The Effect of Atmospheric Drag Force on the Elements of Low Earth Orbital Satellites at Minimum Solar Activity

Najlaa Ozaar Hasan[1*], Wafaa Hasan Ali Zaki[2], Ahmed Kader Izzet[3]

Abstract

Researching and modeling perturbations is essential in astrodynamics because it gives information on the deviations from the satellite's normal, idealized, or unperturbed motion. Examined the impact of non-conservative atmospheric drag and orbital elements of low-earth-orbit satellites under low solar activity. The study is consisting of parts, the first looks at the effects of atmospheric drag on LEO satellites different area to mass ratios, and the second looks at different inclination values. Modeling the impacts of perturbation is included in each section, and the final portion determines the effects of atmospheric drag at various node values. The simulation was run using the Celestial Mechanics software system's SATORB module (Beutler, 2005), which solves the perturbation equations via numerical integration. The findings were examined using Matlab 2012. Conclusion that the impacts are stronger for retrograde orbits, which is due to the fact that the satellite moves in the opposite direction. The atmospheric drag effects for all orbital elements were increased by increasing the area to mass ratio. When the node value rises, the size parameter changes slightly, but the other orbital elements change. At varying inclinations, it is found that the changes in orbital elements due to atmospheric drug.

Key Words: Perturbations, Orbital Elements, Atmospheric Drug, Low Orbit Satellites.

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Introduction

There are various sources of perturbations affecting satellite orbital motion. In general, orbital perturbations are divided into gravitational and nongravitational force. The gravitational force includes the gravitational influences of the Sun and Moon, the Earth's gravitational force, the Earth's oblateness, and the zonal sectorial spherical harmonics, while the non-gravitational force include the atmospheric drag force (the dominant for Low Earth Orbital), the solar radiation pressure (effective for geosynchronous satellites), and magnetic forces (due to the interaction of the Earth's magnetic field with the dipole moment which motivated in the satellite). The gravitational potential of the non-spherical Earth models was initiated by a short period and long period perturbations (Khodairy et al. 2020). In this investigation the effect of perturbation of atmospheric drug on the orbital elements of low orbital satellite, were applied at low solar activity which defined as (the Extreme Ultra-Violet, EUV, radiation that heats the upper atmosphere, FEUV) (Vallado 1997).

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So, the low value of magnetic index A_P(The earth has a magnetic field surrounding it called the geomagnetic field) (Vampola and Korth 1992). Were applied, and studied the influence of their disturbances on the satellite orbital elements of multiple periods of prograde and retrograde satellite orbits Movement of orbital elements toward and against the Earth's rotation (Cornish 2008).

Celestial Mechanics transaction with the orbital and rotational movement of celestial bodies (Brumberg 2013) The dynamics of stellar systems, the movement of stars within galaxies, the dynamics of planetary systems. This study will concentrate on the orbital movement of artificial satellites around the Earth (Beutler 2005). In this study, the Gaussian and Lagrange equations are used to find the extent of the perturbations of the elements, this study will discuss the perturbations of atmospheric drug on low earth orbit. The (Celestial mechanics) program used to simulation the orbital elements under the atmospheric drug perturbations. The main goal of this study is to find out the effect of atmospheric perturbations on the orbital elements at low solar activity.

Orbital Elements

The orbital elements are the parameters required to uniquely identify the orbits for the satellite, these called Keplerian. These are six orbital elements for elliptical orbit, two of them are describe the size and the shape of the orbit, these are the **semi major axis (a)** and the **Eccentricity (e)**. the other elements are the rotation elements, which are describe the orientation of the orbital plane. These are: **The Inclination angle (i)** of the orbital plane from the equatorial plane, the **argument of perigee (ω)**, that is the angle from the ascending node to the perigee and **the right ascension of ascending node (Ω)**, that is the angle from the vernal equinox to the ascending node of the equatorial plane (Roy 1988; Weiner and Homsley 1976) **Mean anomaly (σ_0)**: It is alternative for the time of pericenter, by replacing T_0 by the mean anomaly σ_0 referring to the initial epoch t_0. The transformation equation relating σ_0 and T_0 is [6]: $\sigma_0 \stackrel{\text{def}}{=} n\,(t_0 - T_0)$

where: $n = \sqrt{\frac{\mu}{a^3}}$-

n : is the osculating mean motion of the celestial body.

