Electrostatic Force for Swarm Navigation and Reconfiguration

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Abstract

In this work the concept of a swarm of satellites controlled by a hybrid thrusting electrostatic actuation system is assessed. On one side the propulsion system is investigated. First from the model of the interaction between the space plasma and a charged spacecraft a set of requirements is derived. This allows to define an actuation system for charge control. Then the applicability of the electrostatic actuation for formation keeping and reconfiguration of swarms of satellites is assessed. In particular this work aims at demonstrating that the electrostatic actuation can be exploited in a decentralized control scheme to trigger high fuel savings in reconfiguration maneuvers of swarms of satellites. To this end a novel charging strategy has been developed. The resulting system has been tested under different possible simulations and it has shown good performances in terms of reduction of the fuel expenditure for the whole swarm.

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1 Introduction

Recently it was identified the possibility to perform active electric charge control on spacecraft in order to achieve a certain relative dynamic exploiting the inter-satellite electrostatic forces.[3] On the basis of this result in [6] a charging control law able to stabilize the motion of a formation of satellites only by means of the electrostatic force was developed. Only in a recent paper [4] a hybrid approach has been considered where a chief satellite can control its position by thrusters and in the meantime exploit the charge control to deploy a group of deputy satellites. However, being the electrostatic forces internal, it is not possible to use them to control the position of the formation center. For this and some other reasons, it is quite likely that if electrostatic forces will ever be used to control the relative motion or positioning of orbiting satellites, these will be coupled with conventional thrusting propulsion systems, able to control the absolute position of the satellites. In the present work we discuss the possibility to use such an hybrid actuation system in connection with decentralized path planning algorithms to improve the fuel efficiency of acquisition and reconfiguration maneuvers of swarms of satellites. Following this idea the problem is not anymore to find a formation that can be stabilized or a maneuver that can be steered only relying upon the inter-spacecraft electrostatic interaction. Rather, the proposed approach tries, for a general maneuver, to find the instantaneous spacecraft charge levels that minimize the residual force to be provided by the thrusting actuation system. To this end our investigation will be divided in two parts. In the first part the spacecraft interaction with the surrounding plasma is investigated to define requirements and evaluate the charge actuation system. Then we introduce a novel charge control feedback to derive, from the output of the path planning law, a course of desired charges for the spacecraft belonging to the swarm. Finally, the performance of the charge control feedback is illustrated by means of simulations.

2 Hybrid propulsion system definition and evaluation

2.1 Model of space charging

In several works [3, 4, 5, 6] the geostationary (GEO) environment was identified as a suitable environment for the application of inter-spacecraft electrostatic forces for formation control. The space plasma in GEO consists mainly of electrons and protons. In this region of space several natural phenomena exist, which tend to change the electrical charge of the spacecraft. These are mainly represented by the solar radiation and complex interactions between the Earths magnetosphere, the local space plasma and the solar wind. This causes the plasma environment to change over time and space and leads to varying fluxes of charged particles to and from the satellite. The incoming fluxes are the primary electron I_e and the primary proton I_i current. Furthermore there is the secondary electron emission from the spacecraft. This is mainly caused by the impact of primary electrons, while a smaller portion is caused by the impact of primary ions and backscattered secondary electrons. Another important flux is the emission of photoelectrons I_{photo} during sunlit periods.

In order to define the requirements of the electrostatic actuation system, a model of the natural space charging process has been derived and is widely discussed in [5]. The model will be explained briefly here. The spacecraft is assumed to have a spherical shape, to be perfectly conductive and to have an uniform charge distribution. Neglecting the secondary and backscattered electrons, the sum of all natural currents I_{tot} amounts to:

$$I_{tot} = I_e + I_i + I_{photo}. \quad (1)$$

If the total current I_{tot} is different from zero, the spacecraft will change its charge and thus its voltage V_{SC} with respect to the plasma electric potential by:

$$\frac{dV_{SC}}{dt} = \frac{I_{tot}}{C_{SC}} \quad (2)$$

where C_{SC} is the capacitance of the spacecraft. Based on this model, voltage-current characteristics can be calculated. The characteristics give an overview of the total current, the spacecraft receives, when it is charged at different voltage levels. In Figure 1 an example for such a V-I diagram is given.

