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STATION-KEEPING STRATEGY FOR A SOLAR SAIL SATELLITE AT LOW EARTH ELLIPTICAL ORBIT

H. C. POLAT and O. TEKINALP

Middle East Technical University, Ankara, Turkey
E-mail: halis.polat@metu.edu.tr
E-mail: tekinalp@metu.edu.tr

Abstract: An elliptical Low Earth Orbit for Earth observation mission is sized and shaped such that aerodynamic drag effects are minimized with a perigee altitude of 400 km and an apogee altitude of 1400 km. The lower parts of the orbit (below 800 km) are allocated for Earth observation whereas remaining parts of the orbit (above 800 km) is used for solar sailing for reshaping, resizing, or reorienting the orbit. The mission orbit is then optimized for a given region of interest (latitude and longitude) in terms of classical orbital elements (inclination, argument of perigee and right ascension of ascending node). This mission orbit for the given region of interest has a lifetime which can be given in advance by the customer/end user or caused by environmental disturbances (aerodynamic drag and/or asphericity effects of the Earth's gravitational attraction). To increase the mission lifetime, a station-keeping strategy is proposed in this study. The adopted dynamics of the classical orbital elements of the proposed strategy is based on the Gauss Variational Equations. Different cases in terms of station-keeping of perigee altitude, semi-major axis, argument of perigee, right ascension of the ascending node or combination of those, are investigated. The coupling effects of variations in the classical orbital elements are taken into considerations. Detailed mathematical derivation of the proposed strategy is provided along with the simulation results.

Keywords: solar sail satellite, station-keeping, gauss variational equations, elliptical orbit, low earth orbit.

1. INTRODUCTION OF THE SUBJECT

Satellites are used for various missions with capable sensors to provide information. The information may be in different forms; communication signal, position-time tag for navigation and images of the Earth at several wavelengths. The demand for satellite images has been present since the start of space age. The information can be used for various applications such as disaster monitoring or environmental surveillance for forestry and agriculture, to name a few. To achieve desired performance from observation satellites, mission designers try to optimize the resolution and coverage (or revisit time in other perspective). Low Earth Orbit (LEO) provides the utmost performance for such considerations. Even though, mission designers generally choose sun-synchronous or polar orbits at LEO for global coverage resulting in constant revisit/coverage performance, the observation performance in terms of revisit frequency can be increased by using inclined prograde orbits at LEO. However, the orbit is then successful only for the given region of interest not for the new sites. Moreover, changing the orbital parameters is too expensive in space, even with the electric propulsion. On the other hand, Solar Sail Satellites can be utilized for their propellant-free manoeuvre capability due to the Solar Radiation Pressure. The concept of Rapid Response Solar Sail (R2S2) is proposed to combine the advantages of LEO and Solar Sail Satellites.

1.1. R2S2 CONCEPT

Propulsion through solar sails is expected to be the future of the spaceflight within the solar system mainly for interplanetary travels or escape trajectories (McInnes 1989, Wie 2008, Vulpetti et al. 2008). Moreover, large sail areas with a low total satellite mass (low ballistic coefficient) result in quick decay of the orbit at lower altitudes. To overcome this difficulty while utilizing the advantages of employing LEO an elliptical orbit is proposed for the R2S2 concept (Figure 1).

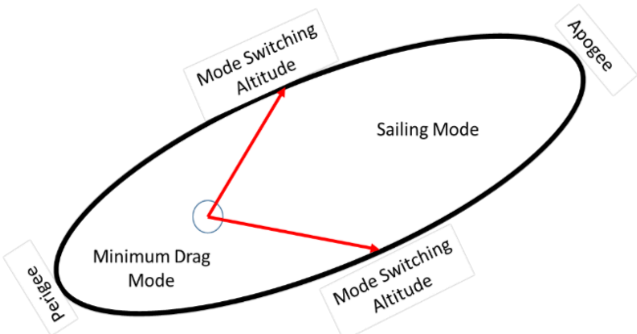

Figure 1: Elliptical Mission Orbit

The lower altitude part of the orbit is where the Earth observation is carried out while orienting the sail such that aerodynamic drag is minimized. The higher altitude part of the orbit is where the advantages of solar sailing are utilized. The sailing manoeuvres, when resizing, reshaping, or reorienting is necessary, are carried out in this part of the orbit while orienting the sail such that required control force is provided with limitations coming from the relative Sun’s position with respect to the sail and shadowing of the Earth.

This mission orbit has a lifetime due to two reasons: Mission-specific and Disturbance-specific. Mission-specific lifetime comes from the mission designer depending on the observation of the region or post-disaster campaign duration. Disturbance-specific lifetime comes from the environmental disturbances exerted on the satellite such as aerodynamic drag and asphericity effects of the Earth. The mission phase ends at the end of mission-specific or disturbance-specific lifetime (whichever comes first). Then, transfer phase is started to raise the orbit above mode switching altitude. This means that solar sailing can be done at all locations along the orbit. Transfer phase ends when suitable standby orbit is achieved. At standby orbit, the solar sailing manoeuvres are used for resizing, reshaping, and reorienting the orbit to maximize the revisit performance for the new region of interest. This cycle is maintained as shown in Figure 2 and the feasibility study results show that successive phase transitions (400 km of perigee altitude and 1400 km of apogee altitude) are achieved with characteristic acceleration ranging from 0.1 to 1.5 mm/s^2 (Polat & Tekinalp 2019).

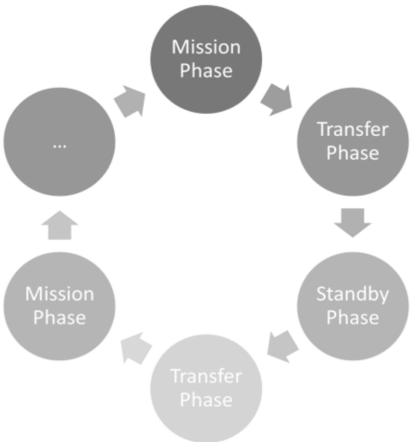

Figure 2: Phase Transitions

1.2. MODE HIERARCHY

The R2S2 concept with continuous phase transitions and solar sailing manoeuvres uses 4-tier mode hierarchy (Figure 3) for mission control. The first mode (Mission Mode) is for defining the orbit type, then Task Mode is defined with respect to the aim (station-keeping, minimum energy loss, raising or decreasing the orbit). After that, the Orientation Mode is determined according to the altitude of the satellite with respect to the mode switching altitude. This allows to give green flag to make sail rotations for desired attitude with negligible aerodynamic drag effects. The last tier is used for only Station-keeping Task Mode. The definitions are given in Table 1.

