

A Very Low Altitude Constellation For Earth Observation

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Abstract:

This paper discuss the possibility of flying a constellation of earth observation satellites at very low altitude, compare with those that are presently in service, with the objective of enhancing service performance and reducing cost of the satellite constellation, hence making it more efficiency and profitable. The concept of a very low altitude earth observation satellite constellation is now made possible with the advance of a new type of propulsion technology, the electric propulsion, which allow a considerable reduction of propellant mass needed onboard a spacecraft for the same mission delta-V requirement.

Keywords: Earth Observation, Electric Propulsion, Satellite Constellations, Low Orbital Altitude.

I. INTRODUCTION

With the advance of space satellite and sensor technologies, the earth observation market sees an increasing growth over the past couple of years. This is also driven by the increase demand of satellite acquired information regarding the conditions of the Earth. While in the past such information were mostly used by military and government agencies such as for the purpose of defense and weather forecast, today it is common for individuals to utilize these information from earth observation satellites, one example would be the popularity of Google Earth.

The objective of this project is to investigate the possibilities of flying earth observation satellite constellation at very low altitude, with the help of electric propulsion technologies. The potential benefits of such a scheme include the enhancement of image resolutions and revisit time, reduction in size and weight, which will leads to reduction in cost and increase of revenue, hence profit.

II. COMMERCIAL MARKET ANALYSIS

From our research into the earth observation market, two key demand drivers emerged. They are the demand for high resolution imagery and the timeliness of data delivery. These two demands can be met precisely by satellites flying at low altitude, because the orbital period decreases with orbit altitude, (this means that the satellite make more orbits in a given time), and the resolution of the sensor increase with decrease altitude as well.

The reduction in orbital period can be seen from the

well-known Kepler equation, which relate the semi-major axis of the orbit to the orbital period.

$$\frac{T^2}{a^3} = \frac{4\pi^2}{\mu}$$

Where T is the orbital period, a is the distance of the semi-major axis, μ is the earth gravitational parameter. In the case of circular orbit, the distance of the semi-major axis is constant, and can be taken to be the distance from the satellite and the center of the Earth.

The improvement of the image resolution with reduction in altitude can be seen from the Rayleigh diffraction limit:

$$X' = 2.44h \frac{\lambda}{D}$$

Where X' is the image resolution, h is the orbit altitude, λ is the wavelength of the electromagnetic wave that the sensor is capturing, and D is the diameter of the aperture. This relation is for nadir pointing sensor, but the relation between resolution and altitude still hold. It can be seen that to enhance the resolution (to have a smaller value) for a given wavelength, would require the diameter of the aperture to increase, or the altitude to be reduced. The approach of reducing altitude instead of increasing the aperture diameter would allow a smaller aperture to be used, hence achieve saving in size and weight.

Fig 1 given by [1], the global market for earth observation is expected to grow, in terms of raw data, processing service, and final information products.



Fig 1

III. ANALYSIS TOOLS DEVELOPED

A. Orbit Decay and Propellant Consumption Analysis

The purpose of this tools is to analysis the effect of atmospheric drag on the satellite flying at low earth altitude, with the goal of finding the propellant consumptions or delta V requirement. This is a software tools developed using excel, based on the mathematical model of low thrust trajectories from orbital mechanics. The set of differential equations from this model is numerically integrated to obtain the results as a function of time. The major assumptions are (1) circular orbit, (2) drag is predominately in the circumferential direction, (3) constant thrust, (4) the thruster is turned on when the satellite fall below a set altitude allowance, and turned off when the satellite reaches its target altitude, (5) the density model is taken from the NRLMSISE-00 density model. Fig 2a and Fig 2b are the sample results for altitude without and with thruster firing:

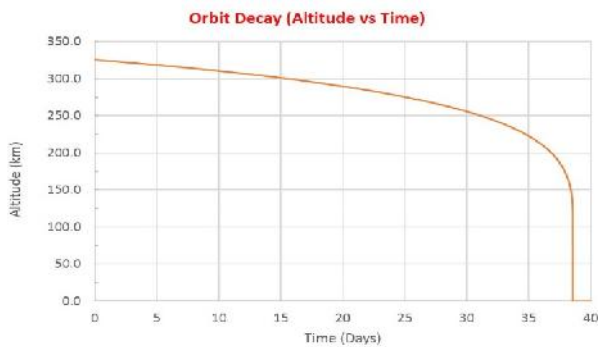


Fig 2(a)

