OPTIMAL FORMATION FLYING RECONFIGURATION AND STATION KEEPING MANEUVERS USING LOW THRUST PROPULSION

Roberto Armellin[1], Mauro Massari [1] and Amalia Ercoli Finzi[1]

[1]*Politecnico di Milano, Via La Masa 34-21056-Milano, Italy*
E-mail: armellin@aero.polimi.it, massari@aero.polimi.it, finzi@aero.polimi.it

ABSTRACT

In this work both reconfiguration and station keeping maneuvers of spacecraft flying in formation with low-thrust propulsion has been faced. The aim is the development of a real-time control strategy that reduces the propellant fraction required for these maneuvers. First a method to generate the optimal trajectories has been developed using a direct approach and a simplified model of the relative dynamics with respect to elliptic reference orbits, then the same method has been applied to control the maneuvers. Finally simulations considering a complete model have been carried out for different test cases.

1. INTRODUCTION

Formation flying is an enabling technology for many future space science missions. The use of a distributed array of simple but high coordinated satellites replaces the big systems used today. Formation flying technology increases the performance of interferometric instruments through the syntetization of large passive aperture and enables the design of missions otherwise unfeasible. Indeed distributed sensor concept overcomes the limitations given by having a single large structure in space. These limitations concern with the technological problem of stabilizing such a structure, a complex deployment mechanism as well as an increased lunch mass or a restriction of appropriate launchers.

Reconfiguration maneuvers can be required to realize multi-objective missions, to make up for a satellite failure and to introduce new satellites in the formation. Station keeping maneuvers are required to maintain the formation geometry under the influence of perturbative forces. Both reconfiguration and station keeping maneuvers have been faced as a low thrust optimal trajectory design problem. In the reconfiguration maneuver each satellite has to reach a new given position inside the formation to generate a new formation geometry; in the station keeping maneuver each satellite has to move from its perturbed position to its ideal one in order to maintain the prescribed geometry.

The problem has been solved developing a method for the generation of the control sequence based on a discretization of the differential constraints and a parametrization of the controls: this leads to a parametric optimization problem. The differential constraints are represented by a set of linear time-varying differential equations that describe the relative motion of each satellite with respect to a general elliptic reference orbit. To reduce the optimization parameters to the initial states and to the control sequence only, the dynamics has been discretized using the state transition matrix and the discrete convolution. To do that the controls have been considered constant over a single discretization step. In this way the differential equations are transcribed into a set of equality constraints, while the limits of thrust level are introduced as bounds on the optimization parameters. In order to avoid collision during the reconfiguration maneuvers a repulsive term, function of the inverse of the distances between the spacecrafts, has been added to the objective function. Once the optimal sequence of controls has been generated using a simplified model, it is necessary to control that the errors on the real system remain low. To make this possible the calculated control sequence is applied to a complete non linear model which consider the disturbances coming from the non spherical Earth, from atmospheric drag and from sensors noise. Instead of controlling the trajectory with a classical feedback controller, the control strategy developed consists in checking that the real trajectory stays in the vicinity of the optimal one computed. When the error becomes greater than a predefined threshold a new optimization is carried out computing a new optimal control sequence. In this way the real reconfiguration and station keeping maneuvers are made up by a series of optimal trajectories.

2. DYNAMICS AND DISCRETIZATION

This paper addresses to formation flying with small baseline (<1km). In these conditions linearized dynamics describe the satellites motion inside the formation with good accuracy. Furthermore linearized equations are useful to reduce computational burden for the trajectory design. The reference frame can be either centered on a formation satellite or on a virtual satellite which can represent the formation center of mass. The *X* axis has radial direction, *Z* axis is the unity vector of the momentum $\boldsymbol{h}$ (where $\boldsymbol{h} = \boldsymbol{r} \times \boldsymbol{v}$) and it defines the cross-track direction, *Y* axis complete the right handed orthogonal triad and defines the in-track direction. In the general case of elliptic orbit reference frame, the linearized equations of motion are given by the set of differential equations [1]