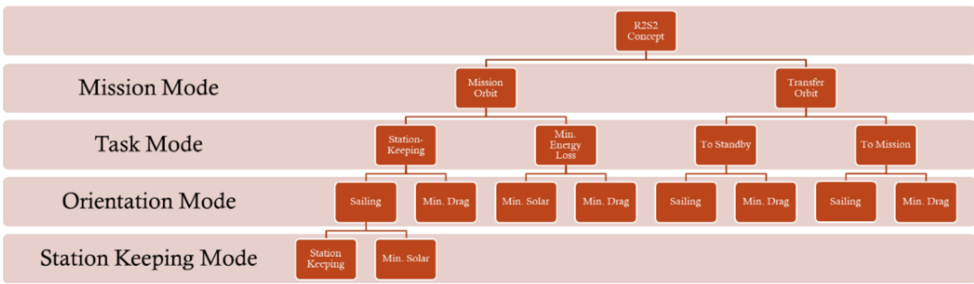

Figure 3: Mode Hierarchy

Table 1: Mode Definitions

Mode Name	Definition
Mission Orbit	This mode is the indication of the orbit type. The selected elliptical orbit is employed for the Earth observation mission and pre-determined orbital parameters are maintained by station-keeping.
Transfer Orbit	This mode is the indication of the orbital transfers between mission and standby orbits.
Station-Keeping	This mode is active when mission orbit is employed and triggering conditions for selected orbital parameters are activated. Necessary orbital manoeuvres are performed to maintain the mission-specific orbital parameters.
Minimum Energy Loss	This mode is the indication of that there is no manoeuvring needs, therefore minimum energy loss is aimed. This aim is accomplished by the Minimum Drag and Minimum Solar modes.
To Standby	This mode, belonging the Transfer Orbit, is the indication of necessary manoeuvres to be performed to reach the standby orbit by raising the perigee altitude above the mode switching altitude.
To Mission	This mode, belonging the Transfer Orbit, is the indication of necessary manoeuvres to be performed to reach the mission orbit by altering the perigee altitude, inclination, right ascension of ascending node and argument of perigee.
Sailing	This mode is the indication of that the altitude of the satellite is above the mode switching altitude and the satellite is free to orient its sail to generate required thrust for manoeuvring needs.
Minimum Drag	This mode is the indication of that the altitude of the satellite is below the mode switching altitude and sail orientation should be kept tangent to orbit to minimize the aerodynamic drag.
Minimum Solar	This mode is the indication of that the altitude of the satellite is above the mode switching altitude and there are no manoeuvring needs. Therefore, the sail orientation should be kept tangent to the ecliptic plane to minimize the Solar Radiation Pressure effects on the orbit.

2. STATION-KEEPING STRATEGY OF THE MISSION ORBIT

The station-keeping strategy of the mission orbit of the R2S2 concept is explained below in detail. First, orbital elements effecting the mission lifetime is explained to give the rationale behind the need for station-keeping. Then, principles adopted in this study are provided with a list of all cases considered. After that, mathematical formulation of the strategy is given with case-based control logic. Finally, simulation results are presented with discussion.

2.1. ORBITAL ELEMENTS EFFECTING MISSION LIFETIME

The mission lifetime, as explained above, depends on two reasons. Mission-specific lifetime will not be discussed further here since it is coming from the end user. On the other hand, disturbance-specific lifetime is the concentration point of this study.

Four orbital parameters (out of six) are taken into consideration for station keeping: semi-major axis, eccentricity, argument of perigee and right ascension of ascending node (RAAN). Since the ellipticity of the mission orbit is vital to the concept, semi-major axis and eccentricity are tracked in terms of perigee and apogee altitudes. Moreover, perigee and apogee altitudes are crucial for the safety of the concept because the failure of keeping these altitudes will result in decay of the orbit exponentially. Without station-keeping, the lifetime in the mission orbit may vary between 20 days to 3 months depending on the Sun-Earth-Satellite orientation. As it will be seen in the final section, proper station-keeping of the perigee and apogee altitudes may raise the mission life to 6-12 months.

Argument of perigee and RAAN values are important for mission success (revisit frequency and access number within the finite mission lifetime). The details of these effects may be found in another study (Polat & Tekinalp 2020). In summary, there are some intervals of values for argument of perigee effecting the mission performance relevant to the altitude criteria for an acceptable access of the satellite to the observation site. Similar intervals are also present for RAAN values for given day of the year effecting the mission performance relevant to daylight condition. To better understand, three conditions of the R2S2 concept conditions for an acceptable access are given below:

- The site is illuminated by the Sun (daylight condition is needed for electro-optical satellite to view the site),
- The altitude of the satellite is below 800 km (R2S2 Concept proposes the use this altitude range, 400-800 km for mission mode),
- Small Line-of-Sight angle between site and satellite is achieved (i.e., 20 degrees for half sensor angle for pointing off-nadir).

Changes due to J2 effects (apsidal and nodal precessions) are to be reduced by solar sailing to extend the mission lifetime with high performance observation metrics. The intervals shall be optimized for the mission orbit at the beginning with the trends of J2 perturbation. Although it is impossible to compensate for the asphericity effects of the Earth continuously, this tendency may be reduced through solar sailing.

2.2. PRINCIPLES

Two-tier control is applied to perigee and apogee altitudes to avoid chattering. 350 km is defined as the lower limit to trigger the perigee altitude station-keeping mode and is active until 400 km is achieved. Similar approach is applied for apogee altitude with 1350 km as lower limit resulting in an active station-keeping until reaching 1400 km of apogee altitude.

Depending on the initial orbital parameters and analysis on the observation performance of the mission orbit with requested mission lifetime, station keeping may be triggered for argument of perigee and/or RAAN as well. However, this should be done at the beginning of the observation task and should be continuously carried out.

When possible, in-plane and out-of-plane force components shall be used separately for controlling the orbit size (in-plane for perigee and/or apogee altitudes) and orientation (out-of-plane for argument of perigee and/or RAAN) control.

For cases with multiple orbital parameters, the priority order is as follows: perigee altitude, apogee altitude, argument of perigee and RAAN. The rationale behind for the priority is simple: First mission safety, then mission performance.

2.3. CASES FOR STATION-KEEPING

With four parameters to be considered for station-keeping a list of cases is generated and given in Table 2. All 16 combinations are considered for station-keeping strategy including Case Zero where no station-keeping condition is triggered. Each case is indicating the need for a time rate of change of a specific orbital parameter or combination of those depending on the triggering conditions. The need to generate the time rate of change of these orbital parameters are then met by solar sailing manoeuvres.

