Space Dynamics
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Analysis of Distributed Satellite Deployment Strategy

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Contents

1. Acknowledgement

It is a pleasure to express my deep thanks and gratitude to my guide Dr. D.K. Giri. His keen interest and dedication to the subject and overwhelming attitude towards his students is the main motivation for the completion of this project.

I would also like to extend my gratitude to the department for granting me such a wonderful opportunity and providing me with all the facilities needed in completion of the project.

2. Certificate

This is to certify that the work contained in the report entitled "Analysis of Distributed Satellite Deployment Strategy" by Hitesh Kumar Rawat (Roll no. 160298), has been carried out under my guidance.

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3. Introduction

The deployment of multiple small satellites has been an area of keen interest since the last coupe of years. It is due to multiple successful missions in the past and the important learning and realisations that came along with them. The experimentation with launch and deployment of a number of small satellites (*pallet* as coined by King and Beidleman) began primarily with the miniaturisation of electrical devices and the availability of relatively inexpensive commercial off-the-shelf components. This reduced the cost of hardware components. A number of value propositions of small satellite constellation are:

- Lower cost of development per satellite
- Ability to be launched in large numbers
- Perform many simultaneous and distributed observations. This naturally results in improved temporal resolution of data.

A way of access-to-orbit in the form of secondary payload has increased, but with limiting maneuver and problems such as lack of control on launch schedule and destination orbit prohibit the use of multiple secondary launch opportunities. This is further compounded by technical, mass and volume constraint on the propulsion system.

Some of strategies have been proposed which enable multiple small satellites comprising a constellation to be launched together both efficiently and in a cost effective way.

The ***deployment using natural Earth perturbations to indirectly achieve plane separations*** (also called **Differential Nodal Precession Strategy**) way is analysed in the upcoming sections in terms of its implementation and required mission parameters, challenges and an efficient way to overcome those challenges.

4. Differential Nodal Precession Strategy

One of the main challenges when working with multiple small satellites in a constellation is that how should an individual satellite be launched in its respective orbits. In most cases the constellation configuration is fixed. If many small satellites are launched from a single launch vehicle, the payloads must be dispersed into their respective mission orbits. However, to reduce system complexity, especially when working with secondary payload option, these platforms are typically limited in capability. On top of that, there could be technology, mass, volume constraints.

The ability of these satellites to manoeuvre from their point in insertion orbit into their required mission orbits is therefore constricted, especially if expensive out-of-plane manoeuvres in *Right Ascension of Ascending Node.* The use of nodal precession due to natural Earth perturbation is discussed in the following section. Using natural orbital perturbations to separate orbital planes in constellation deployment was patented by King and Beidleman in 1993. This method utilises time change of precession $\dot{\mathbf{\Omega}}$ due to non-spherical geometry of the Earth.

$$\dot{\mathbf{\Omega}}_{J_2} = \frac{-3}{2} \cdot \frac{R^2{}_E}{(a(1-e^2))} \cdot J_2 \cdot n \cdot cos(i)$$

where **a** is the semi-major axis, eccentricity **e**, and inclination **i**. This is an approximate formulation for nodal precession.

In general, the process begins from a common orbit after which one by one satellite is released by in-plane maneuver with different nodal precession. Here, every satellite has its own nodal precession value which depends on the sequence of launch.

The drifting required by this in-plane maneuver depends on the rate of separation and drift rate and is therefore affected by the drifting capabilities of a satellite.