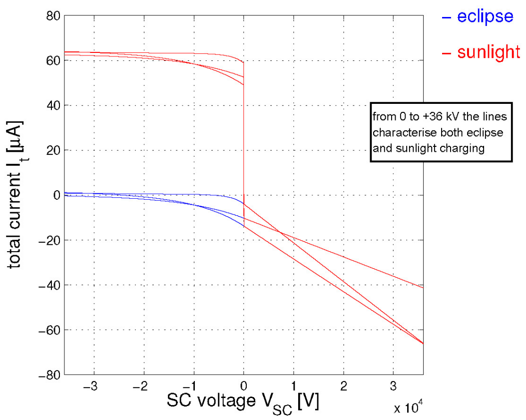

Figure 1: Total current in GEO for different spacecraft potentials, in different plasma environments, during sunlit and eclipse phases

2.2 Requirements

The exploitation of inter-satellite electrostatic forces requires the control of each spacecrafts total current. Therefore, the sum of all natural fluxes has to be equalized and an additional current dependent on the desired charge state must be provided. With the utilization of a charge actuation system Eq.(1) extends to:

$$I_{tot} = I_e + I_i + I_{photo} + I_{control}. \quad (3)$$

For being able to control the spacecraft charge regardless of its own charging state, several worst case scenarios have been simulated to determine the maximum possible natural currents I_e, I_i and I_{photo}. The highest desired spacecraft charges have been set to $\pm 2\ \mu$C, which translates into a maximum potential of $\pm$ 36 kV. Figure 1 displays the results, showing the highest expected total natural currents to be at $\pm 65\ \mu$A. While it is necessary to compensate for the natural currents, the control current should be as low as possible to keep the mass flow at small numbers. On the other hand the spacecraft shall charge in a given time span. Considering that, the currents demanded from the actuation system have been set to $\pm$ 73 μA. Assuming the satellite being at a certain potential, charges with an opposite sign must be accelerated to be able to actually leave the spacecraft. Thus the actuator must be able to provide an acceleration voltage of $\pm$36 kV. The requirements of the charge ejection system are summarized in Table 1.

As the spacecraft shall be charged in negative and positive direction the requirements are the same for both the electron and the ion ejection system.

- range of acceleration voltage $V_{acc} = \pm$ 36 kV
- independent adjustability of $I_{control}$ and V_{acc}
- fast adjustability of $I_{control}$ and V_{acc} in less than 1 s
- provision of emission currents $I_{control}$ in the order of $\pm$ 73 μA

Table 1: Requirements of the charge actuation system [5]

2.3 Definition of the actuator

In order to assess whether the requirements defined in the previous section can be fulfilled, we propose here a candidate system to be used for charge actuation. Furthermore an algorithm for the operation of this actuator is proposed.

After a survey of available charge mitigation techniques and μN ion thrusters in [5] it turned out, that the best suitable actuator for a charge control system is the Radio Frequency Ion Thruster (RIT), as it can be operated with a high bandwidth of emission currents and acceleration voltages and is capable to adjust these two parameters independently. The RIT will be used for the emission of positive charges. With the present technology the emission has a lower boundary in the order of 250 μA for the RIT-4 and - possibly available in future - 100 μA for the RIT-1. An electron gun has been selected for the emission of negative charges.

The approach presented here to operate the charge actuator builds on existing technology and uses the simultaneous emission of two currents, which differ in magnitude and sign. They comprise the charging current I_{ch} and the stabilizing current I_{st}. To charge the spacecraft, particles must be emitted such that $\frac{dV_{SC}}{dt}$ leads V_{SC} in the direction of the desired voltage. In this approach I_{ch} is emitted with an acceleration voltage V_{acc} equal to V_{des}.