Most Keplerian problems were treated as ideal or under the basic assumptions that the motion of a body in the orbits is a result of the gravitational attraction between two bodies, this ideal situation does not exist, additional forces acting on any moving body must be taken into account, these additional forces called the perturbing forces. The orbital elements are deviation due to external forces and disturbances from their mean values. These forces can be divided into two categories: gravitational and non-gravitational forces. Gravity is generated by the mutual gravitational force between various celestial bodies such as the earth and satellite. Non-gravitational forces are generated in the space environment, such as solar radiation pressure, atmospheric drag, geomagnetic field, etc (Chobotov 1996; Roy 1988; Weiner and Homsley 1976).

Atmospheric Drag

Atmospheric drug is atmospheric force acting opposite to the relative motion of an object, and could slow the motion of a satellite when its orbit is low enough to be affected by the force of Earth's atmosphere (Bhatnagar and Mitra 1966). So, in the case of low solar activity solar wind to word the earth will heat up and expand to enclose the orbits of some low Earth orbiting satellites but relatively less than the influences of high solar activity (Bhatnagar and Mitra 1966). The greatest value of drag is during launch and reentry. The action of drag of the satellite will cause it to spiral back in to the atmosphere and eventually disintegrate or burn. If the spacecraft enters the Earth's surface (120 to 160 km), atmospheric drag will lower it within a few days, and the final disintegration will occur at an altitude of approximately (80 km). Atmospheric drag causes satellite reentry (Bhatnagar and Mitra 1966; du-Toit, Du Plessis, and Steyn 1996; Official website of china Manned space 2018).

The perturbing forces that cause the satellite orbit deviate from a theoretically regular orbital motion are classified into two types conservative and non-conservative perturbing forces, with the effected of perturbations, a satellite orbit will not simply be the two-body motion. The magnitude of perturbation varies with the altitude at LEO orbit, the atmospheric drag could be ignored because of the dramatic drop of the atmospheric density. In this research the drug perturbation type is calculated and studied the influence of it's on the satellite orbital elements through multiple cycle of low prograde (inclination <90°) and retrograde (inclination >90°) satellite orbits and (Romagnoli and Theil 2012; Snider 1989; Wesam 2011). And different node, and different area to mass under low solar activity.

$$\vec{a}_{drag} = -\frac{1}{2}\frac{C_D A}{m}\rho v_{rel}^2 \frac{\vec{v}_{rel}}{|\vec{v}_{rel}|} \quad (1)$$

c_D: the coefficient of drag,
ρ: is the atmospheric density,
m: the mass of satellite,
. A: is the cross-sectional area,
$\vec{v}_{rel}$: relative velocity vector.
As well as the expression perturbed movement hint that there is an undisturbed motion. In Celestial Mechanics the undisturbed motion is the orbital motion of two spherically symmetric bodies (Beutler 2005). The perturbed motion of a celestial body is defined as the solution of an initial value problem of the following type:

$$\ddot{r} = -\mu\left(\frac{r}{r^3}\right) + \delta f(t, r, \dot{r}) \quad (2)$$

The differential equation system (2) is named the system of perturbation equations or simply the perturbation equations (Beutler 2005).

$$r(t_0) = r_0 \text{ and } \dot{r}(t_0) = v_0 \quad (3)$$

μ: is: the constant of gravitation and the sum of the masses of the two bodies considered.
For a system of point masses there is one such equation for each of the bodies, except for the central body to which the position vectors refer.
$-\mu(r/r^3)$: is called the two-body term,
δf: is the perturbation term (Beutler 2005).
The terminology makes sense if the perturbation term is considerably smaller than the two-body term, i.e., if

$$|\delta f| \ll |-\mu(r/r^3)| \quad (4)$$

1. Gaussian Perturbation Equations

The concept of osculating elements, appointed one set of osculating orbital elements to each epoch t by the position and velocity vectors r(t) and $\dot{r}$(t). There is a relationship
(one-to-one) between the corresponding state vector and the osculating elements of epoch t. Perturbations transformation equations between the two sets of functions are those of the two-body problem (Beutler 2005):

$$I(t) \in \{a(t), e(t), i(t), \Omega(t), \omega(t), T_0(t)\} \quad (5)$$

t: is the osculation epoch.
The Gaussian perturbation equations for the orbital elements are (Beutler 2005):