Table 2: Case List

Case Code	Condition	Case Code	Condition
Case A	$\dot{r}_p$	**Case I**	$\dot{r}_p+\dot{r}_a+\dot{\Omega}+\dot{\omega}$
Case B	$\dot{r}_a$	**Case J**	$\dot{r}_p+\dot{\Omega}$
Case C	$\dot{\Omega}$	**Case K**	$\dot{r}_p+\dot{\omega}$
Case D	$\dot{\omega}$	**Case L**	$\dot{r}_a+\dot{\Omega}$
Case E	$\dot{r}_p+\dot{r}_a$	**Case M**	$\dot{r}_a+\dot{\omega}$
Case F	$\dot{r}_p+\dot{r}_a+\dot{\Omega}$	**Case N**	$\dot{r}_p+\dot{\Omega}+\dot{\omega}$
Case G	$\dot{r}_p+\dot{r}_a+\dot{\omega}$	**Case O**	$\dot{r}_a+\dot{\Omega}+\dot{\omega}$
Case H	$\dot{\Omega}+\dot{\omega}$	**Case Zero**	No Condition

Due to the principles of the station-keeping strategy of the R2S2 concept, triggering condition for the argument of perigee and/or RAAN will be initially conditioned and continuous. For that reason, the need for the time rate of changes of argument of perigee and/or RAAN will be always the same (they exist or not). Only perigee and apogee altitude needs may change. As triggering conditions for perigee and apogee altitudes occur, case codes change. Similarly, as perigee and apogee altitudes are increased to their mission parameters by station-keeping manoeuvres, case codes change again. This behaviour results in four different cycles for case transitions (Figure 4) and the subplots are ordered as:

- no argument of perigee and RAAN station-keeping,
- only argument of perigee station-keeping,
- only RAAN station-keeping,
- both argument of perigee and RAAN station-keeping.

During the mission lifetime, station-keeping cases change to a different case within the cycles shown in Figure 4. No interchange for case codes is possible between case cycles.

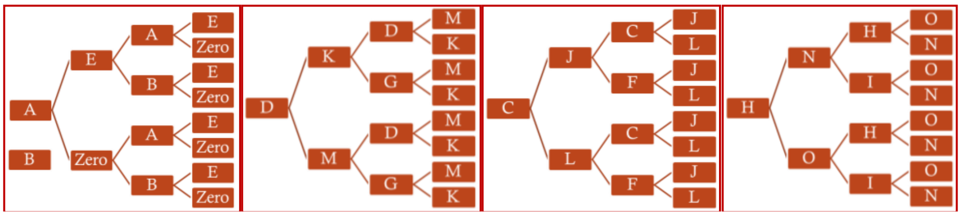

Figure 4: Case Cycles

3. VARIATION DYNAMICS OF THE ORBITAL ELEMENTS

In this section, the proposed approach to the control of shape, size and orientation of the orbit is given.

3.1. GAUSSIAN VARIATIONAL EQUATIONS

The control logic is implemented by utilizing the Gaussian Variational Equations (Battin 1987). GVEs, listed in Eq. (1), provides the dynamics for classical orbital elements under the influence of two-body Earth's gravitational attraction. All the other disturbances/control forces are treated as control input. Nomenclature for Eq. (1) is given in Table 3.

$$
\begin{aligned}
\dot{a} &= \frac{2a^2}{h}\left[eR\sin\theta + \frac{pT}{r}\right] \\
\dot{e} &= \frac{1}{h}\left(pR\sin\theta + \left[\left(p+r\right)\cos\theta + re\right]T\right) \\
\dot{i} &= \frac{r\cos(\omega+\theta)}{h}N \\
\dot{\Omega} &= \frac{r\sin(\omega+\theta)}{h\sin i}N \\
\dot{\omega} &= -\frac{r\sin(\omega+\theta)}{h\tan i}N + \frac{1}{eh}\left[-pR\cos\theta + (p+r)T\sin\theta\right] \\
\dot{\theta} &= \frac{h}{r^2} + \frac{1}{eh}\left[pR\cos\theta - (p+r)T\sin\theta\right]
\end{aligned}
\qquad (1)
$$

The dynamics of the classical orbital elements given in Eq. (1) depends only on their values with no control input. For variables h, p and r, Eq. (2) is given for demonstrating the strict dependence on classical orbital elements.

Table 3: Nomenclature

a	Semi-major axis	h	Specific Angular Momentum
e	Eccentricity	p	Semi-Latus Rectum
i	Inclination	r	Magnitude of Position Vector
Ω	Right Ascension of Ascending Node	R	Force per Unit Mass-Radial Direction
ω	Argument of Perigee	T	Force per Unit Mass-Transverse Direction
θ	True Anomaly	N	Force per Unit Mass-Normal Direction

$$
\begin{aligned}
r &= \frac{a\left(1-e^{2}\right)}{1+e\cos\theta} \\
p &= a\left(1-e^{2}\right) \\
h &= \sqrt{\mu a\left(1-e^{2}\right)}
\end{aligned}
\tag{2}
$$

When we evaluate the principles of station-keeping of the R2S2 concept, perigee and apogee altitude dynamics are important. Therefore, Eq. (3) is given with differentiation of the perigee radius which is equivalent of the differentiation of perigee altitude.

$$
\begin{aligned}
r_p &= a\left(1-e\right) \\
\dot{r}_p &= \dot{a}\left(1-e\right) - a\dot{e}
\end{aligned}
\tag{3}
$$

Then, Eq. (1) is inserted into Eq. (3) and Eq. (4) is obtained. Same is applied to apogee radius as can be seen in Eq. (5).

$$
\begin{aligned}
\dot{r}_p = \frac{2a^{2}}{h}&\left[e\sin\theta\left(1-e\right) - \frac{\left(1-e^{2}\right)\sin\theta}{2}\right]R + \ldots \\
&\left[\frac{2a^{2}p\left(1-e\right)}{hr} - \frac{a}{h}\left[\left(p+r\right)\cos\theta + re\right]\right]T
\end{aligned}
\tag{4}
$$

$$
\begin{aligned}
r_a &= a\left(1+e\right) \\
\dot{r}_a &= \dot{a}\left(1+e\right) + a\dot{e}
\end{aligned}
\tag{5}
$$

After setting the dynamics, approximations to these formulations are calculated with respect to the mission orbit parameters (400 km of perigee altitude and 1400 km of apogee altitude). As can be observed from Eq. (6), approximated dynamics of the classical orbital elements now depend on angles (argument of perigee, true anomaly, inclination, and alpha angle) and forces. In Eq. (6), in-plane forces, *T* and *R* are replaced by force *F* and angle α as can be seen in Figure 5.

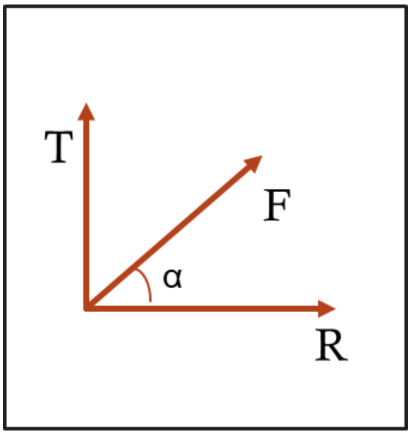

Figure 5: In-Plane Force Component Diagram

$$
\begin{aligned}
\dot{a} &= 135.44F\cos\alpha\sin\theta - 1971.54F\sin\alpha\left(0.068\cos\theta - 1\right)\left[m/s\right] \\
\dot{e} &= 1.35\times10^{-4}F\cos\alpha\sin\theta\ldots \\
&+1.35\times10^{-4}\left(\frac{2\cos\theta + 0.068\cos^2\theta + 0.068}{1+0.068\cos\theta}\right)F\sin\alpha\left[1/s\right] \\
\dot{i} &= \frac{1.35\times10^{-4}\cos\left(\omega+\theta\right)}{1+0.068\cos\theta}N\left[rad/s\right] \\
\dot{\Omega} &= \frac{1.35\times10^{-4}\sin\left(\omega+\theta\right)}{\sin i\left(1+0.068\cos\theta\right)}N\left[rad/s\right] \\
\dot{\omega} &= -\frac{1.35\times10^{-4}\sin\left(\omega+\theta\right)}{\tan i\left(1+0.068\cos\theta\right)}N - 1.98\times10^{-3}F\cos\theta\cos\alpha\ldots \\
&+1.35\times10^{-4}\left(\frac{29.4\sin\theta + 0.5\sin2\theta}{1+0.068\cos\theta}\right)F\sin\alpha\left[rad/s\right]
\end{aligned}
\tag{6}
$$