E.g. to change the spacecraft potential from zero to a negative value, I_{ch} will consist of positive ions. As soon as the satellite reaches a negative potential, the ions are attracted by it. At $|V_{SC}| < |V_{acc}|$, the particles can escape. In the moment V_{SC} assumes V_{acc} the expelled charges can not leave the spacecraft anymore and therefore cease to further charge it. Thus the desired voltage can be achieved by setting V_{acc} to the value of V_{des}.

If, due to external events like sunlight, the spacecraft would float to a positive potential, the particles of I_{ch} can escape again, bringing the satellite back to V_{des}. This is true as long as the charging current outnumbers any natural current. For positive V_{des} the same method is used, but changing the flux from positive ions to electrons.

This approach lacks the ability to charge the spacecraft below the floating potential V_f. This is the potential, the satellite would assume without any artificial current. In general, the desired voltage can not be achieved if:

$$\begin{array}{rcl} |V_{des}| & < & |V_f| \\ sign(V_{des}) & = & sign(V_f) \end{array} \qquad (4)$$

This is extensively explained in [5]. To solve that problem, the stabilizing current I_{st} is introduced. It is simultaneously operated, possessing the opposite sign of I_{ch}. While the charging current is charging the spacecraft in direction from zero to the desired voltage, I_{st} pulls it to a zero potential. Considering the application of charge control the balance in Eq. 3 changes now to:

$$I_{tot} = I_e + I_i + I_{photo} + I_{ch} + I_{st} \qquad (5)$$

where $I_{control}$ is replaced by $I_{ch} + I_{st}$.

With the control scheme introduced in [5] the signs and magnitudes for both currents are set in such a way, that I_{tot} of Eq. (5) is zero when V_{SC} reaches V_{des}. Furthermore a V-I characteristic of the spacecraft-plasma system is created, which is stable around V_{des}. Thus the satellite is kept at its desired voltage level, even if the environmental parameters are changing. This is valid also for the worst case environmental conditions expected at GEO.

2.4 Performance evaluation

In this section the performance and applicability of the charge actuation system shall be evaluated in terms of the duration of the charging process, the accompanying residual forces and the mass flow. The time performance of the charging process has been evaluated by simulating the cases, where the longest charging times are expected. Therefore the control system is ordered to switch between the given limits of desired voltages from −36 kV to +36 kV and vice versa. The simulations take place in both sunlit and eclipse phases, experiencing worst case conditions of the space plasma, where the natural currents have their highest magnitudes. In Figure 2 the response of the system under various environmental conditions is shown. It can be seen that the highest charging time is not greater than 100 ms.

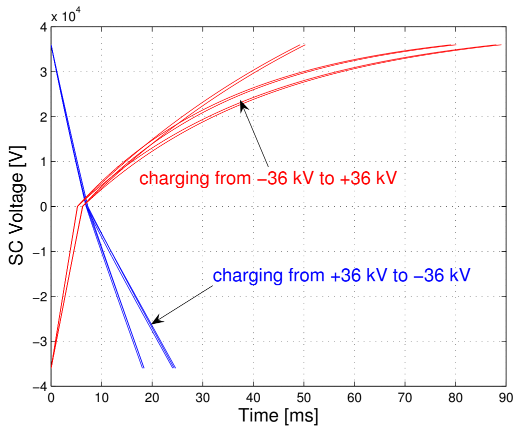

Figure 2: Charging durations for different worst case scenarios [5]

Besides the electrostatic inter-spacecraft forces, the emission of the control currents creates a repulsion force, which is considered here as completely residual. For the proposed devices the emitted species are electrons as negative charges and Xenon ions or protons as positive charges. For the evaluation the acceleration voltage of the ion thruster and the effect of the spacecraft potential field have been considered. The residual forces have been calculated for the case of formation keeping, where the spacecraft charges are expected to vary in small ranges and for the case of an acquisition maneuver with frequently changing V_{des}. In Table 2 the worst case residual forces due to emission of protons are given for a charge actuation system with the RIT-1 engine using hydrogen as ion delivering gas. The forces caused by electrons can be neglected. An extensive explanation can be found in [5].