$$\frac{d}{d\theta}\begin{bmatrix} x \\ x' \\ y \\ y' \\ z \\ z' \end{bmatrix} = \begin{bmatrix} 0 & 1 & 0 & 0 & 0 & 0 \\ \frac{3+e\cos\theta}{1+e\cos\theta} & \frac{2e\sin\theta}{1+e\cos\theta} & \frac{-2e\sin\theta}{1+e\cos\theta} & 2 & 0 & 0 \\ 0 & 0 & 0 & 1 & 0 & 0 \\ \frac{2e\sin\theta}{1+e\cos\theta} & -2 & \frac{e\cos\theta}{1+e\cos\theta} & \frac{2e\sin\theta}{1+e\cos\theta} & 0 & 0 \\ 0 & 0 & 0 & 0 & 0 & 1 \\ 0 & 0 & 0 & 0 & -\frac{1}{1+e\cos\theta} & \frac{2e\sin\theta}{1+e\cos\theta} \end{bmatrix} \begin{bmatrix} x \\ x' \\ y \\ y' \\ z \\ z' \end{bmatrix} +$$

$$+\frac{\left(1-e^2\right)^3}{\left(1+e\cos\theta\right)^4 n^2}\left(\begin{bmatrix} 0 & 0 & 0 \\ 1 & 0 & 0 \\ 0 & 0 & 0 \\ 0 & 1 & 0 \\ 0 & 0 & 0 \\ 0 & 0 & 1 \end{bmatrix}\begin{bmatrix} u_x \\ u_y \\ u_z \end{bmatrix} + \begin{bmatrix} 0 & 0 & 0 \\ 1 & 0 & 0 \\ 0 & 0 & 0 \\ 0 & 1 & 0 \\ 0 & 0 & 0 \\ 0 & 0 & 1 \end{bmatrix}\begin{bmatrix} w_x \\ w_y \\ w_z \end{bmatrix}\right) \quad (1)$$

Eqn. 1 can be compactly represented as a general linear time varying (LTV) state space model

$$\boldsymbol{x}' = \boldsymbol{A}(\theta)\boldsymbol{x}(\theta) + \boldsymbol{B}(\theta)\boldsymbol{u}(\theta) + \boldsymbol{D}(\theta)\boldsymbol{w}(\theta) \quad (2)$$

where $\boldsymbol{x}$ is the state vector, $\boldsymbol{u}$ and $\boldsymbol{w}$ are the control and the disturbance vectors respectively. There are two main advantages given by the use of linearized equations of dynamics in the general case of eccentric reference orbit. The first is that also for reference orbit of small eccentricity the effects of modeling errors using Hill's equations are comparable to that given by the main perturbative forces (J_2 and atmospheric drag disturbances). The second one is that Eqn. 1 allows to consider formations in high eccentricity reference orbit like Molnya. The analytical solution of the unforced part of the differential equations has been carried out by Marec [2]. The analytical solution is an important tool to define the initial conditions of the state vector $\boldsymbol{x}$ for a formation initialization [1]. To reduce the optimization problem to a parametric optimization one Eqn. 1 has to be discretized; this has been accomplished for the unforced part of the motion by means of the state transition matrix $\boldsymbol{\Phi}$. The discrete counterpart of Eqn. 1 is

$$\boldsymbol{x}_{k+1} = \boldsymbol{\Phi}_{k+1,k}\boldsymbol{x}_k + \boldsymbol{\Gamma}_k\boldsymbol{u}_k + \boldsymbol{\Lambda}_k\boldsymbol{w}_k \quad (3)$$

where

$$\boldsymbol{\Gamma}_k = \boldsymbol{\Gamma}(t_k) = \int_{t_k}^{t_{k+1}} \boldsymbol{\Phi}(t_{k+1},\tau)\boldsymbol{B}(\tau)\boldsymbol{u}(\tau)\,d\tau \quad (4)$$

$$\boldsymbol{\Lambda}_k = \boldsymbol{\Lambda}(t_k) = \int_{t_k}^{t_{k+1}} \boldsymbol{\Phi}(t_{k+1},\tau)\boldsymbol{D}(\tau)\boldsymbol{w}(\tau)\,d\tau \quad (5)$$