Then rates of changes of orbital elements are plotted with respect to true anomaly, argument of perigee, and in plane force angle α as shown in Figure 6 and 7. In those graphs, all *F* (per unit mass inputs) are treated as unity. In these plots, the current inclination, which appears in argument of perigee and RAAN rate equations, is taken as 55^0. Because the R2S2 concept always uses prograde orbit which results in the situation that inclination does not flip the sign of the formula output. The magnitude changes are negligible in the scope of this study. However, for information, as inclination decreases the changes in argument of perigee and RAAN increase.

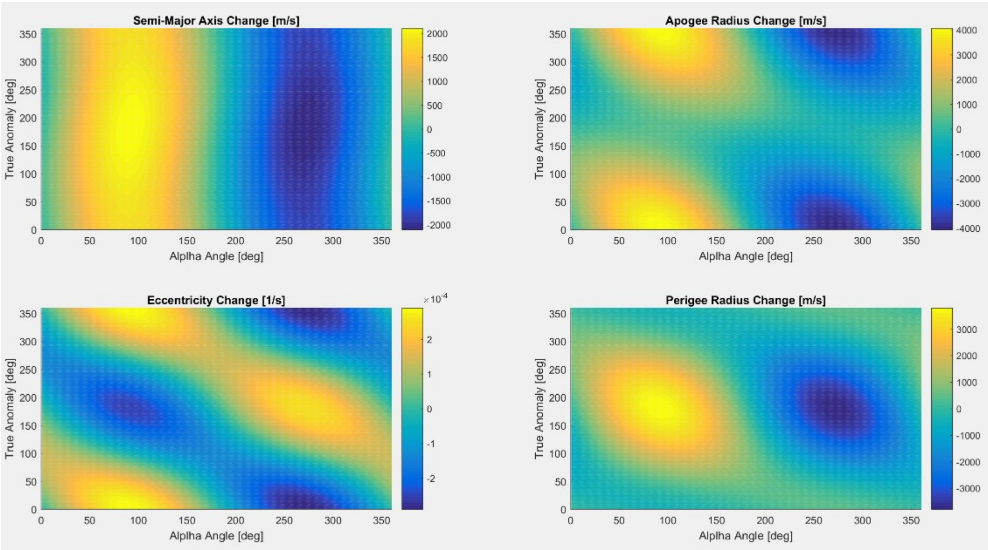

Figure 6: Orbit Resizing Graphs

Since, the lowest characteristic acceleration allowed in the R2S2 concept is in the order of 0.1 mm/s^2, these graphs can be interpreted as the change times 10^{-4}. This implies that approximated changes for orientation angles (Figure 7) are minor relative to the size parameters (Figure 6).

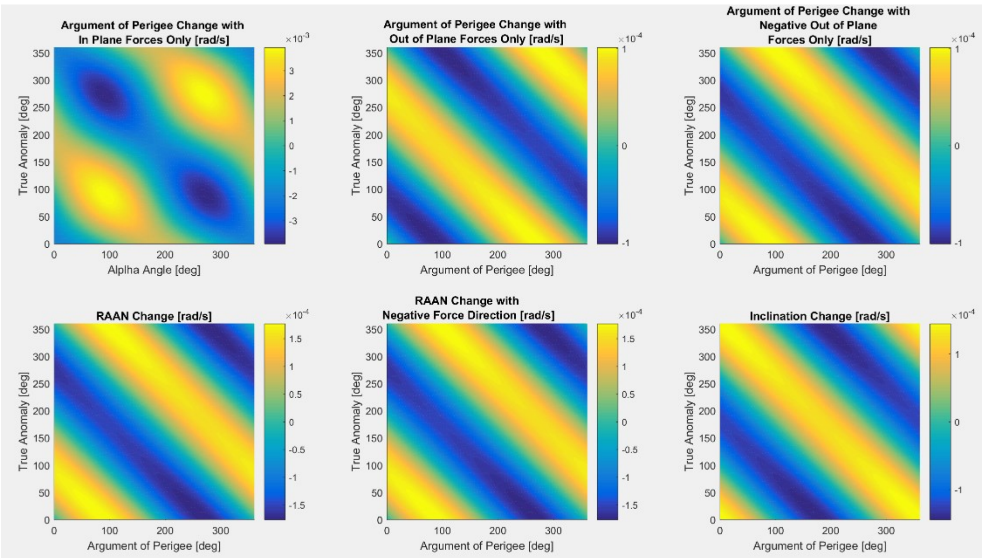

Figure 7: Orbit Reorienting Graphs

This results in only 1% improvement over the mission lifetime due to the argument of perigee or RAAN changes by the asphericity of the Earth. As mentioned earlier, the tendency of shifts of argument of perigee and RAAN by the asphericity effects of the Earth cannot be balanced by solar sailing manoeuvres, only reducing the shift rate is possible. This reduction rate over shifts of the argument of perigee and RAAN at the selected characteristic acceleration of the R2S2 concept is at 1%. But this improvement can be increased if a larger sail area is used (resulting an increase on the characteristic acceleration value to the orders of 1 mm/s^2 or more and eventually an increase on the reduction rate over the shifts).

3.2. CASE-BASED CONTROL LOGIC

The approximated dynamics output of the classical orbital elements are used to determine the control force direction. This process is carried out for each case with their own properties and requirements.

3.2.1. *Case A*

The perigee altitude dynamics depending on the true anomaly and alpha angle is manipulated excluding the true anomaly region from -83^0 to $+83^0$ (due to the no Sailing Zone from the concept itself), values above -10^{-4} s^{-1} of eccentricity change (due to the aim of decoupling the effects of perigee and apogee altitudes) and values below 1000 m/s of perigee altitude change are discarded. The remainder region for manoeuvres is shown in Figure 8. The optimal alpha angle for given true anomaly is then used for controlling the sail orientation when *Case A* is triggered.