	worst case formation keeping V_{des} is varying in small range
H^+	$\ll 1\ \mu$N constant (I_{ch}) -
	worst case acquisition V_{des} may change rapidly
H^+	5 μN short period (I_{ch}) 2.7 μN constant (I_{st})

Table 2: Worst case residual forces for RIT-1 using hydrogen as ion delivering gas

To prove the applicability of electrostatic actuation, the mass flow needed for charge emission shall be compared with the mass flow needed by the same actuator to generate the same amount of force, but working as a conventional ion thruster. It turned out, that for hydrogen usage, the electrostatic actuation is in advance for forces higher than 3.5 μN (RIT-1) and 7 μN (RIT-4). The case of hydrogen as ion delivering gas is displayed in Figure 2.

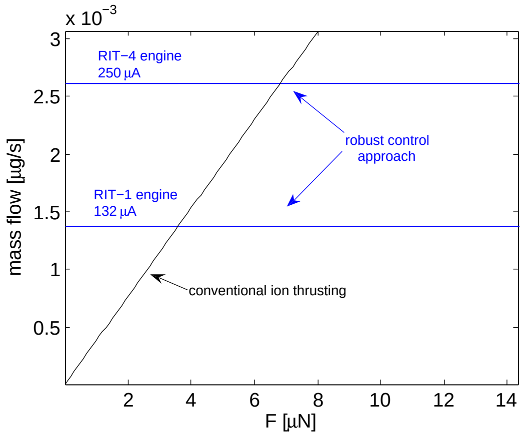

Figure 3: Comparison of mass flows between electrostatic actuation and ion thrusting

The evaluation shows, that all requirements of Table 1 are fulfilled and a fast and robust control of the individual spacecraft charges can be provided.

3 Charging strategy definition

Let us consider a swarm of N of satellites having mass m_i flying in a tight formation around a circular orbit with orbital angular velocity ω. Let $\mathbf{r}_i = [x_i, y_i, z_i]^T$ be the relative position of the $i-$th spacecraft with respect to the center of the swarm projected onto the relevant Local-Horizontal-Local-Vertical (LHLV) frame with the x axis pointing radially away from the center of the Earth, and with the z axis pointing in the direction normal to the reference orbit plane. Then the equation of motion of the $i-$th spacecraft belonging to the swarm can be approximated by the Clohessy-Wiltshire equations

$$\ddot{\mathbf{r}}_i + \mathbf{D}\dot{\mathbf{r}}_i + \mathbf{K}\mathbf{r}_i = \mathbf{u}_{des_i}$$

where:

$$\mathbf{D} = \begin{bmatrix} 0 & -2\omega & 0 \\ 2\omega & 0 & 0 \\ 0 & 0 & 0 \end{bmatrix}, \mathbf{K} = \begin{bmatrix} -3\omega^2 & 0 & 0 \\ 0 & 0 & 0 \\ 0 & 0 & \omega^2 \end{bmatrix}$$

and where, under the hypothesis of perfect actuation, $\mathbf{u}_{des_i}$ is the acceleration command issued by the on-board path-planning system. We assume that each spacecraft is able to exploit the thrusting actuation system to generate accelerations $\mathbf{u}_{t_i}$ in any direction and that the net currents I_i flowing from the spacecraft can be actively modulated to control the electrical charge q_i of each satellite. Then

$$\mathbf{u}_{des_i} = \mathbf{u}_{e_i} + \mathbf{u}_{t_i} = \frac{k_c q_i}{m_i} \sum_{j=1, j\neq i}^{N} \frac{q_j}{|\mathbf{r}_{ij}|^3}\mathbf{r}_{ij} + \mathbf{u}_{t_i} \quad (6)$$

where k_C is the Coulomb constant, $\mathbf{r}_{ij} = \mathbf{r}_i - \mathbf{r}_j$ and $\mathbf{u}_{e_i}$ is the resultant of the electrostatic forces acting on the $i-$th spacecraft. When the electrostatic force is exploited to provide part of the required control force, a coordination algorithm needs to be implemented because the dynamic of the swarm becomes highly coupled, i.e. a charge variation on the $i-$th spacecraft generates a variation in the forces acting on the other charged satellites. In this work we consider as main objective of the coordination scheme the reduction of total fuel consumption required to the whole swarm during the maneuver. Hence the following problem must be solved on line:

$$\forall t, \quad \min_{\mathbf{q}} \sum_{i=1}^{N} |\mathbf{u}_{t_i}|. \quad (7)$$

Other figures of merit could be also considered such as balanced fuel consumption. A more in depth discussion of this second problem is given in [5].