These integrals have been computed using the trapezoidal method with the assumption of constant control and perturbative values within each time step. To have only control sequence as variable of the optimization problem, discrete convolution has been used. The state vector at any instant of time can be expressed as a function of the initial state vector and the control sequence. Compactly this can be written as

$$\boldsymbol{x}_k = \boldsymbol{A}_k\boldsymbol{u}_k + \tilde{\boldsymbol{b}}_k \quad (6)$$

where

$$\boldsymbol{A}_k = \left[\boldsymbol{\Phi}^{(k-1,k)}\boldsymbol{\Gamma}_0 \quad \boldsymbol{\Phi}^{(k-2,k)}\boldsymbol{\Gamma}_1 \ldots\ldots \quad \boldsymbol{\Phi}^{(0,k)}\boldsymbol{\Gamma}_{k-1}\right] \quad (7)$$

$$\boldsymbol{u}_k = \left[\boldsymbol{u}_0^T \quad \boldsymbol{u}_1^T \ldots\ldots\ldots \quad \boldsymbol{u}_{k-2}^T \quad \boldsymbol{u}_{k-1}^T\right]^T \quad (8)$$

$$\tilde{\boldsymbol{b}}_k = \boldsymbol{\Phi}^{(k,k)}\boldsymbol{x}_0 \quad (9)$$

$$\boldsymbol{\Phi}^{(j,k)} = \begin{cases} \boldsymbol{\Phi}_{(k)}\ldots\ldots\boldsymbol{\Phi}_{(k-1)}\boldsymbol{\Phi}_{(k-j+1)}, & 2 \le j \le k \\ \boldsymbol{\Phi}_{(k)}, & j = 1 \\ \boldsymbol{I}, & j = 0 \end{cases} \quad (10)$$

In the last definition $\boldsymbol{\Phi}_{(k)}$ stands for $\boldsymbol{\Phi}_{k,k-1}$. When also perturbative forces are taken into account, Eqn. 6 has to be modified into

$$\boldsymbol{x}_k = \boldsymbol{A}_k\boldsymbol{u}_k + \tilde{\boldsymbol{b}}_k + \boldsymbol{A}_k\boldsymbol{w}_k \quad (11)$$

The term $\boldsymbol{w}_k$ can't be neglected when concerning with a reconfiguration maneuver about a unperturbed reference orbit. When analyzing a reconfiguration maneuver about a perturbed reference orbit, vector $\boldsymbol{w}_k$ represents only differential perturbations acting on a satellite. These have been neglected in first instance. This approximation is good when dealing with close formations made up satellites with similar geometrical and inertial properties (in order to reduce both earth oblateness and differential atmospheric drag perturbations).

3. OPTIMIZATION FORMULATION

An optimal control problem is made up of tree main parts: the objective function, inequality constraints and equality constraints [4]. In a reconfiguration maneuver each satellite has to move from the initial state $\boldsymbol{x}_0$ to the final state $\boldsymbol{x}_{fin}$. The maneuver length (duration) is fixed and equal to kT_s where k is the number of intervals in which the maneuver has been discretized and T_s is the sample period. By defining $\boldsymbol{b}$ vector as

$$\boldsymbol{b} = \boldsymbol{x}_{fin} - \tilde{\boldsymbol{b}}_k - \boldsymbol{A}_k\boldsymbol{w}_k \quad (12)$$