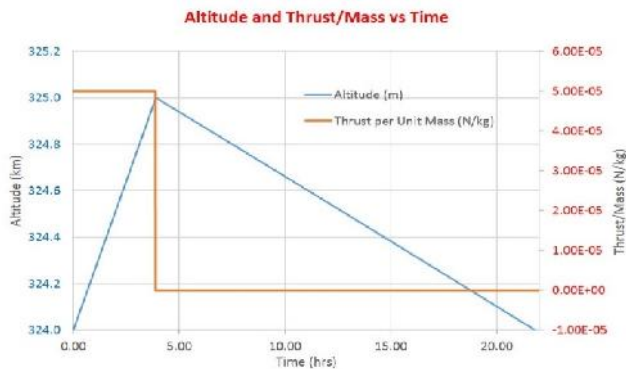


Fig 2(b)

B. Satellite Constellation Performance Analysis

Two important performance parameters of a satellite constellations are the revisit period and dwell time. Revisit period is the time period between passages of satellites over a ground location. Dwell time is the time that the satellite is in view from the ground during a single passage. Hence a software tool has been developed to simulate these parameters of a satellite constellation.

This is a software tools developed using Matlab, the positions of each satellites as well as a location on the earth

surface, for example a city, are calculated at every minute in time. In every minutes the distance between each satellites and the earth city is compared with a minimum detection distance to test if the city is in view by any satellites. For the case of a passive sensor (one that work only if the target is to be illuminated by the sun), a detection only occur if 1) distance between satellite and city is less than minimum detection distance, and 2) the city is illuminated by the sun. For the case where active sensor is used onboard the satellite, only condition 1 need to be satisfied. In this way we recover the time history of the revisit period and dwell time. The assumptions used in this analysis are (1) circular orbit, (2) the line of nodes of the orbital planes are evenly spaced on the equatorial plane, (3) the satellites are evenly phased in every orbital plane, (4) the minimum elevation angle (as view from the earth) is 20deg, which affect the minimum detection distance, (5) only perturbation due to the earth oblatness is taken into account.

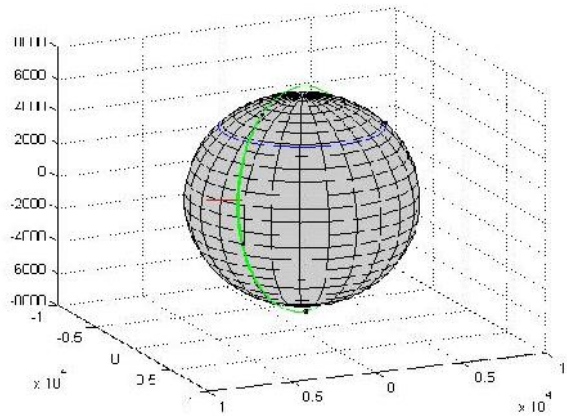


Fig 3

Some of the sample results are shown below:

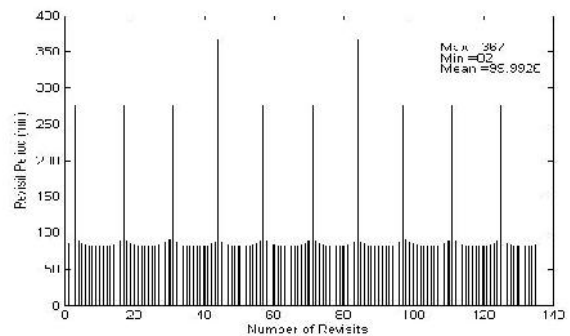


Fig 4(a)

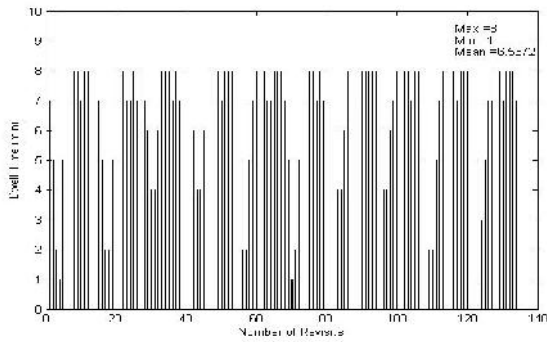


Fig 4(b)

IV. ORBIT AND CONSTELLATION DESIGN

The design of a satellite constellation means addressing the questions such as, what is the lowest altitude possible for the satellites to fly. How many orbital planes should the constellation has. How many satellite should be placed on each orbital plane? What is the orbit altitude and inclination? The solution to all of these question should lead to the most efficient and profitable earth observation satellite constellation.