3.1 The charge feedback

In this section we introduce a charge feedback law strategy that provides a solution to the problem in Eq.(7). On the basis of the results described in section 2, we assume that active emission of charged particles is used to produce fast variations of the spacecraft charge and that a measurement of the level of charge of each satellite is available. Let us introduce the following charge feedback law for $i = 1, .., N$

$$I_i = \kappa_I [q_{des_i}(\mathbf{q}, \mathbf{r}_1, .., \mathbf{r}_N, \mathbf{u}_{des_i}) - q_i] \quad (8)$$

where κ_I is a scalar constant and q_{des_i} is the level of charge that the $i-$th spacecraft instantaneously wants to acquire and where $\mathbf{q} = [q_1, .., q_N]^T$ is the vector of the spacecraft actual charge values. There are different options to define the desired charge. In this work we assume that each spacecraft computes its individual desired charge trying to minimize its individual fuel expenditure e.g.

$$q_{des_i} = \mathbf{u}_{des_i} \cdot \mathbf{R}_i\mathbf{q}/|\mathbf{R}_i\mathbf{q}|^2 \quad (9)$$

where we define the actual configuration matrix $\mathbf{R}_i$ as

$$\mathbf{R}_i = \frac{k_c}{m_i}\left[\ \frac{\mathbf{r}_{i1}}{|\mathbf{r}_{i1}|^3}, \quad \dots \quad ,0, \quad \dots \quad , \frac{\mathbf{r}_{iN}}{|\mathbf{r}_{iN}|^3}\ \right].$$

Note that, given the actual charges of the other swarm members in $\mathbf{q}$ and the matrix $\mathbf{R}_i$, the direction of the electrostatic force acting on the $i-$th satellite is also given

$$\hat{\mathbf{u}}_{e_i} = \mathbf{R}_i\mathbf{q}/|\mathbf{R}_i\mathbf{q}|.$$

Then the charging strategy in Eq.(9) requires to assume that value of charge so that the resultant electrostatic force acting on the $i-$th spacecraft is the projection of $\mathbf{u}_{des_i}$ onto $\hat{\mathbf{u}}_{e_i}$. A first attempt to study the stability properties of this feedback law and to relate the equilibrium position of the dynamical system in Eq.(8) to the solution of Eq.(7) is given in [1].

4 Simulation Results

In this section a simulation is proposed to illustrate the performances of the proposed charge feedback law. The simulation integrates the following dynamical system

$$\begin{array}{l} \ddot{\mathbf{r}}_i + \mathbf{D}\dot{\mathbf{r}}_i + \mathbf{K}\mathbf{r}_i = \mathbf{u}_{des_i} \\ \frac{dq_i}{dt} = I_i = \kappa_I(q_{des_i} - q_i) \end{array} \quad (10)$$

with $i = 1, .., N$, and where, according to Eq.(6), we assume that the hybrid actuation system can instantaneously deliver the desired acceleration. The fuel savings triggered by the use of electrostatic actuation is clearly dependent from the maneuver that is performed i.e. from the time history of the desired acceleration $\mathbf{u}_{des_i}$ generated by the path planning algorithm. In this paper we described the derivation of the charging strategy assuming that a path planning algorithm that computes a suitable $\mathbf{u}_{des_i}$ is already available. In particular, in the simulation below, the path planning technique introduced in [2] is used to calculate the desired acceleration vector for each spacecraft. It is shown in [5] how such path planning technique can be easily adapted to generate desired acceleration signals suitable for the application of the electrostatic actuation system. In Figure 4 the trajectories of a three spacecraft formation performing a deployment maneuver are shown. The satellites are supposed to have 50 kg mass and the center of the formation moves along a GEO orbit. Both the initial charges and positions have been randomly selected respectively in the sets $|q_i(t=0)| \leq 0.4\ \mu$C and $|\mathbf{r}_i(t=0)| \leq 5$ m and are listed in Table 3. The final desired configuration is an

