Deployment and Maintenance of Solar Sail-Equipped CubeSat Formation in LEO

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Abstract

We study small satellite formations comprising a number of spacecraft equipped with solar sails, that can be employed as sunlight reflectors. In the appropriate lighting conditions such formations can be visible from Earth and provide graphic images in the sky. The formation's visibility is defined in terms of magnitude, which entails the requirements for the solar sails' reflective properties and geometry for any given orbit altitude. The number of satellites and the distances between adjacent satellites in the formation are estimated given the desired image characteristics. The study then focuses on the coupled orbital and attitude dynamics of the satellites. Numerical experiments test whether the use of the differential drag technique and the solar sail propulsion can help minimizing propellant consumption.

Keywords: CubeSat, formation flying, satellite flares, solar sail, differential drag.

Introduction

Satellite formation flying has long become a point of vivid interest for researchers and engineers for reasons of mission flexibility, cost reduction, and enhanced performance. It implies replacement of a single large satellite by a set of coordinated smaller and simpler spacecraft, maintaining a stable close distance configuration and sharing signal processing, information exchange, payload, and other functions. A good state of the art review providing an overall idea of various uses small satellites formations may be put to and listing the missions that have been performed by satellite formations may be found in [1].

This paper revisits the space mirror concept [2] and provides a feasibility study of its implementation by a CubeSat formation. The idea of a mirror in space to reflect sunlight and thus generate power and light on Earth was proposed in 1928 by Oberth who postulated a space-manufactured 5mm thick mirror using sodium for the reflective layer, orbiting Earth at 1000-5000 km altitude [3]. The first practical step towards this end was the Russian Space Mirror Project ‘‘Znamya’’ which was to illuminate high latitude earth regions during winter months [4]. After the successful deployment of the segmented disk mirror (20 m in diameter) made of 5 mm thick Mylar film, it produced on the Earth's surface the spot of light of about 5 km in diameter, which travelled across the Earth’s surface at 8 km/s. The brightness of the mirror as seen from the Earth was reported to have been similar to that of a single full moon.

Our study is focused on designing a mission of CubeSats, equipped with reflectors and coordinated in a formation to produce in the sky a graphical image, which is seen from a given point of interest (POI) on Earth. The formation is deployed in LEO and the sunlight reflecting surfaces act simultaneously as solar sails and drag augmentation devices, and therefore can be employed to control the relative positions of satellites in the formation. The paper has the following structure. Section 1 presents the problem formulation and main assumptions. Section 2 provides the orbital motion analysis in terms of formation's visibility in certain points of interest. Section 3 deals with the requirements for individual spacecraft geometry as well as the formation build up. Finally, Section 4 unfolds the estimates of what it takes to control the relative positions of spacecraft within the formation during the mission lifetime.

1. Objectives and assumptions

The objective of the study is to design a mission of CubeSats, flying in a formation and carrying sunlight reflecting surfaces with dimensions to be defined, and orbiting the Earth to produce a pixelated image at a given POI visible for a naked eye. We shall assume Moscow as the POI and two letters "S" and "k" as an example image to be demonstrated from the orbit. Each satellite is considered as a single pixel of a complex image.

Obviously, the quality of the image and the time during which the image is seen from a POI essentially depends on the attitude control quality. We shall not discuss the attitude control in this paper, assuming that the stabilization is ideal and each reflector keeps its planar form, while pointing accuracy is about 5°. Another assumption is made regarding the orbital altitude: in this study, only LEO satellites are considered, because only this way it might be realistic to expect the spacecraft to be cheap enough, or to be launched at all.

It is also important to have a formal understanding of what is understood by “visibility”. In theory, human eye can distinguish a star with a magnitude of 6 in the clear sky at night. Adding a few geometrical considerations, we define visibility by the set of conditions below:

- (1) the satellite should be in the direct line of vision from the POI;
- (2) the satellite should be lit by the Sun in order to be able to reflect the light to the ground;
- (3) the Sun must be below some established low elevation as seen from the POI. In this work, the limit is set to 5 degrees of elevation;
- (4) the spacecraft must be in the darker part of the sky when it passes, so the angle between the directions from the POI to the Sun, on one hand, and the satellite on the other, should not be less than a certain value, which was in this case set to 25 degrees;
- (5) the pixels must be clearly visible for the naked eye, and the message should be readable. In this work, this is defined by requiring the magnitude of -8.0 or better (the magnitude of the well known Iridium satellite flares varies between -8.0 and -9.5).

Based on this set of conditions, a model is built in the subsequent sections to estimate how frequently and to what extent a moving satellite is seen from a given POI.