$$\begin{aligned}
\dot{a} &= \sqrt{\frac{p}{\mu}}\frac{2a}{1-e^2}\left\{e\sin vR' + \frac{p}{r}S'\right\} \\
\dot{e} &= \sqrt{\frac{p}{\mu}}\{\sin vR' + (\cos v + \cos E)S'\} \\
\dot{T}_0 &= -\frac{1-e^2}{n^2ae}\left\{\left(\cos v - 2e\frac{r}{p}\right)R' - \left(1+\frac{r}{p}\right)\sin vS'\right\} - \frac{3}{2a}(t-T_0)\dot{a} \\
\frac{di}{dt} &= \frac{r\cos u}{na^2\sqrt{1-e^2}}W' \\
\dot{\Omega} &= \frac{r\sin u}{na^2\sqrt{1-e^2}\sin i}W' \\
\dot{\omega} &= \frac{1}{e}\sqrt{\frac{p}{\mu}}\left\{-\cos vR' + \left(1+\frac{r}{p}\right)\sin vS'\right\} - \cos i\,\dot{\Omega}
\end{aligned} \quad (6)$$

$$\sigma_0 \stackrel{\text{def}}{=} n(t_0 - T_0) \quad (7)$$

$$\dot{\sigma}_0 = \frac{1-e^2}{nae}\left\{\left(\cos v - 2e\frac{r}{p}\right)R' - \left(1+\frac{r}{p}\right)\sin vS'\right\} + \frac{3n}{2a}(t-t_0)\,\dot{a} \quad (8)$$

Where σ_0 : mean anomaly,
the separation of the perturbing acceleration into the:
R': radial,
S': normal to radial in the orbital plane,
W': the out-of-plane direction (Beutler 2005).
T_0: The gradients of the time of pericenter passage,
And
v: true anomaly,
E: the eccentric anomaly,
The semi parameter, is the distance from the primary focus to the p:
orbit. It's measured perpendicular to the major axis except on the circle,
u: the argument of latitude of the celestial body considered, where:

$$u = \omega + v \quad (9)$$

2. Lagrange's Planetary Equations

The Lagrangian perturbation equations is derive by their general form (10) for the six elements (Beutler 2005):

$$\dot{I}_k = \sum_{j=1}^{6}[I_k, J_j]\frac{\partial R}{\partial I_j}, \; k=1,2,\ldots,6 \quad (10)$$

Where R: is the perturbation function.
Lagrange equation can be defined as:

$$\begin{aligned}
\dot{a} &= \mp\frac{2}{n^2a}\frac{\partial R}{\partial T_0} \\
\dot{e} &= -\frac{\sqrt{|1-e^2|}}{na^2e}\frac{\partial R}{\partial \omega} - \frac{1-e^2}{n^2a^2e}\frac{\partial R}{\partial T_0} \\
\frac{di}{dt} &= -\frac{1}{na^2\sqrt{|1-e^2|}\sin i}\frac{\partial R}{\partial \Omega} + \frac{\cot i}{na^2\sqrt{|1-e^2|}}\frac{\partial R}{\partial \omega} \\
\dot{\Omega} &= \frac{1}{na^2\sqrt{|1-e^2|}\sin i}\frac{\partial R}{\partial i} \\
\dot{\omega} &= \frac{\sqrt{|1-e^2|}}{na^2e}\frac{\partial R}{\partial e} - \frac{\cot i}{na^2\sqrt{|1-e^2|}}\frac{\partial R}{\partial i} \\
\dot{T}_0 &= \frac{2}{n^2a}\frac{\partial R}{\partial a} + \frac{1-e^2}{n^2a^2e}\frac{\partial R}{\partial e}
\end{aligned} \quad (12)$$

Above are Lagrange equations for perturbations of orbital elements.

Application and Results

In this study the Celestial mechanics program used to calculate the variation in orbital elements for artificial satellite under the effect of drug force for low solar activity. In the figures (1 - 13) examine the effects of atmospheric drug on LEO over one day at low solar activity.