the equality constraints due to dynamics (Eqn. 11) can be written as

$$\boldsymbol{A}\boldsymbol{u} = \boldsymbol{b} \quad (13)$$

where k subscript has been dropped. Only inequality constraints given by thrust saturation have been considered. These constraints are written as

$$l_b \leq u \leq u_b \tag{14}$$

where

$$u_b = \left[\left(u_{\max,x}, u_{\max,y}, u_{\max,z} \right)^T \cdots \left(u_{\max,x}, u_{\max,y}, u_{\max,z} \right)^T \right] \tag{15}$$

$$l_b = -u_b \tag{16}$$

Finally objective function has to be introduced. The aim of the algorithm is to find optimal fuel reconfiguration maneuvers. For reconfiguration maneuvers without collision risk the objective function is the normalized sum of squares of the controls

$$f = \frac{1}{2} \frac{u^T u}{u_b^T u_b^T} \tag{17}$$

The problem statement can be summarized by Table 1.

Table 1. Optimization problem statement: no collision risk

Optimization parameters	Objective function	Equality constraint	Inequality constraint
$\boldsymbol{u}$	$f = \frac{1}{2} \frac{\boldsymbol{u}^T \boldsymbol{u}}{\boldsymbol{u}_b^T \boldsymbol{u}_b}$	$\boldsymbol{Au} = \boldsymbol{b}$	$\boldsymbol{l}_b \leq \boldsymbol{u} \leq \boldsymbol{u}_b$

Note that when the reconfiguration maneuver involves more than one satellite $\boldsymbol{u}$, $\boldsymbol{b}$ vectors and $\boldsymbol{A}$ matrix are assembled by the vectors and matrixes of each satellite. When dealing with this kind of maneuvers the number of knots k must be taken small enough to guarantee that the number of optimization variables fits with the requirement of real time application of the algorithm. When collision risk is detected the objective function is modified adding a penalty term

$$f = \frac{1}{2} \frac{\boldsymbol{u}^T \boldsymbol{u}}{\boldsymbol{u}_b^T \boldsymbol{u}_b} + \alpha \sum_{l=1}^{k} \frac{1}{d_{l,ji}} \quad \boldsymbol{i}, j = 1, \ldots N \tag{18}$$

where $d_{l,ji}$ represents the distance between each satellite from any other satellite of the formation at any time step of the maneuver. If two satellites get close the objective function explodes and the optimization process avoids that solution. Note that in case of collision risk the optimization problem is more complicated due to the presence of terms (satellites distances) that are not explicit functions of the control variables. This implies that computational time increases when collision risk is detected [5]. Instead of adding a penalty term to the objective function, collision avoidance could have been obtained through the imposition of inequality constraint on satellite distances at any time step. These techniques have been discharged because of computational burden. In case of collision risk the optimization problem is summarized in Table 2. The so defined problem has been solved using Matlab *fmincon* routine which allows solving both linear and sequential quadratic programming problems [6].
The formation station keeping has been faced as a particular reconfiguration maneuver which involves one spacecraft at time. At each time step and for each spacecraft an error box is defined around its nominal position; this error box defined all the positions allowed by the formation specifics. When a spacecraft reaches the edge of the error box an optimal maneuver brings it to its corresponding nominal position. Once that the time duration of the maneuver is chosen the final state $\boldsymbol{x}_{fin}$ is determined and the optimal trajectory can be computed.

Table 2. Optimization problem statement with collision risk

Optimization parameters	Objective function	Equality constraint	Inequality constraint
$\boldsymbol{u}$	$f = \frac{1}{2} \frac{\boldsymbol{u}^T \boldsymbol{u}}{\boldsymbol{u}_b^T \boldsymbol{u}_b} + \alpha \sum_{l=1}^{k} \frac{1}{d_{l,ji}}$ $i, j = 1, \ldots N$	$\boldsymbol{Au} = \boldsymbol{b}$	$\boldsymbol{l}_b \leq \boldsymbol{u} \leq \boldsymbol{u}_b$