2. Orbit Selection

For a mission where it is required that the satellite appears above a given point on the ground at given times, it is essential to make sure that the orbit is such that the satellite is indeed observable and visible. The set of mission requirements and assumptions given in Section 1 leave us with a narrow set of points in time and orbital position where the demonstration is possible. There can be no view at midnight (in the latitudes well below the polar circle), because the satellite, if in the direct line of sight, is too far into the umbra part of the orbit, and the 2nd condition is not met. During the day, the Sun lights up the sky rendering the observation impossible due to the 3rd condition. Thus, the orbit of the satellite should pass over the POI at the time when the POI is near the terminator line. The orbit, therefore, should come close to the terminator line at least at the latitudes corresponding to POI.

In Fig. 1, such situation is shown: the satellite is in an inclined orbit (red line), and both the satellite (cyan dot labeled with magnitude value) and the POI (labeled "Moscow") have passed the terminator (dashed yellow line) into the unlit part of the Earth, but the satellite is still lit. The yellow lines show the direction of a Sun beam from the Sun to the spacecraft, and then to the POI. It is assumed that the satellite keeps the attitude so that the beam is reflected onto the POI whenever it is possible.

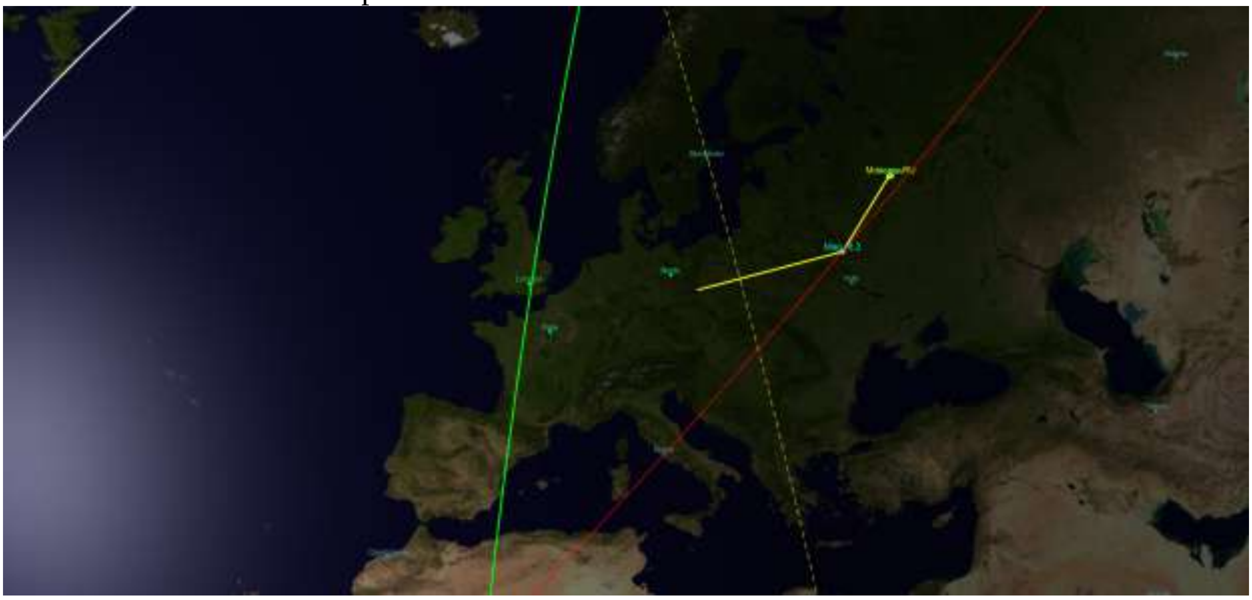

Fig. 1 View of the orbiting satellite and the POI

If one were to estimate how often a satellite is seen from a point depending on the orbital inclination of the satellite, it could be established that in order to maximize the time the satellite spends visible from a certain point, the inclination of its orbit should be slightly bigger than the latitude of the point. In addition, the bigger the latitude, the better mean observability can be achieved.

Another strategy might be to align the orbit with the terminator line itself. If the orbit remains in accordance with the terminator, the amount of views is more stable and easily controlled by selecting the correct LTAN. As the terminator line moves in the inertial frame in accordance with the sun, the orbit of interest is, of course, a Sun-synchronous one (SSO). While a generically inclined orbit will inevitably be driven away from the terminator by $J2$ effect, the same effect will keep the sun-synchronous orbit aligned with the terminator. This is a perfect solution for the equatorial areas, where the terminator line stays relatively stable during the year. In the areas closer to the pole (of which the POI in this experiment is one) however, the situation gets worse as the terminator changes its apparent inclination, of which the satellite is incapable. This effect renders the satellite unobservable from the POI for an amount of time depending on a) how big is the absolute value of the latitude of the POI, and b) at which time of the year the "aligned" time interval (when the orbit is close to the terminator) is centered.