A. Lowest Altitude Limit

In order to assess what is the lowest orbit altitude a satellite can be flown, the orbit decay and propellant consumption tools were used to study this problem and the results were shown in Fig 5

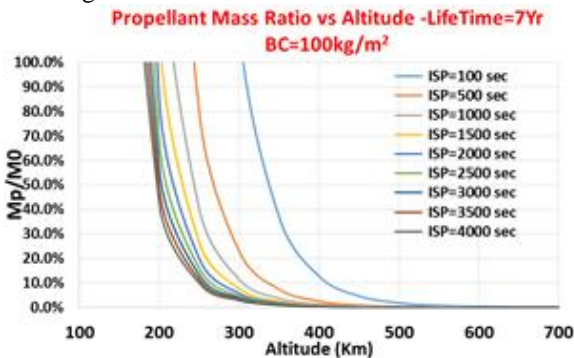


Fig 5: Propellant Mass ratio vs Altitude

The study were carried out with the assumption of a ballistic coefficient of 100kg/m², and a satellite mission life of 7 years. The results were shown for propellant mass to initial mass ratio verse different orbit altitude and for different Isp of the thruster. It can be observed that the lower the altitude the higher fraction of the satellite mass has to be propellant, which leave less rooms for other systems of the satellite. However the use of high Isp propulsive system mitigate these problem. This is the reason for the choice of electric propulsion technologies, because it is capable of offering thruster with higher Isp value, (typically range from 1000sec to 4500sec), than any other type of propulsive concept. From the result shown the lowest altitude limit is set at 250km.

B. Number of Orbital Planes

In order to decide the number of orbital plane appropriate

for the constellation, it is of interest to investigate how the number of orbital plane affect the performance of the constellation, namely the revisit period and dwell time. To facilitate this investigation, four arbitrary, but different constellations are selected to carry out a parametric study. These constellations are shown below in Fig 6.

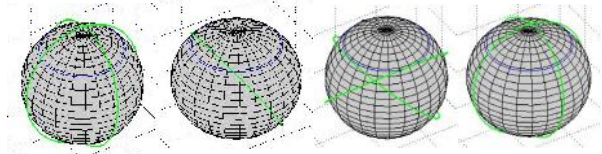


Fig 6: Different constellation architectures

Each of these constellation have different values for their parameters such as orbit altitude, inclination, number of orbital planes, and number of satellites in each orbital plane. Each of these constellation is referred to as an architecture. Two of these architecture are having sun-synchronous orbit. Using the software tools developed, the following results were obtained:

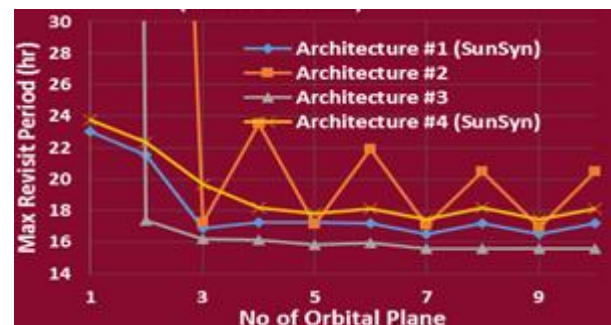


Fig 7: Maximum Revisit Period vs Number of Planes

From the analysis it was found that when passive sensor is used, sun-synchronous orbit outperform the non-sun-synchronous orbit at low number of orbital plane. However it is possible for the non-sun-synchronous orbit to catch up in terms of performance with the sun-synchronous orbit when more orbital planes are used. However the additional cost in terms of launch associated with the use of more orbital planes does not outweigh the marginal benefit (in terms of 7 hours of maximum revisit period improvement) of using non sun-synchronous orbit with more orbital planes. Hence sun-synchronous orbit with only one orbital plane was selected. The selection of sun-synchronous orbit means that there is now a constraints between the orbit altitude and inclination. Once the orbit altitude is determined, the inclination will be fixed to preserve the sun-synchronous condition.