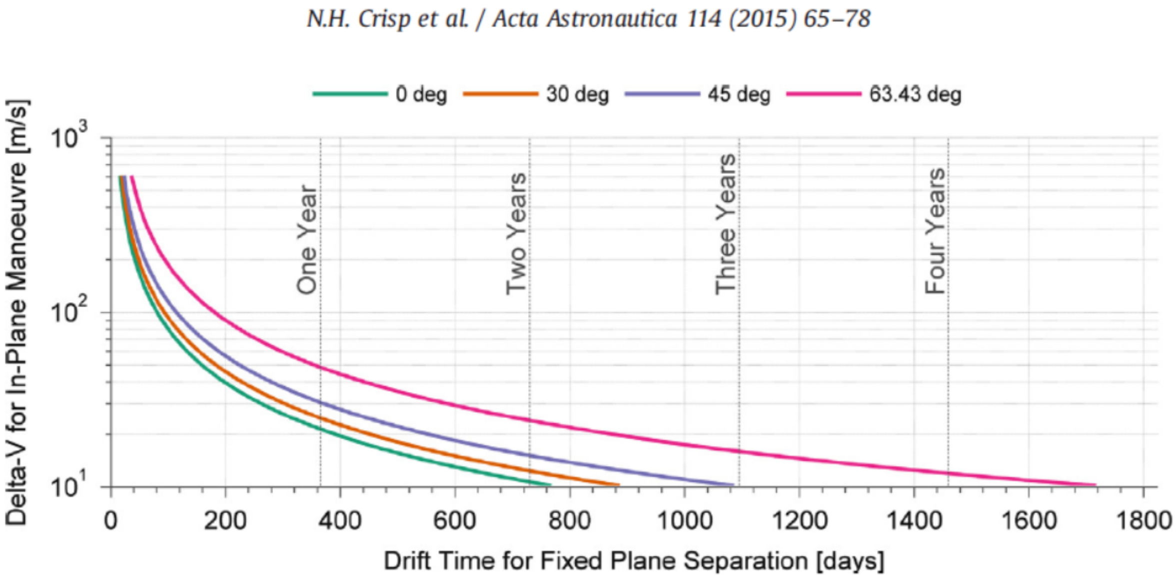

Figure 4.1: Δv with drift time for a fixed plane separation of 60° at varying inclination.

From the figure it can be concluded that for modest Δv, theoretical drift periods for the complete constellation deployment can vary from months to years. This is undesirable from mission and the satellite's lifetime point of view and even may bring up issues like orbital decay and components reliability.

5. Factors Affecting Deployment Strategy

There are many factors that affect the deployment strategy. It is very important to consider the effect of the following factors:

- **Atmospheric Drag.** Satellites in orbits of a different semi-major axis and eccentricity can experience significantly different levels of drag and will therefore decay at different rates. For low altitude constellations, there may be a risk of satellite de-orbit before the deployment has been completed. Propagation of the satellites throughout the deployment procedure can be used to determine the effect of drag on the constellation.
- **Physical Characteristics of Satellites.** The physical characteristics of the satellites and optional carrier vehicles can also influence the deployment process, affecting the magnitude of drag experienced and propulsive system requirements.
- **Time and energy**
- **Mission Requirements**

5.1 Satellite Propagation

This method uses Semi-Analytical Liu Theory (SALT) jointly developed by Liu and Alford. It makes use of differential nodal precession method to study the orbits of small satellites during their deployment lifetime. It accommodates for atmospheric drag and non-spherical shape of the Earth which leads to non-spherical geopotential. It was chosen to speed up execution and to analyse the results for many satellites in a constellation over a fully numerical solution.

5.2 Atmospheric Density Variation Due to Solar Flux

Solar flux incident on the upper atmosphere of the Earth, primarily Extreme Ultraviolet (EUV) radiation, has a heating effect which affects the local atmospheric density. The variation of incident solar flux affects the level of heating within the atmosphere, and therefore the drag that a satellite in orbit will experience.

The solar flux data at the level of $\mathbf{F_{10.7}}$ is considered from year 1980-2014. Curve fitting is applied on the flux data. The level of $\mathbf{F_{10.7}}$ has number of periodic trends and random variation as shown in the following distribution:

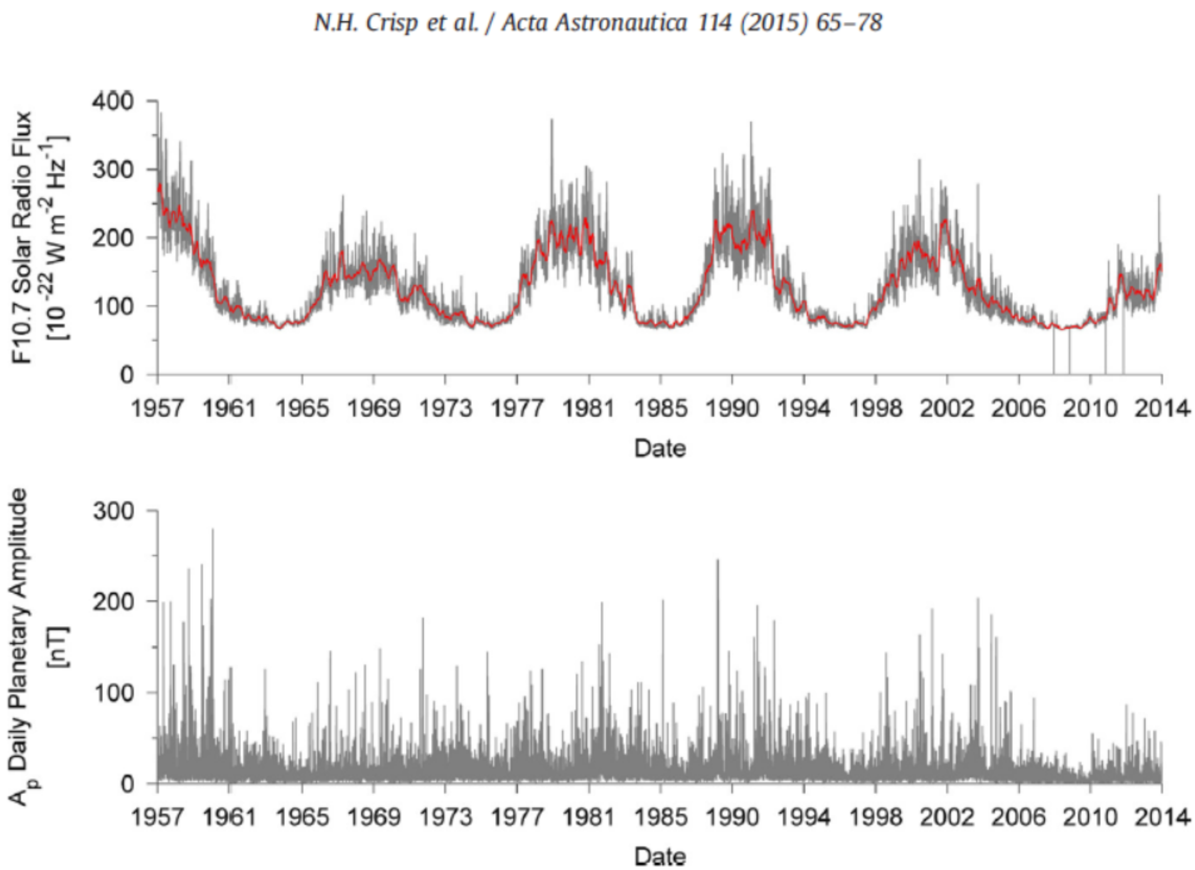

Figure 5.1: Daily and 81-day centred average $\mathbf{F_{10.7}}$ solar radio flux and daily planetary amplitude, $\mathbf{A_P}$

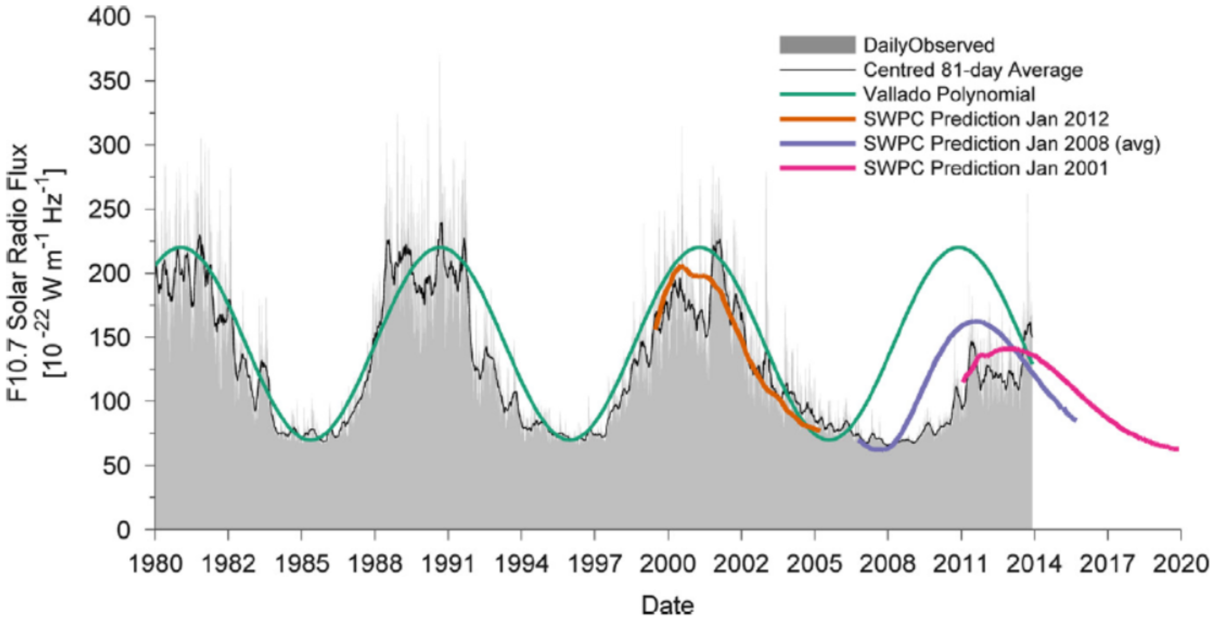

Figure 5.2: Comparison of measured and predicted F10.7 solar radio flux

Also, the prediction of geomagnetic index which is a measurement of interaction of energetically charged particles and atmospheric molecules, causes particle ionisation and a heating effect. This has an effect on atmospheric density. For the deployment of satellite constellations, the long term forecasting of solar flux and geomagnetic index is critical due to the length of analysis involved.