The latter parameter might be played with in case the duration of the mission is expected to be 1-3 months. If the "aligned" interval roughly corresponds to the winter, a substantial amount of views is available during that time. During this period, the message is demonstrated at the POI twice a day: once in the morning, and once in the evening. However, if the mission is flown in summer, when it is generally sunnier in the Northern hemisphere, there are fewer views, and an orbit with inclination of about 60 degrees should also be considered.

3. Payload Design

Let us consider a single satellite with a large sunlight reflector orbiting the Earth (Fig. 2). The scattering angle α is the included angle of the Sun measured from the Earth, d is the distance between reflector and ground spot, A_r and A_{gs} are the areas of the CubeSat reflector and the ground spot respectively.

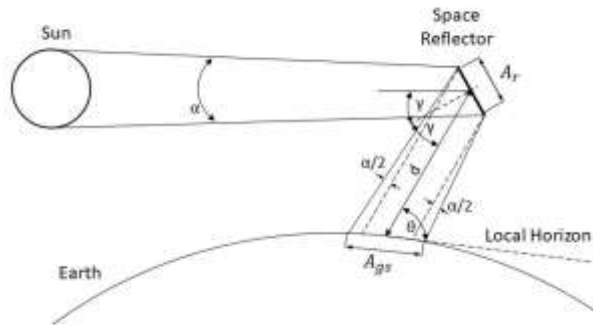

Fig. 2 Scheme of the experiment

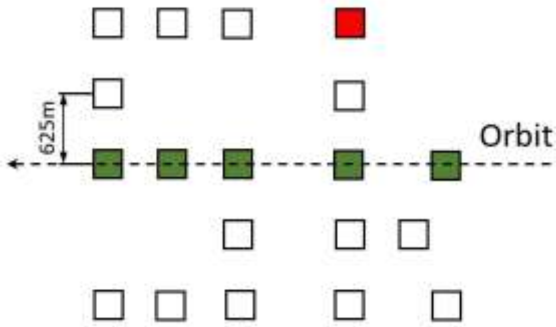

Fig. 3 Formation line-up at 600 km orbit

The magnitude of the reflector is usually calculated as the ratio of incident light intensity to a reference intensity [5]

$$m = -2.5\log(I / I_{ref}), \tag{1}$$

The intensity of the light at the POI is given by [5]:

$$I = I_0 A_r \rho \tau \cos\gamma \sin\theta \Big/ 4d^2 \tan^2\frac{\alpha}{2}, \tag{2}$$

where $I_0 = 1360\ \mathrm{W/m^2}$ is the average intensity of solar energy at the Earth distance, ρ is the mirror reflectivity coefficient, γ is the incident angle of solar rays, θ is the elevation angle of the satellite; τ is the atmospheric transmissivity [5]:

$$\tau = 0.1283 + 0.7559e^{-0.3878\sec(\pi-\theta)}. \tag{3}$$

Thin Mylar film coated with aluminum is chosen as the reflector material because of low weight and high reflectivity coefficient $(\rho = 0.92)$. Mylar solar sails have been tested in space missions [3, 6]. The reflector, which is deployed and maintained by a rigid support structure, is assumed to be of square shape as in [6].

Let us now estimate the size of the reflector necessary to ensure the required magnitude. Eq. (2), given light intensity and orbit altitude, allows expressing the reflector area A_r as a function of angles θ and γ. The extreme values of these two angles (yielding the maximum A_r) are $\theta_{\min} \approx 30^\circ$ (see conditions 3 and 4 in Section 1) and $\gamma_{\max} \approx 60^\circ$ (conditions 2 and 3). Hence, we obtain the characteristic linear size of the reflector in the range of 2-2.6 m for SSO altitudes of 600-800 km. Seeing the need to pack the solar sail material along with the supporting beams into the CubeSat structure along with all other necessary satellite subsystems (including attitude control and propulsion units), we come to choose the 12U as a suitable CubeSat size for the designed formation.