C. Orbit Altitude

The selection of the orbit altitude was made with the goal of maximizing the profitability of the constellation. As such, an analysis was carried out to analyze the revenue and cost of the constellation as the orbit altitude is varied. The revenue consider is the amount of image that the satellite can

download to the ground station, together with the resolution of the image, and hence the price at which these image could be sold in the market. The cost considered was the cost associate with the mass of the satellite, as taken from an appropriate cost model, which is a function of orbit altitude base on the requirement such as telecommunication, power, attitude control, etc. Three existing commercial off-the-shell sensors with different resolutions at a reference altitudes were chosen to study also the effect of how difference in resolution, (or aperture diameter) of the sensor affect the profitability. Consideration was given to the constraint that the satellite can only download the amount of data in a time limited by the dwell time of the satellite passage over the ground station. The following results were obtained from the analysis:

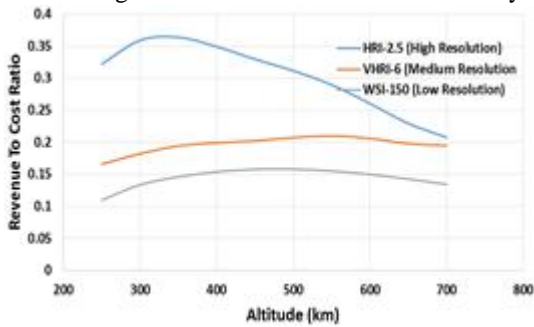


Fig 8: Revenue to Cost Ratio vs Altitude

The revenue-to-cost ratio is used as an index to indicate relative profitability across different altitude and for sensors having different aperture diameter (or resolutions at fixed altitude). It can be observed that the profitability increase as higher resolution sensor is used, also the optimum altitude decreases as the resolution of the sensor is increased. In order to achieve a high profitability, an orbit altitude of 325km was chosen, with high resolution sensor.

D. Number of Satellite in Each Orbital Plane

The selection of the number of satellites was based on the performance consideration, in terms of maximum revisit period. Again with the help of the software tool that was developed, the results was shown below:

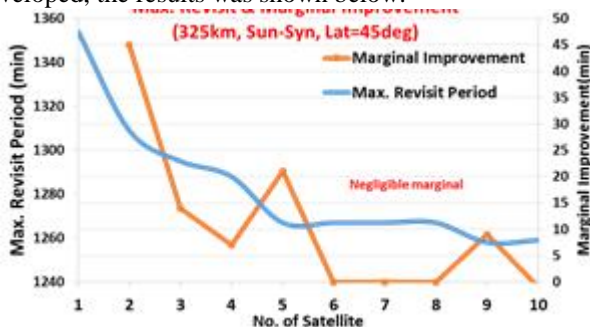


Fig 9: Maximum Revisit Period vs Number of satellite

It was found that the marginal improvement of maximum revisit period after the addition of the fifth satellite is negligible. Hence the number of satellite for each orbital plane was set at five.

V. SUBSYSTEMS AND BUDGETS

A. Power Budget

In analyzing the power budget, consideration has been given to minimize the mass of the solar array and batteries without reducing the performance of the spacecraft in terms of

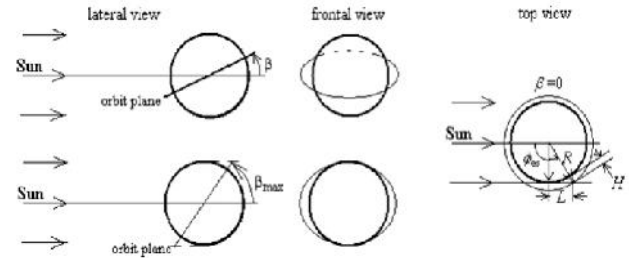


Fig 10: Beta Angle [3]

Image service. The orbital parameter have been decided and the final orbit is chosen to be Sun-Synchronous orbit. In Sun-synchronous orbit the eclipse time and the daylight time remains same throughout the year [2], eclipses time and daylight time strongly effects the sizing of solar array and the batteries and this eclipse time and daylight time is effected by the orientation of the satellite orbit plane with respect to the Sun-Earth line measured by angle denoted by beta in the Fig: 10. An analysis has been done by using the relations [3] to calculate the eclipse time as a function of beta angle.

