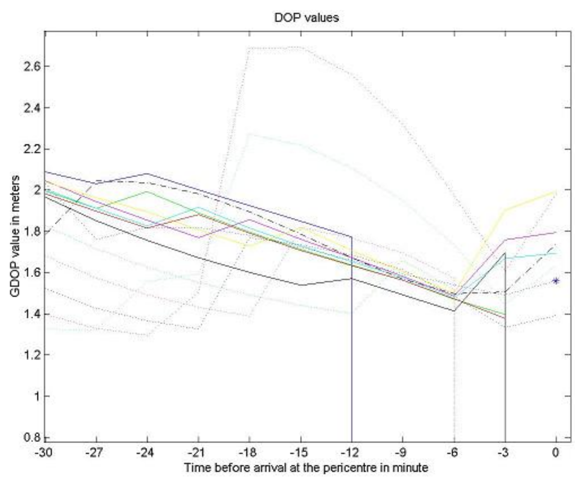

Figure 9: GDOP values from 30 minutes before the periapsis to the periapsis arrival

	Semi-major axis change (km)	Cost (kg)	Time of maneuver (days)
1st satellite	+3867.43	1.54273	49.26
2nd satellite	+1163.45	0.92499	29.54
3rd satellite	+509.40	0.127424	4.07
4th satellite	+4121.42	0.29984	9.57
5th satellite	+1248.12	0.98231	31.37
6th satellite	-244.70	0.01261	0.40
7th satellite	+1528.13	1.0275689	32.81
8th satellite	+879.90	0.6909248	22.06

Table 7: Reconfiguration maneuver details for 8 satellites

	Time of non availability summed through all the inclinations for 4 days	DOP value through all the inclinations in the last 30 minutes
R0	8589 minutes=5.96 combined hours	<5 (but only 2 inclinations are defined)
R0*	33 minutes	<2.67

Table 8: Navigation services performances for 8 satellites

The reconfiguration maneuver enables the formation of a constellation R0* that has very good navigation figures of merit. The maximum reconfiguration maneuver cost per satellite is 1.543 kg of Xenon, which is within the constraints and takes less than 50 days to be performed. The local DOP is close to the Galileo constellation one and at some inclinations it is defined at all times.

CONCLUSION

In this article it has been found that for a constellation of 8 satellites, a single multi-micro spacecraft mission can achieve very effective incoming navigation services and highly valuable atmospheric science return. On the other hand, even if the results were quite promising with 4 satellites, such a constellation would fail to perform valuable navigation services by itself but would still be a good aid for orbit insertion. For both constellations and over a 6-7 year period, 4 maneuvers can be effected in order to alternate between 3 occultation configurations and 2 navigation configurations. At a system level and according to the results shown in Tables 3 and 7, it could be envisaged to grant one spacecraft with more fuel than the others for which the maneuver cost is lower in order to prolong the lifetime of the whole mission. Further work to this paper will be to analyze the sensitivity of the navigation performances to the J_2 oblateness and also study the standard deviation accuracy of the orbit insertion defined by the B-plane semi-major axis.

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