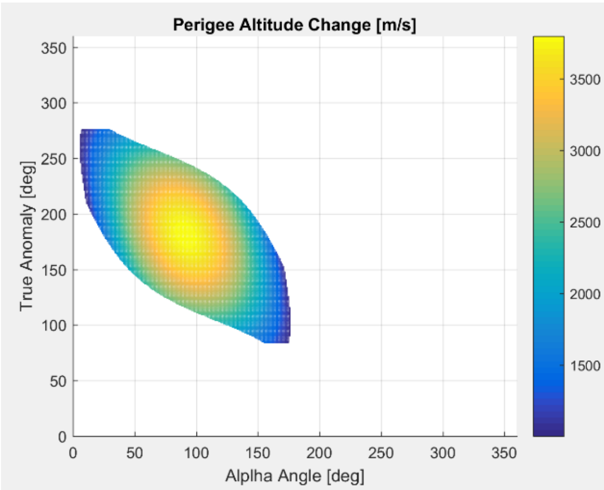

Figure 8: Perigee Altitude Change

3.2.2. *Case B*

Similarly, apogee altitude dynamics are given under the consideration of eccentricity and no Sailing Zone true anomaly region. The resultant graph is given in Figure 9. As may be observed from the graph, the region for manoeuvres is limited. This is since portions of the orbit around the perigee altitude, which are the most effective region for altering the apogee altitude, are treated as no sailing zone. When these portions of the orbit are discarded, there exist very limited opportunities to carry out necessary manoeuvres for increasing the apogee altitude. These opportunities are tried to be increased by lowering the limit of 1000 m/s change to 100 m/s.

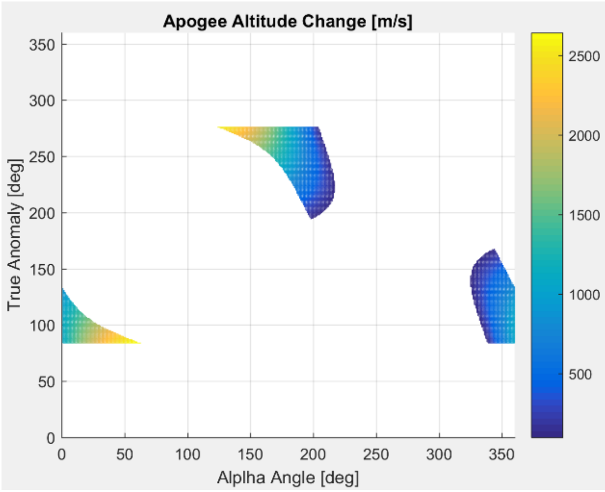

Figure 9: Apogee Altitude Change

3.2.3. *Case C* and *Case D*

The dynamics for RAAN and argument of perigee are given in Figure 10. In here, only out-of-plane force components are used for argument of perigee due to the reasons given above.

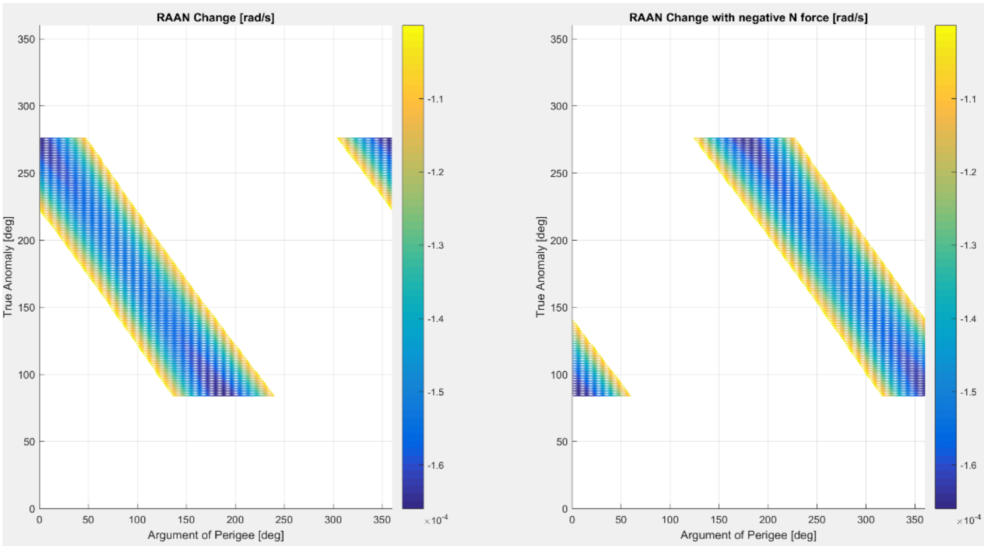

Figure 10: (a) RAAN and Argument of Perigee Changes

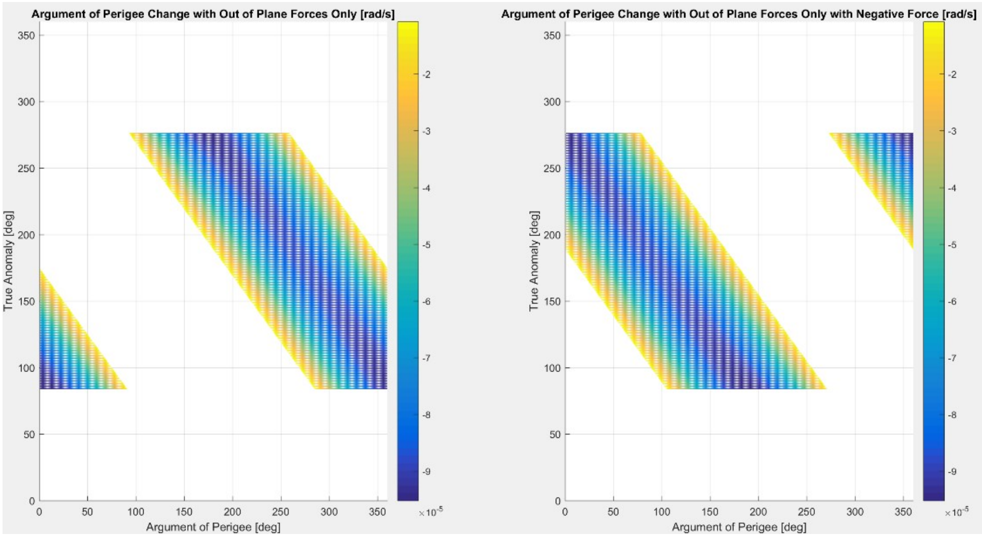

Figure 10: (b) RAAN and Argument of Perigee Changes

3.2.4. *Case E*

This case requests the station-keeping of perigee and apogee altitudes together. Since semi-major axis is the half of the sum of the apogee and perigee radii, the case request is well-met with station-keeping of semi-major axis. The dynamics of the semi-major axis is given in Figure 11. The alpha angle for optimal change for semi-major axis follows the velocity vector as expected.