4. TRAJECTORY CONTROL

To enable real-time application of the optimization techniques described above tree main simplifications has been done. Even if close formation flying has been taken into account the linearized dynamics introduce some modeling errors. When dealing with maneuvers around a perturbed reference orbit, differential perturbations acting on satellites have been neglected; for maneuvers around an unperturbed reference orbit perturbative accelerations has been considered constant within each time step. Applying the input vector computed with these simplifications to a more realistic model the real trajectory could be very different from the ideal one. To respect the high precision requirements imposed on satellites relative positions, a control algorithm has to be performed. Instead of tracking the ideal trajectory a dynamic refresh of optimal trajectories has been done. The real trajectory of each satellite is computed through the integration of the nonlinear dynamics

$$\ddot{\boldsymbol{r}} = -\frac{\mu}{r^3} \boldsymbol{r} + \boldsymbol{d} + \boldsymbol{u} \tag{19}$$

Vector $\boldsymbol{d}$ includes both Earth oblateness and atmospheric drag perturbative forces and $\boldsymbol{u}$ is the control vector computed by solving the optimization problem. Also sensor noises on relative positions and velocities have been taken into account. Errors on position and velocity estimation have been introduced referring to DCGPS sensors. Errors of $\pm$ 2 cm on relative position and $\pm$.3 mm/s on relative velocities have been demonstrated for this kind of sensor [7],[8]. The dynamic refresh of a trajectory starts whenever the difference between the ideal trajectory and the real one becomes bigger than a predefined threshold. Two kinds of trajectory refreshment have been considered. If collision risk is absent each satellite refreshes its optimal trajectory through the minimization of the objective function

$$f = \frac{1}{2} \frac{\boldsymbol{u}_r^T \boldsymbol{u}_r}{\boldsymbol{u}_b^T \boldsymbol{u}_b} \tag{20}$$

where $\boldsymbol{u}_r$ is the control sequence from the instant of time in which the new optimization starts to the end of the maneuver. Note that in this case the real trajectory is made up of a sequence of optimal trajectories. The time needed to compute the new optimal trajectory is always smaller than that required for the previous one due to receding horizon ($dim(\boldsymbol{u}_r)<dim(\boldsymbol{u})$) Using dynamic refresh instead of tracking the first computed trajectory allows intensive fuel savings. Indeed tracking the first optimal trajectory should require a continuous additive control with the same order of magnitude of the perturbative forces that act on each satellite. In reconfiguration maneuvers with collision risk objective function given by Eqn. 20 would not guarantee a safe maneuver. Indeed the new trajectories computed trough the refresh technique could cause collision between satellites. In this case the objective function used is

$$f = \sum_{i=a}^{k} \left(\boldsymbol{x}_{i,real} - \boldsymbol{x}_{i,ideal} \right)^2 \tag{21}$$

In this way the distance between the real path and the optimal path has been minimized.

5. RESULTS

Different test cases have been analyzed to demonstrate the good performance of the algorithm proposed for both the reconfiguration and station-keeping maneuvers. For each analyzed maneuver the control strategy shown in section 5 has been applied. An error of 10 cm on the final position and of .5 mm/s on the final velocity has been considered.

5.1 Collision avoidance

The first test demonstrates the effectiveness of the algorithm when collision risk is detected. The formation reconfiguration involves four satellites that exchange their relative positions on the vertexes of a square of 20 meters of side. The reference orbit is characterized by the following parameters

$$a = 7178000 \;[\mathrm{m}] \quad e = 0 \quad i = 90° \quad \Omega = 0 \quad \omega = 0\,.$$

Four identical satellites has been considered with these properties

$$m = 50 \;[\mathrm{Kg}],\; u_{\max} = 50 \;[\mathrm{mN}],\; C_b = 100$$

where C_b is the ballistic coefficient. The value of the control can be obtained with a cold gas propulsion system or with an electric propulsion system like a Hall effect engine [10]. If the reconfiguration maneuver were computed without adding the penalty term to the objective function satellite$_3$ would collide with satellite$_1$ and satellite$_2$ with satellite$_4$, as shown in [5]. Using the penalty term the maneuver is accomplished in a safe manner as it is shown in Figs.1-2.