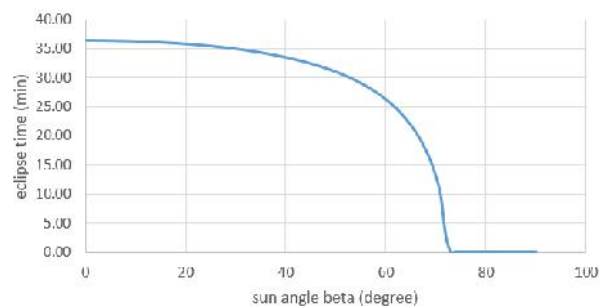


Fig 11: Eclipse time vs Beta angle

We see from Fig 11 that when beta angle is 72 degree, the satellite is eclipse free and this may be the best beta angle for designing solar array without the need to size the batteries since the orbit corresponding to this beta angle is always illuminated by the Sun but, by choosing beta angle of 72 degree or more, we degrade the image quality since the optical sensor will encounter a lot of shadow on the ground track. So, an optimal beta angle has to be chosen corresponding between minimum eclipse time and maximum eclipse time. By taking into account the illumination of the earth surface by the Sun, beta angle of 60 degree has been selected as the design parameter for sizing the solar array and batteries. In the next step, assuming the solar arrays to be sun pointed before going into earth observation mode, using the efficiencies of path from solar array to loads [4], and referring to [5] to get an average efficiency of Gallium Arsenide solar cells at the End of Life and following [6]. We get a preliminary approximation for the size of Solar Array and

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I

Operating power during daytime	276 W
Operating power during eclipse time	244 W
Area of Solar array	1.3 m.sq
Mass of Solar array	4 kg
Battery Capacity	367 w-hr
Mass of the battery	2.4 kg

B. Structure

The primary structural elements have the most mass budget among structure mass budget in comparison with secondary and tertiary structural elements and the whole satellite structure design in early stages can be based on primary structures design.

Honeycomb solution have some advantages, for example very low weight, high stiffness and durability and cost savings, it reduces the needs in terms of secondary structures. Composite allows to easily attach components directly to structural panels, and this is ideal to reach high degree of flexibility in the management of internal envelops and therefore to have a primary structure the simplest and the most light possible [6]. The primary structure mass will be only indicatively quantified according to Proba Vegetation experience (European Earth observation satellite). Material characteristics have been found in [7]:

II

Materials	Material density, [kg/m ³]	Modulus of elasticity, [GPa]	Average Strength, [MPa]	Thickness [mm]
Al 2024	2780	70	270	0.8 inner 0.4 outer
Al 3003	54	0.540	2.5	10.8 core

The structural shape of the satellite is configured in order to reduce the cosine loss encountered by the solar array while pointing the sun. For the preliminary design stage the satellite is shaped as frustum with its side making an angle of 120 degree from the smaller horizontal base. The angle of 120 degree is chosen so that the solar array normal vector makes an angle of 0 degree with the Sun-Earth line during the majority of daylight time in the orbit. Additional solar cells have be installed for contingency. The attitude subsystem will slew the spacecraft in order to point to the sun to extract the maximum solar power before it reaches the target latitude range where the spacecraft will again be slewed in order to point the earth for its optical sensor situated on the nadir side to take the imagery.

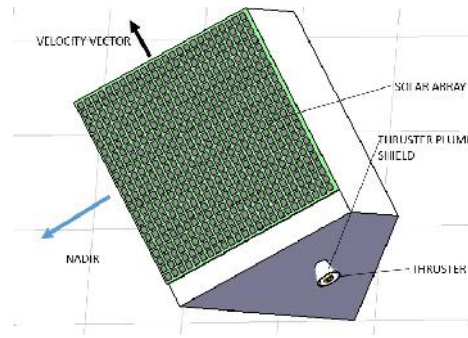


Fig 12: 3d Model of one spacecraft

C. Payload

The payload suitable for our constellation is the one showing at least 1.5 meter of resolution at 325 km altitude. Such demanding payload has been found in [8]. The payload is called HRI-2.5, it has a mass of 30 kg, a swath of 10 km at 325 km altitude. Data of power consumption have been implemented in the power subsystem. Using this payload we can play important role in the market of urban planning and Precision agriculture.

D. Mass Budget

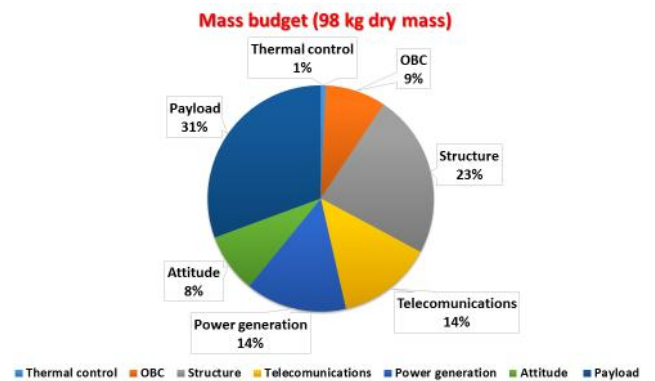


Fig 13: Mass Budget

We estimated the final dry mass of the satellite is 97 kg. Distribution of dry mass over the different subsystems is shown in the following pie chart. Final dry mass differs from the one estimated in the preliminary mass budget less than 20 percent.