	Position(m)	Charge (μC)
SC1	$[4.12, 0.84, -1.15]$	-0.28
SC2	$[-0.61, -1.65, -1.06]$	-0.25
SC3	$[-3.51, 0.81, 2.21]$	0.2

Table 3: Initial conditions for the three spacecraft deployment maneuver.

equilateral triangle of approximately 35 m size. Finally, a saturation cycle has been included in the proposed charging strategy to limit the achievable maximum charge to 2 μC.

In Figure 4 and in Figure 5 the three dimensional and planar view of the spacecraft trajectories during the maneuver are shown.

At the very beginning of the simulation the spacecraft are at rest condition and must acquire a certain radial velocity to reach the final desired position. Therefore in the first 150 seconds all the spacecraft assume a positive charge so that the electrostatic repulsion force can provide a considerable portion of the initial required acceleration. This is shown in the lower plot of Figure 6 where the time variations of the spacecraft charges at the very beginning of the simulation are displayed. After 250 s the satellites must start the deceleration process. For this reason attractive electrostatic forces must be established. The spacecraft marked with blue and green line experience during their motion a gravitational and inertial force along the x axis that, if not properly counteracted, would make their trajectories divergent. On the other hand the satellite marked with the red line moves approximately along the y axis where it experiences only an inertial force that pushes it toward the direction of the negative x. The charge time history for the whole simulation is presented in the upper plot of Figure 6. In order to better explain the behavior of the charging strategy, the electrostatic forces acting on the spacecraft after approximately 0.2 orbital periods of simulation are sketched as arrows in Figure 5 (electrostatic forces not in scale). This plot shows that the spacecraft marked in blue attracts the other two members of the swarm. This charge sign selection indeed allows to counteract part of the inertial and gravitational forces acting on the

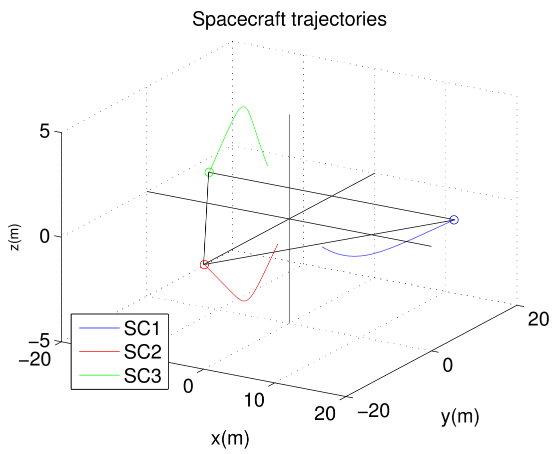

Figure 4: Three spacecraft deployment maneuver: 3D view.

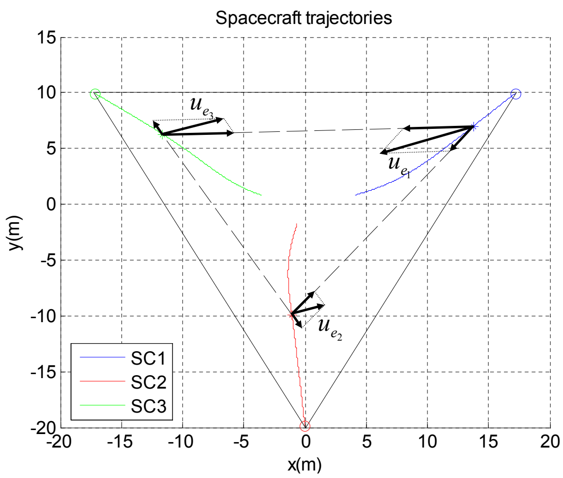

Figure 5: Three spacecraft deployment maneuver: planar view with frozen configuration after 0.2 orbital periods (electrostatic forces not in scale).