The aggregated set of solar flux and geomagnetic index data from NOAA/SWPC, available from the CelesTrak website, was chosen due to the completeness and availability of up-to-date measured and forecast data.

5.3 Affect of Physical Characteristics on Drag

The measure of the amount of drag experienced by the body (in this case the individual satellites launch vehicle) based on its physical characteristics is termed as Ballistic Coefficient. Mathematically, it can be easily computed using:

$$BC = \frac{C_D \cdot A}{m}$$

where **BC** is the Ballistic Coefficient, $\mathbf{C_D}$ is the total drag coefficient which can be averaged out with altitude variation to be **2.2**, cross-sectional area **A** taken normal to the velocity vector of the satellite, mass of the satellite that can be dynamically computed from propellant usage **m**.

Note that for a tumbling spacecraft with constantly varying attitude, an averaged cross-sectional area can be calculated either by integrating the different cross-sectional areas over the range of possible attitudes or using a simplified flat-plate model.

6. Constellation Deployment Procedure

Considering the predictions on the above described parameters, the deployment of constellation using the method of differential nodal precession from a single launch insertion point can be analysed. This analysis can be represented by the following procedure:

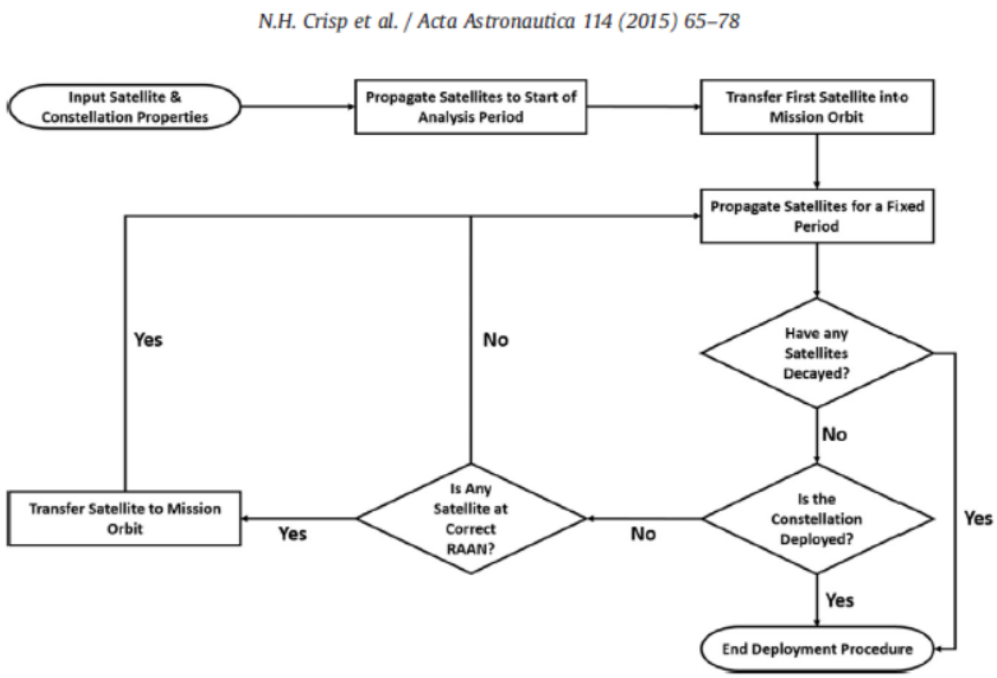

Figure 6.1: Procedure for deployment analysis of a constellation with one satellite per plane

The semi-analytical method enables the observation of satellite orbital decay through the use of predicted Earth environment data and a complex atmospheric density model. For constellations inserted or operated in low altitude orbits, the decay of satellites prior to the complete deployment of the constellation would constitute an bad design. Furthermore, any change in orbit shape and size alters the rate of nodal precession and can therefore affect the rate of separation of planes in the constellation. For constellation configurations in which there is more than one satellite in each plane additional phasing manoeuvres are required to achieve the configuration of payloads. In these circumstances, carrier vehicles can be used in order to perform the more intensive in-plane manoeuvres and then dispense the payloads into each plane.