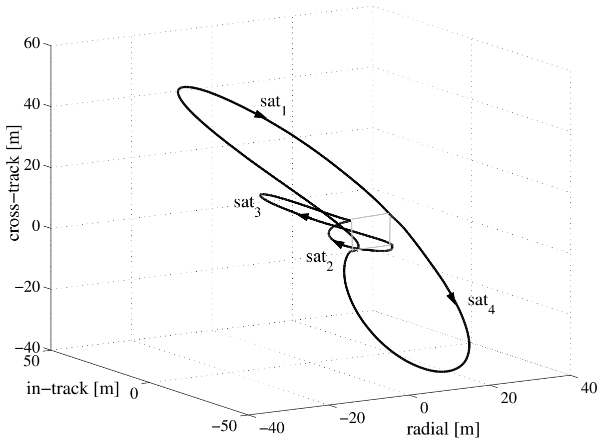

Fig. 1. Safe reconfiguration maneuver of four satellites

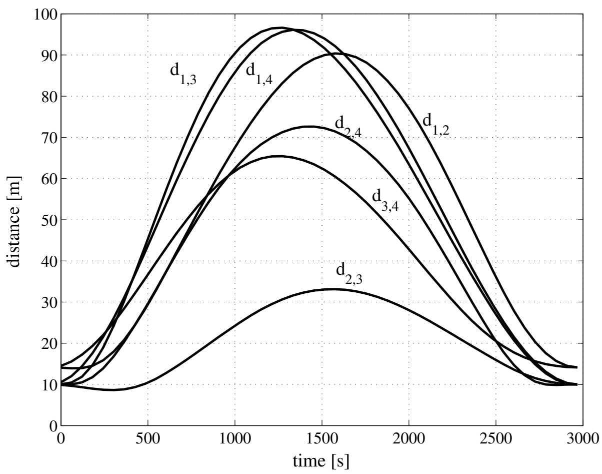

Fig. 2. Satellites relative distances during the maneuver

5.2 Formation reconfigurations

A reconfiguration maneuver of a three satellites formation has been considered. Through the maneuver the formation increases the radius of the circumference projected in the in-track/cross-track plane from 200 m to 600 m. The reference orbit parameters are

$$a = 7178000\ [\mathrm{m}] \quad e = 0 \quad i = 90° \quad \Omega = 0 \quad \omega = 0$$

Three identical satellites of the following properties have been considered

$$m = 50\ [\mathrm{kg}],\ u_{\max} = 5\ [\mathrm{mN}],\ C_b = 100$$

The control action required can be realized with multi-feep propulsion system [11]. The reconfiguration trajectories and the Δv required are shown in Fig.3 and Table 3 respectively.

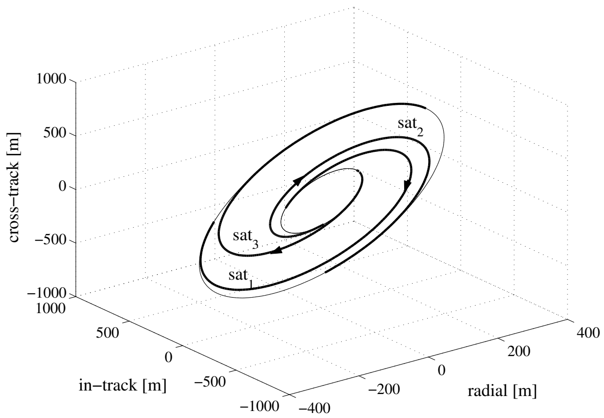

Fig. 3. Three satellites reconfiguration maneuver, circular reference orbit

Table 3 Δv required for three satellites reconfiguration maneuver , circular reference orbit.