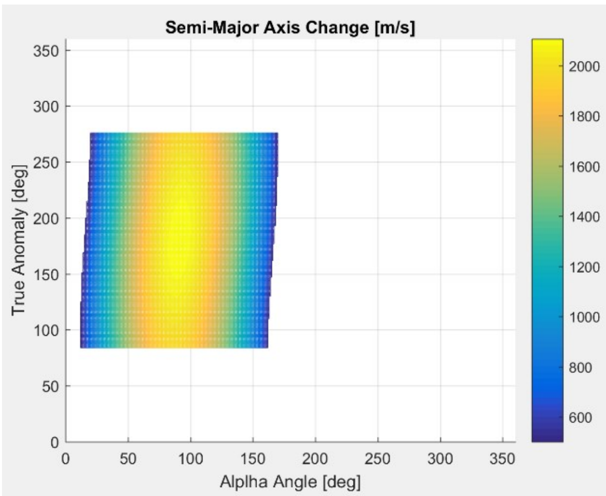

Figure 11: Semi-major Axis Change

3.2.5. *Case F*

Since we have covered the primary cases for individual orbital elements (*Case A-E*), we will see some combination of these strategies. *Case F* is the first example of this. The control force direction is given in Figure 12. *F* force is determined from *Case E* and *N* force is determined from *Case C*. Then, these directions are combined with weighting parameter which gives more force component to the in-plane forces due to the urgent need for station-keeping of the orbit size.

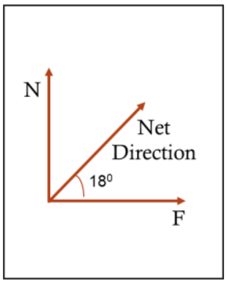

Figure 12: Weighting between In-Plane and Out-of-Plane Forces

3.2.6. *Case G*

In this case, out-of-plane forces are not considered for the argument of perigee, since the portion of the semi-major axis graph affecting the argument of perigee is an order of magnitude higher than the out-of-plane force effects. Therefore, that portion in semi-major axis is discarded and in-plane forces are used for argument of perigee. Final status is given in Figure 13.

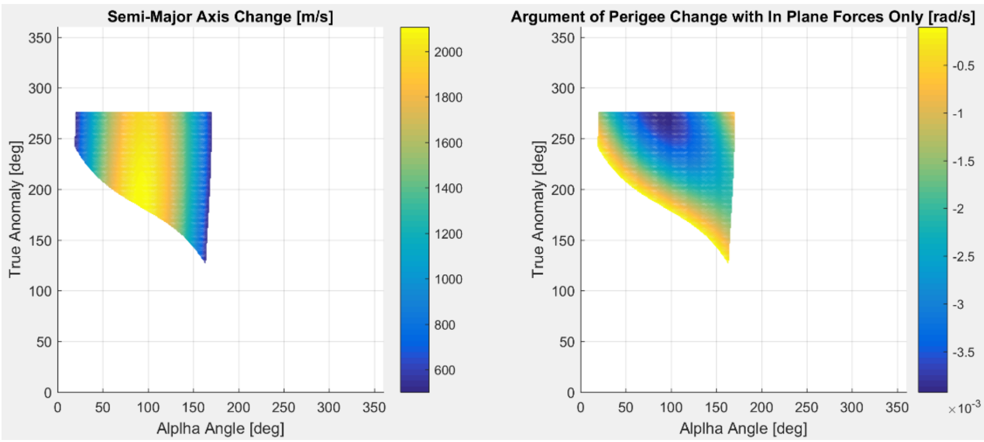

Figure 13: Co-Illustration of Semi-major Axis and Argument of Perigee Changes

3.2.7. *Case H*

When we look at Eq. (1) and Figure 7, the dynamics of argument of perigee and RAAN show that they have the same periodic dynamics with varying magnitude and there is a 90-degree phase shift. This means a particular change for the principles of the R2S2 concept. Hence, RAAN and argument of perigee can't be controlled simultaneously with only out-of-plane forces. Therefore, in-plane forces are considered for argument of perigee. While applying in-plane forces for argument of perigee, regions where affecting the perigee and apogee altitudes are discarded to avoid unnecessary altering of the altitude of the orbit. The resultant graphs for the determination of the control input are given in Figure 14.

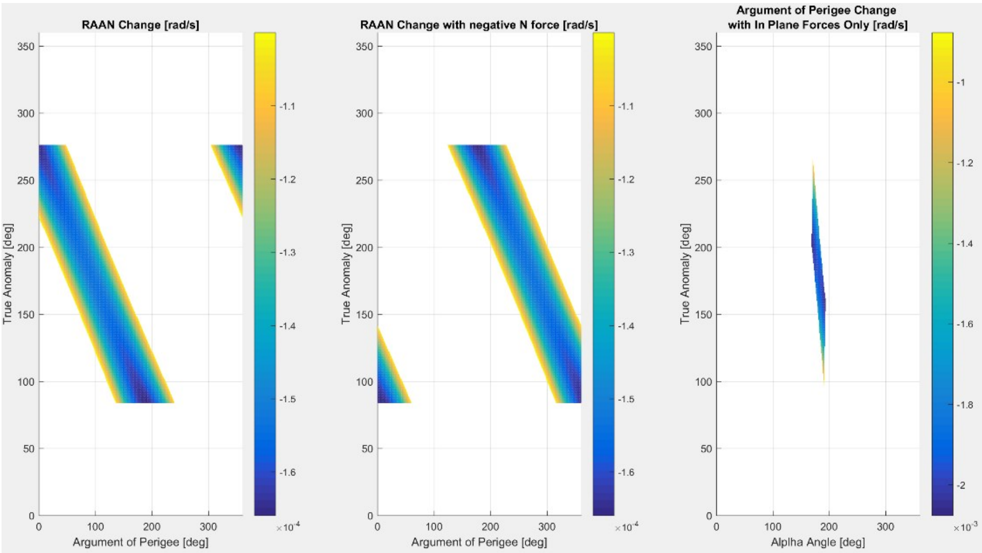

Figure 14: RAAN as a function of Argument of Perigee and Argument of Perigee with respect to α

3.2.8. *Case I*

Similarly, the control force direction is given in Figure 12. *F* force is determined from *Case G* and *N* force is determined from *Case C*.

3.2.9. *Case J*

Similarly, the control force direction is given in Figure 12. *F* force is determined from *Case A* and *N* force is determined from *Case C*.

3.2.10. *Case K*

Like *Case H*, out-of-plane forces for argument of perigee are out of consideration in *Case K*. In-plane forces are used by considering the coupled dynamics of perigee altitude and argument of perigee. The resultant graph for determination of the control force direction is given in Figure 15.

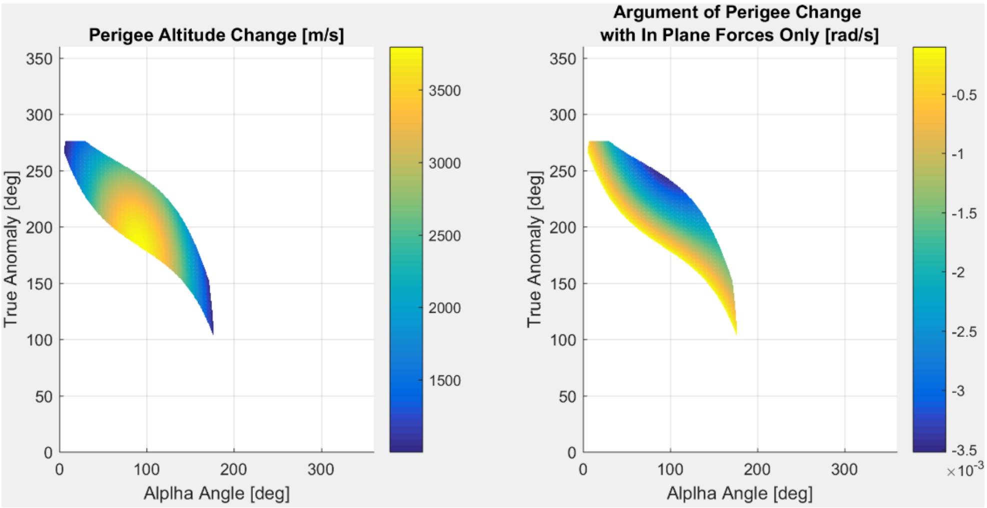

Figure 15: Perigee Altitude and Argument of Perigee with respect to α

3.2.11. *Case L*

Similarly, the control force direction is given in Figure 12. *F* force is determined from *Case B* and *N* force is determined from *Case C*.