Let us recall now that our task is to produce a pixelated image of two letters, which means that the distances between any two CubeSats in the formation must be such that they are distinguished as independent pixels. Human eye resolution is known to be about two arc-minutes, which for the specified SSO altitudes yields the minimum distance between satellites in the formation ranging from 625 to 812 m. With these distances, it takes 19 satellites to produce the required image of two letters (as shown in Fig. 3, the satellites traveling along the reference orbit are colored green, one the formation satellites, which are farthest from the reference orbit is colored red). It is worth mentioning that the relative height of the letters (or the font size) is 26% of the Moon's size (it being 31 arc minutes). The quality of the image is still acceptable when the relative positions errors do not exceed 30 m, which is well within the range of on-board GPS-module position determination.

On the other hand, it will require 76 satellites to create a continuous image of the Moon (same size, same magnitude).

The estimates made in this section can be refined by taking into account various atmospheric effects, which is a definite plan for the future work. At the moment, we shall accept these estimates as a reasonable first approximation, especially seeing the fact that the estimates pass the sanity check by comparison against the Iridium flares, produced by antennas, whose dimensions are 1.88×0.86 m, and the orbit altitude is 780 km.

4. Formation Control

It follows from the previous sections that the formation to be controlled consists of identical 12U CubeSats, each equipped with a reflector $2 \times 2\,\text{m}$ and ion thruster capable of operating at a specific impulse I_{sp} range of 2000 to 5000 s. The estimated mass of each CubeSat is 18 kg. The formation is deployed along the circular Sun-synchronous orbit (near the terminator as suggested by the analysis in Section 2) with altitude of 600 km and inclination 98°. The start date is February 25th, 2019 (important for the atmospheric density and the Sun position).

A simple estimate of the propellant required to maintain the formation can be obtained from the Hill-Clohessy-Wiltshire equations, describing the linearized dynamics of the relative motion (e.g. of a satellite marked red in Fig. 3) in the orbital reference frame with the origin at some point on the circular reference orbit (any reference satellite marked green in Fig. 4)

$$\ddot{x} = -2\dot{z}\omega_0 + \Delta f_x, \quad \ddot{y} = -y\omega_0^2 + \Delta f_y, \quad \ddot{z} = 2\dot{x}\omega_0 + 3z\omega_0^2 + \Delta f_z, \tag{4}$$

where Δf is the difference of disturbing forces (atmospheric drag and solar radiation pressure) acting on the two satellites. For this estimate we shall assume that all satellites in the formation when not seen from the POI maintain such attitude (each with random 5% error) that the plane of the reflector coincides with that spanned on the vectors of incoming airflow velocity and the Sun direction. Passage over the POI, naturally, requires the attitude maneuvers to reflect the sunlight to the POI. Making multiple simulation of the two satellites relative dynamics along the chosen orbit, we obtain the mean propellant consumption of 2.29 kg in 30 days (for a satellite positioned at 1250 m away from the reference orbit) and 1.14 kg (for the position 625 m away).

The specified control laws were implemented in a simulator taking into account the Earth's gravity (first 4 harmonics), Luni-Solar gravity perturbations, atmospheric drag force (with the NRLMSISE-00 model for atmospheric density) and the Solar radiation pressure. Numerical experiments showed that the formation could be maintained within the specified position errors for the mission duration period. Two additional control algorithms were implemented in the simulator and studied: differential drag LQR controller [7], that uses the reflector as the augmented drag device; and solar sail propulsion method to control the relative positions in a formation [8]. It turns out that whereas the first method allows optimizing the consumed propellant and extending the mission period, the second method does not show any benefits in the propellant consumption, the orbit being too low for SRP to dominate over other disturbances. Another obvious means to prolong the mission duration is formation reconfiguration that brings the satellites with the highest level of consumption to lower levels of consumption (i.e. close to the reference orbit). Such maneuvers aided by the use perturbations can take as much as two orbital periods, during which no messages are seen at the POI, however the mission duration is doubled for the formation specified in Fig. 3.

Conclusions

A feasibility study is conducted of deploying and maintaining in LEO a formation of CubeSats with sunlight reflectors used as drag augmentation devices. The study showed that such a formation may be deployed at SSO and reflect sunlight visible with a naked eye from a point of interest on Earth. With the proper size of reflectors the formation can demonstrate graphical images visible from any given POI or even produce an artificial moon of the same size and magnitude as the real one (as seen by a human eye). It is worthwhile noting that the estimates made in the study may be refined. For instance, a better model of atmospheric effects can be employed in defining satellites' visibility and, therefore, reflector's dimensions. Attitude control system must be integrated with the model to compensate for solar sail vibrations inevitable, e.g. when thrusters start working [9] or some attitude maneuver is performed. The information exchange between the satellites is also an important factor that must be taken into account when implementing the formation control algorithms. Finally, collision avoidance with external objects needs to be considered as the SSO region is heavily populated with large space debris objects.

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