	Δv [mm/s]
Satellite 1	509.7
Satellite 2	535.2
Satellite 3	461.7

To demonstrate the algorithm effectiveness also in the case of high eccentricity orbit a formation reconfiguration considering a Molnya reference orbit has been done. Molnya parameters are:

$$a = 26000000\ [\mathrm{m}] \quad e = .74 \quad i = 63° \quad \Omega = 0 \quad \omega = 0$$

A three spacecrafts small relative orbit is reconfigured into a larger one. The resulting trajectories are shown in Fig. 4. Table 4 summarizes the Δv required to execute the maneuver. The computational time necessary for the two reconfiguration maneuvers shown in this subsection is respectively of 4.94 and 4.88 s using a Pentium 4 processor, 2.0 GHz.

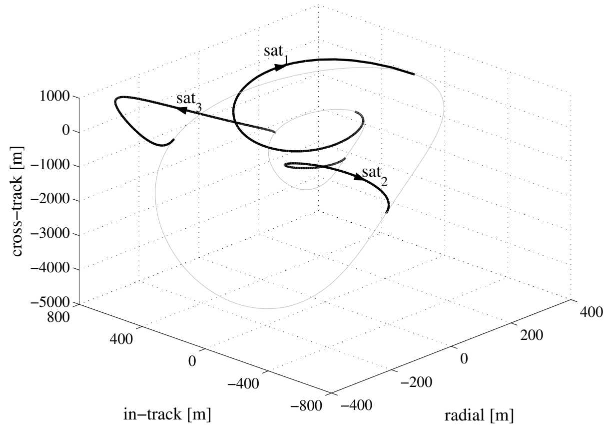

Fig. 4. Three satellites reconfiguration maneuver, Molnya reference orbit

Table 4 Δv required for three satellites reconfiguration maneuver , Molnya reference orbit

	Δv [mm/s]
Satellite 1	824.1
Satellite 2	1293.3
Satellite 3	1635.7

5.3 Formation station keeping

The station keeping algorithm is now performed for both the formations considered in the previous subsection. An error box of ±5 m in the radial direction and ±10 both in the in-track and cross-track direction has been considered. These dimensions are compatible for a typical in-sar formation flying mission as shown in [9]. For the circular reference orbit the station-keeping maneuver acts every three orbits. The control is active for the 4% of the orbital time and requires an average Δv consumption of 4.8 mm/s for each orbit. For the Molnya reference orbit a station keeping maneuver is required at each orbit due to the low perigee altitude (380 km). A control active for the 51% of the orbital time has been considered with an average Δv consumption of 5.1 mm/s for each orbit. Figs 5-6 shows the station keeping maneuvers for both cases where '+' indicates uncontrolled motion and 'o' the controlled part. As it can be seen from Fig. 6 the algorithm allows small overshooting that has to be taken into account in the definition of the error box dimensions.

6. CONCLUSION

Through the test cases shown in the previous paragraph the effectiveness of the trajectory design and control algorithm has been demonstrated. The method is quite general and can be easily modified to introduce particular constraints on the maneuvers. The trajectory control through dynamic refresh allows the respect of formation precision requirements and makes the algorithm effective against any kind of disturbance (not only Earth oblateness, atmospheric drag and sensor noise perturbative effects). The in real-time implementation of the control strategy has been demonstrated. Indeed for the test cases that have been shown, computational times have been always smaller than discretization step. When dealing with collision avoidance maneuvers, it could happen that the first trajectory computation requires a computational time grater than the discretization step. In these cases a forward propagation of the satellites positions is required to enable in real-time application. Due to linearized dynamics the method is suitable only for close formations with baseline of the order of 1 km. Furthermore the need of real time application implies maneuver duration of the order of the reference orbit period. Indeed a longer maneuver would require a greater number of discretization knots and consequently a longer computational time. This problem could be avoided considering maneuvers made up of a sequence of thrust and coast arcs.