3.2.12. *Case M*

Similarly, the control force direction is given in Figure 12. *F* force is determined from *Case B* and *N* force is determined from *Case D*.

3.2.13. *Case N*

Similarly, the control force direction is given in Figure 12. *F* force is determined from *Case K* and *N* force is determined from *Case C*.

3.2.14. *Case O*

The control force direction is given in Figure 12. *F* force is determined as given in Figure 16 and *N* force is determined from *Case* C. The graph given in Figure 16 is generated by considering the coupled dynamics of apogee altitude and argument of perigee.

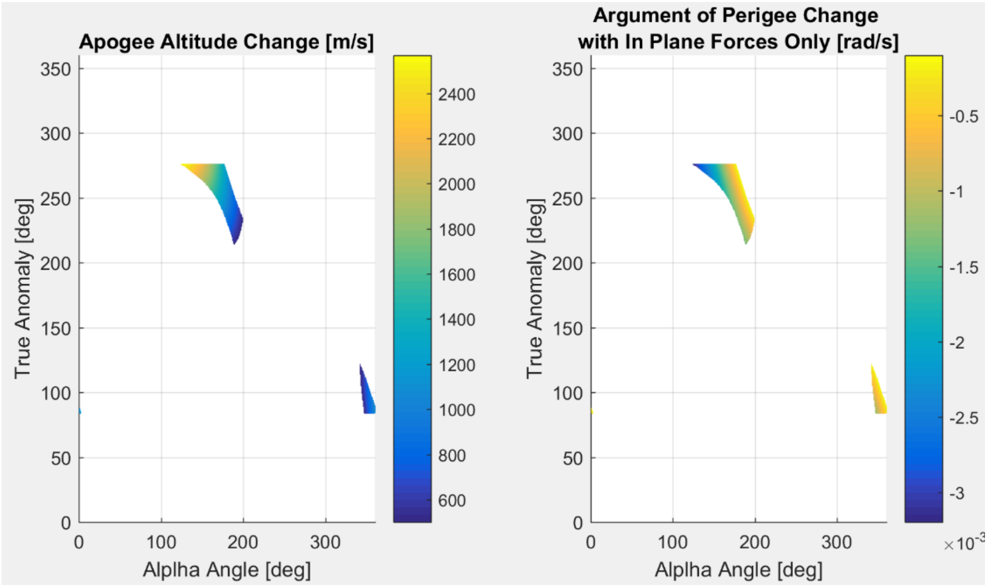

Figure 16: Co-Illustration of Apogee Altitude and Argument of Perigee Changes

3.2.15. *Case Zero* or *Attitude Mode=0* Condition:

This case is valid when no station-keeping condition is triggered. The sail direction is oriented such that minimum energy loss is obtained.

3.3. SIMULATION RESULTS

Finally, an extensive simulation is done. Simulation conditions given in Table 4 result in 384 different combinations. Force models for orbit propagation is modelled as listed in Table 5. Moreover, the satellite properties used in the simulation are also given in Table 5.

Table 4: Simulation Setup

Simulation Conditions	Number of Different Sets
Different Case Cycles	4 sets
Initial Date of Year	4 sets
Inclination	3 sets
RAAN	4 sets
Argument of Perigee	2 sets
Total	384 combinations

Table 5: Force Model and Satellite Properties

Force Models	Satellite Properties
Earth's Gravity	Sail Mass: 77 kg
Earth's Asphericity Effects (up to 4Th Order)	Payload Mass: 37 kg
Third Body Effects (Sun and Moon)	Sail Area: 40x40 m^2
Aerodynamic Drag	Payload: 40x40x75 cm
Solar Radiation Force	Sail Efficiency = 90%

The results of the simulation are given in four graphs to demonstrate the aimed goal of station-keeping strategy of the R2S2 concept:

- Case switching status showing the case transitions based on the triggering conditions,
- Attitude mode showing that there is sailing manoeuvres or not,
- Perigee altitude trends,
- Orbit altitude trends.

First example resultant graphs are given in Figure 17. One can see that successive perigee altitude control between 350 and 400 km. Moreover, apogee altitude drop is diminished.

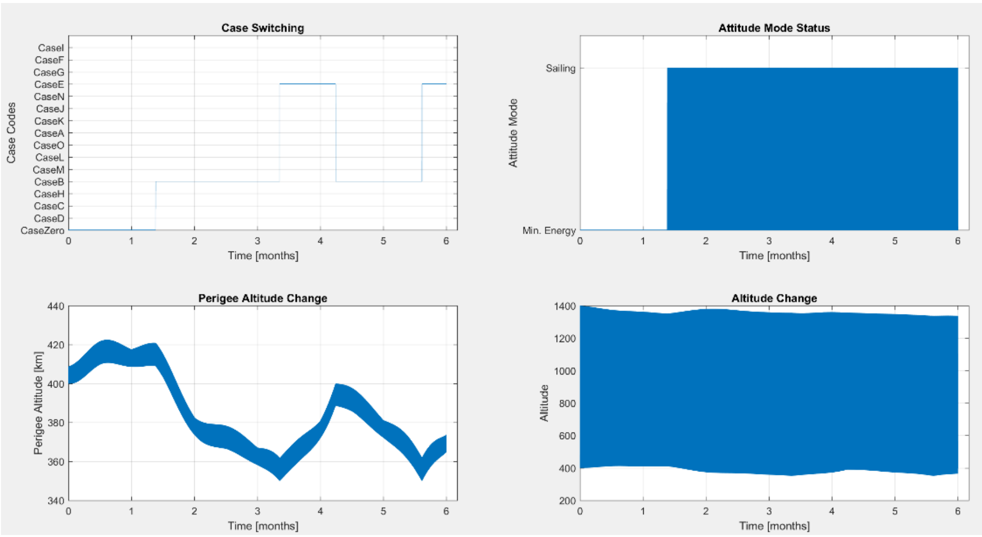

Figure 17: An Example of Simulation Results for First Case Cycle

The example in Figure 18 demonstrates the continuous station-keeping effort on argument of perigee. The switching up and down of cases implies an effective station-keeping effort. One more thing to notice here is the gaps within the attitude mode graph. These are the results of Sun-Earth-Satellite orientation that provides no valid condition for sailing manoeuvre. Thus, providing the requested control force direction is geometrically impossible.