The method of analysis for deployment using the method of indirect plane changes can be validated against the actual deployment of the **FORMOSAT-3/COSMIC mission** in the upcoming section. Moreover, the aim of the report was to explore and validate the deployment strategy using the method of indirect plane changes using the data available for one of INSAT, IRS, IRNSS spacecraft missions.

7. Analysis of FORMOSAT-3/COSMIC Mission

The FORMOSAT-3/COSMIC constellation of six microsatellites was launched in April 2006 on an Orbital Sciences Minotaur vehicle. To achieve the mission configuration, the six payloads were deployed into six orbital planes equispaced about 180° in RAAN using the method of natural nodal precession over a period of 20 months.

Following mission parameters were used for the mission:

S.no.	**Property**	**Value**
1	No. of satellites	6
2	No. of planes	6
3	Satellite dry mass	54 KG
4	Satellite fuel mass	6.65 KG
5	Thrust, BoL to EoL	1.1N to 0.2N
6	Specific impulse	217s to 197s
7	Satellite area	0.5963 m^2
8	Satellite C_D	2.2

Insertion Orbit

S.no.	**Property**	**Value**
1	Semi-major axis	6893 KM
2	Eccentricity	0.00323
3	Inclination	71.992°
4	RAAN	301.158°

Mission Orbit

S.no.	**Property**	**Value**
1	Semi-major axis	7178 KM
2	Eccentricity	< 0.014
3	Inclination	71.992°
4	RAAN	Ω_1 - (30°, 60°, 90°, 120°, 150°)

7.1 Results

Comparison of actual and analytically predicted values of drift required for each satellite is as follows:

S.no.	**Satellite**	**Actual Δv (m/s)**	**Calculated Δv (m/s)**
1	FORMOSAT-3C	153.1	152.2
2	FORMOSAT-3F	154.0	152.6
3	FORMOSAT-3A	153.0	153.0
4	FORMOSAT-3E	154.0	153.7
5	FORMOSAT-3D	107.9	154.1
6	FORMOSAT-3B	129.8	154.1

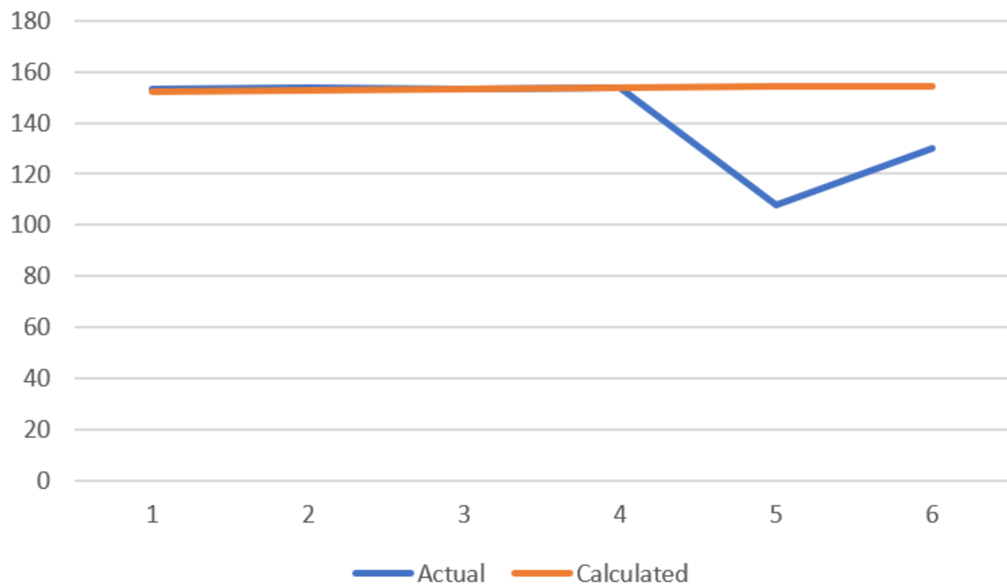

Figure 7.1: Comparison of Actual and Calculated drift speeds

From the results it can be concluded that the difference between actual and calculated Δv is less than 1% except for the propulsive failure of FORMOSAT-3D and anomaly in FORMOSAT-3B.

8. Future Work

To more thoroughly understand the feasibility and potential benefits of constellation deployment using either of INSAT, IRS, IRNSS, INS series missions, an increased understanding of the trajectory design and subsystem requirements and design is required.

Currently, above described method of deployment are not considered by existing design processes.

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