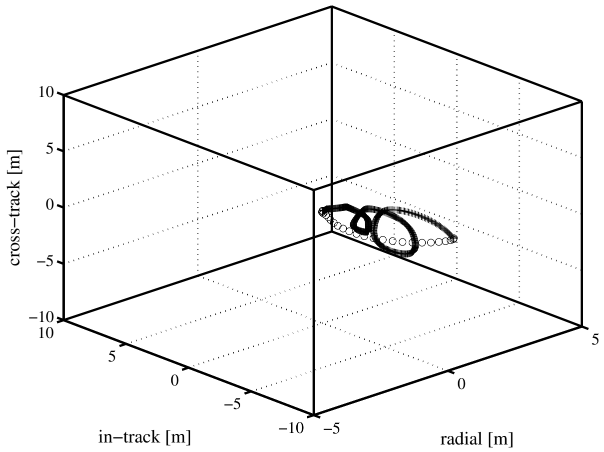

Fig. 5. Station keeping maneuver, circular reference orbit

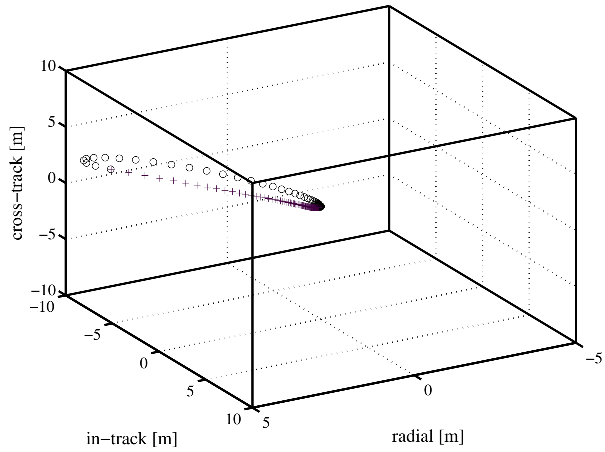

Fig. 6. Station keeping maneuver, Molnya reference orbit

7. REFERENCES

1. Inalhan G., Tilerson M and How J.P., Relative Dynamics and Control of Spacecraft Formations in Eccentric Orbits, *Journal of Guidance, Control and Dynamics*, Vol. 25, No. 1, January-February 2002

2. Marec J., *Optimal Space Trajectories*, Elsevier Scientific, NY 1979

3. Borne G., *Fundamentals of Orbit Determination*, ASEN 5070, University of Colorado, Boulder, Fall semester 2002

4. Bryson A.E. and Ho Y.C., *Applied Optimal Control*, Hemisphere/Wiley, 1975.

5. Massari M., Armellin R. and Ercoli-Finzi A., *Optimal Trajectory Generation and Control for Reconfiguration Maneuvers of Formation Flying using Low-thrust Propulsion*, 14th AAS/AIAA Space Flight Mechanics Meeting, 8-12 February 2004, Maui, Hawaii

6. Coleman T. and Branch M.A., Grace A., *Optimization Toolbox for Use with MATLAB*, User guide Version 2, The Math Works Inc.

7. Corazzini T., Robertson A. and Adams J.C., Hassabi A., How J.P., *GPS Sensing for Spacecraft Formation Flying,* proceedings of the institute of Navigation GPS-97 Conf, Kansas City MO, September 1997

8. Leung S. and Montenbruck O., *High Precision Real Time Navigation for Spacecraft Formation Flying*, ION GPS conference, 9-12 Sept.2003, Portland , Oregon (2003)

9. Martin M., Kuplar P., Kilberg S. and Winter S., *Techsat 21 and Revolutionizing Space Missions using Microsatellites*, AIAA, 2001

10.http://www.aiaa.org/images/about/01_TC_Hghlights/aiaa-ep.pdf