VI. RISK ANALYSIS

A. Risk Matrix Analysis

There are many possible risks that can occur to our mission but, we considered some specific risks that our mission can face and rated the risks based on the below risk matrix. For example we are using electric propulsion to thrust our satellite at 325km altitude. Then we have found the mitigation measures which fall in the red zone.

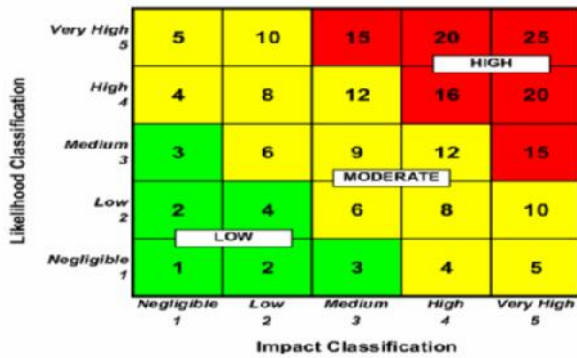


Fig 14 : RISK MATRIX

1. The need of continues orbit keeping

Likelihood = 5	Consequences = 5	Risk rating = 25
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2. The need of continues attitude control

Likelihood = 5	Consequences = 5	Risk rating = 25
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3. Electric Propulsion – over heating

Likelihood = 4	Consequences = 5	Risk rating = 20
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4. Deployment in incorrect altitudes

Likelihood = 3	Consequences = 4	Risk rating = 12
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5. Life time of thruster

Likelihood = 2	Consequences = 5	Risk rating = 10
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6. Failure of one of 5 satellites

Likelihood = 3	Consequences = 3	Risk rating = 9
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B. Mitigation measures

The need of continues orbit keeping, as we are flying in low altitudes there is continues drag on the satellites. The orbit decay will continue if thruster is off and the satellite will fall to the ground within 75 minutes of continues off of thruster. So the risk rate is very high. And the mitigation measure could be in order to overcome the drag we have to thrust the satellites with the specific altitude allowance

The need of continues attitude control. Because of the drag there is always a possibility of change in the attitude of the satellite. We need to continuously adjust the attitude of the satellite otherwise the camera may not look the target anymore and antenna may not target the ground station which results the mission failure so the risk rate is very high. Mitigation measure could be reaction and magneto torque redundancy.

Electric thruster generally delivers large heat that may cause wall erosion or may heat the other components of the satellite. If thruster fails because of overheating which results in the mission failure. Mitigation measure could be to manage the firing duration. In our mission, 11.33min of ON time and 218.43 min of OFF time to cool down the other components which results in small altitude allowance of 0.2km.

There is a chance of deploying the satellites in incorrect altitudes. If they deployed in higher altitudes, we just have to wait until they reach the desired altitudes that will be

automatically achieved because of the drag present on the satellites. But, if they get deployed in lower altitudes then we need additional fuel to put in the correct altitudes. In our mission for example if satellites are put in 275km i.e.,50km lesser than 325km we need additional 0.75kg of fuel for each satellite in turn 3.75kg of total extra mass to the mission.

Life time of thruster is limited in terms of amount of propellant consumed. Mitigation measure could be to find out the life time of thruster if in case it is shorter to the mission duration then carrying another thruster could be suggestable.

If anyone satellite fails in the constellation then we have to rephrase the other satellites which consumes the fuel intern shortens the life time of the mission. Mitigation could be to carry extra fuel.

CONCLUSION

The possibility to exploit very low altitude using electric thrusters with high specific impulse (1,500 s) opens many profitable ways for an Earth Observation mission. A satellite of 98 kg of dry mass is capable of very high ground resolution performance and downloading a huge imagery volume. The number of 5 satellites in a sun synchronous orbit at 325 km altitude grant high performance in terms of revisit frequency of a target in a latitude around 45 degree latitude. A mass and power budget have been conducted. Results have been shown in appropriate graphs .The relatively low volume and low mass contribute to reduce the launch cost. A risk analysis has been conducted. Few mitigation measures have been proposed for next phase of design.

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