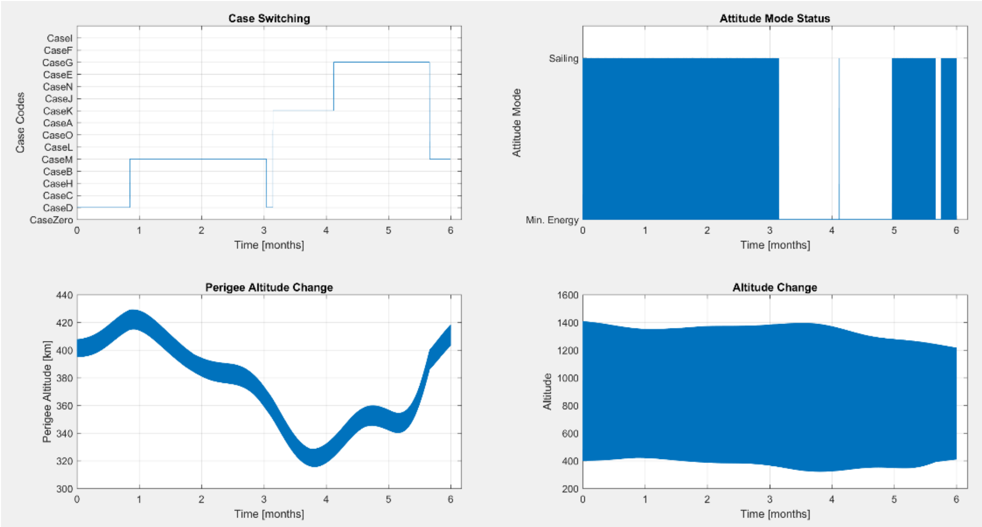

Figure 18: An Example of Simulation Results for Second Case Cycle

Similarly, RAAN station-keeping is continuous in Figure 19 and successful perigee altitude is restored even with the gaps within the attitude mode graph.

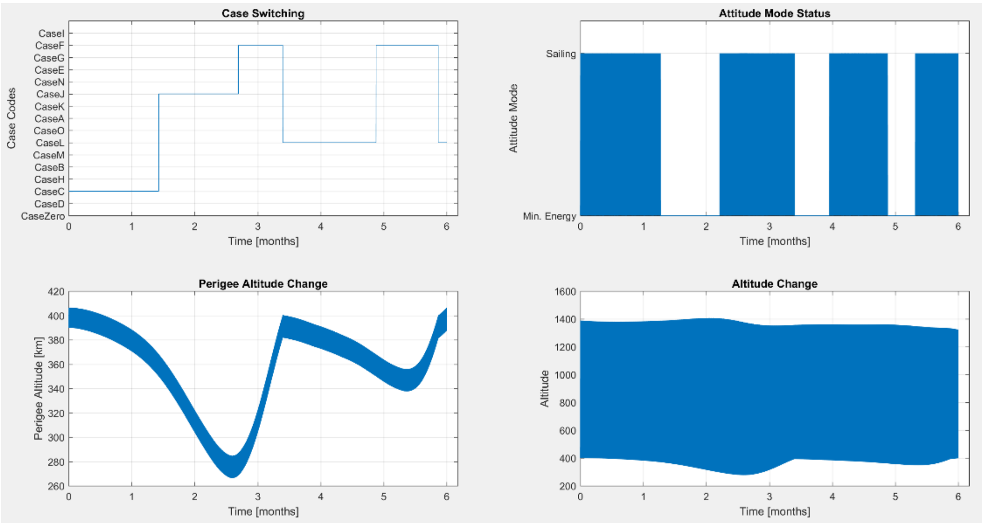

Figure 19: An Example of Simulation Results for Third Case Cycle

Last example is from a case with continuous argument of perigee and RAAN station-keeping efforts and results are given in Figure 20. Case transitions and restoring of the perigee and apogee altitudes are the implications of successful station-keeping strategy.

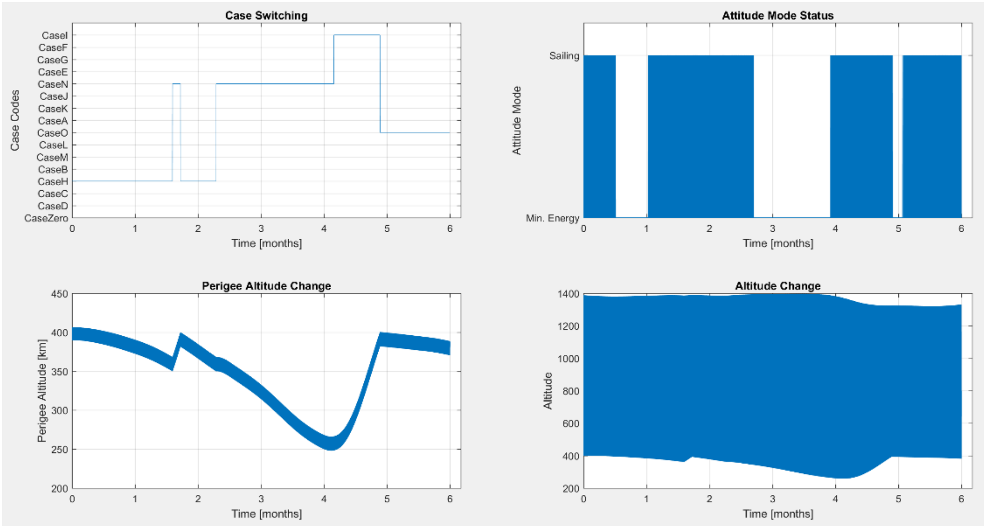

Figure 20: An Example of Simulation Results for Fourth Case Cycle

4. CONCLUSION

In this study, station-keeping strategy for the mission orbit of the R2S2 concept is provided with mathematical formulizations and simulation results. With four orbital parameters in consideration (perigee and apogee altitudes, argument of perigee and right ascension of ascending node), 16 different station-keeping cases are analysed and solution to the control force direction determination problem for each case is explained.

The coupled dynamics of orbital parameters are handled with the approximation of modified Gaussian Variational Equations. The original Gaussian Variational Equations are modified to better exploit for the elliptical orbital dynamics and this modification allows us to determine the control force direction under the coupled dynamics of four orbital parameters in consideration.

Moreover, principles for station-keeping strategy of the proposed concept of R2S2 is also respected when dealing with coupled dynamics of orbital parameters. Finally, simulation results show that solar sailing at low Earth orbit is adequate for station-keeping purposes. However, for the parameters of argument of perigee and right ascension of ascending node, the selected sail dimensions and properties do not give high performance for extending the mission lifetime.

These station-keeping performances (even for the argument of perigee and RAAN) can be increased by either reducing the payload mass (mission satellite), increasing the sail area, or reducing the sail loading (total mass per sail area). The miniaturization technology for the satellites and the improvements on the super-light materials will obviously increase the performance results of this study.

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