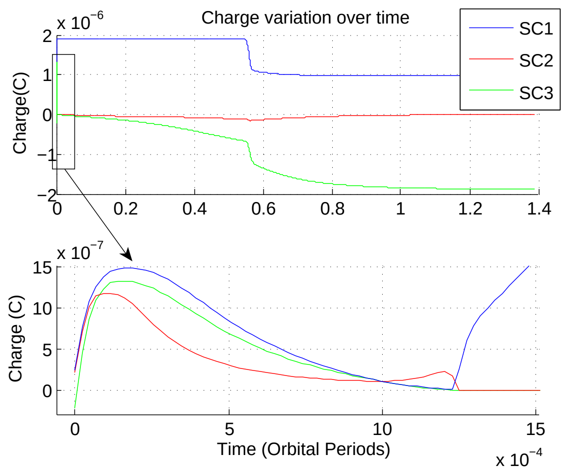

Figure 6: Three spacecraft deployment maneuver: charge variation.

satellites with electrostatic forces. Moreover the charging strategy selects the charge magnitudes trying to minimize the residual thrusting acceleration needed to steer the swarm. The largest disturbing forces acting on the swarm are the gravitational and inertial forces along the x direction on the spacecraft marked in blue and green. In order to counteract this effect the blue and green spacecraft assume charges q_1 and q_3 respectively that yield a large charge product q_1q_3. Therefore a large attractive electrostatic force is established between the two spacecraft. However, $q_1 > q_3$. Hence the attractive force between the spacecraft marked in blue and the one marked in red is larger in magnitude with respect to the repulsive force between the green and the red spacecraft. This implies that a portion of the electrostatic energy is also used to reduce the velocity of the spacecraft marked in red along the y direction. At the very end of the simulation all the spacecraft are again at rest condition. Note that the charge of the spacecraft marked in red converges to a null value since, at rest condition, no inertial force is anymore acting on that spacecraft. In Figure 7 the Δv required to per-

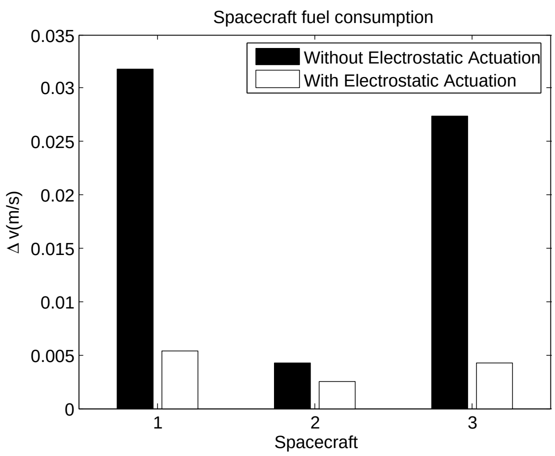

Figure 7: Fuel consumption of the three spacecraft deployment maneuver.

form the whole maneuver with and without electrostatic actuation are displayed. The proposed charge control algorithm allows to save the 81% of the total Δv required to perform the maneuver.

5 Conclusions

In this work we investigate the possibility to steer a swarm of satellites by means of a hybrid thrusting and electrostatic actuation system. First a scheme for a possible charge actuation system has been developed. Simulations have shown that any desired level of charge in the given range can be fast achieved and maintained with high stability. A charge feedback law has been then introduced that, used in conjunction with a decentralized path planning algorithm, allows to increase the fuel efficiency of acquisition and reconfiguration maneuvers for swarm of satellites. In the future the following investigations are recommended for the further development of the concepts introduced in this work:

- Development of a dedicated charge emission device capable of emitting currents in large amperage and energy range.
- Development of a more detailed plasma-spacecraft interaction model in order to have a better estimate of the worst case scenarios especially for high spacecraft voltages.
- Stability analysis of the charge feedback law in order to develop a more reliable algorithm for charge control.

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