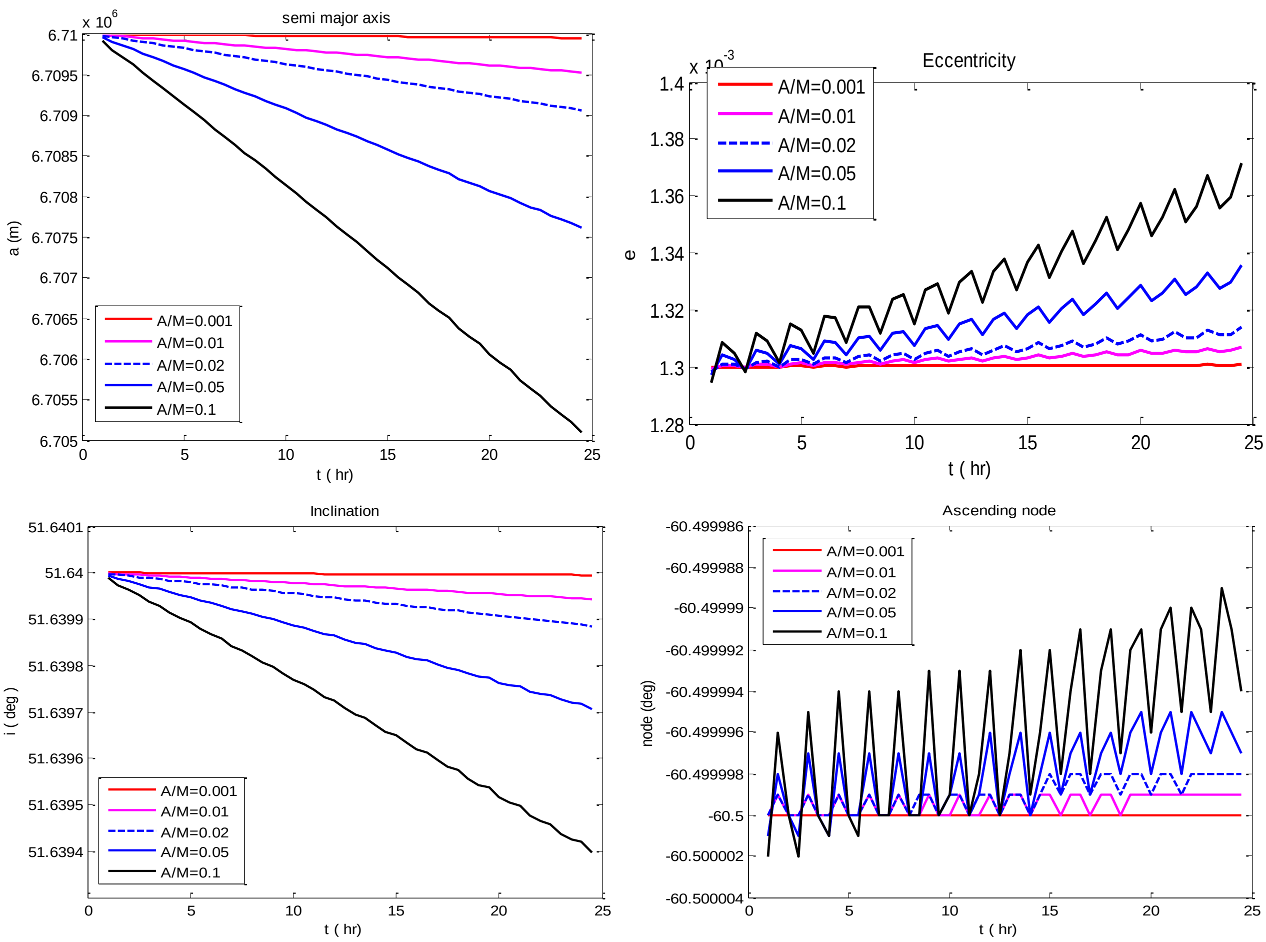
The study of atmospheric drug is divided in to three parts, first one includes their effect on LEO satellite of different area to mass ratio (A/m), second part include the effect at different values of inclination while the third part of study take in to account the effect of atmospheric drug on LEO satellite at different node values, and the values of parameters which are used are: semi major axis a equal to 6710 km, eccentricity e equal to 0.0013, for the low solar flux $F_{10.7}$= 100 SFU (Solar Flux Units), and the magnetic index A_P equal to 4 gammas.

1. Area to Mass

Figures (1-2) represent the change in the orbital elements of prograde and retrograde orbit at different area to mass ratio at low solar activity, its noticed that with increasing the ratio the orbital elements are affected in general and the figures shows secular changes in elements. By comparing the elements in figure 1 with their counterparts in figure 2, it's notice that the effect was greater at retrograde orbit than that of prograde orbit, this due to satellite in retrograde orbit it will encounter a greater obstruction due to its rotation in the opposite direction of earth's rotation. Table 1 gives the changes in orbital elements in the two cases of prograde and retrograde orbit at low solar activity at area to mass ratio equal to 0.1 m^2/kg.

Table 1. The variation in orbital elements at prograde and retrograde orbit at (A/m)=0.1 m^2/kg

Orbital Elements	Δa(km)	Δe × 10^{-3}	Δi(degree)	ΔΩ(degree)
prograde orbit(i=51.46)	4.9	0.077	0.0006	0.000013
retrograde orbit(i=97)	5.1	0.098	0.0008	0.000019



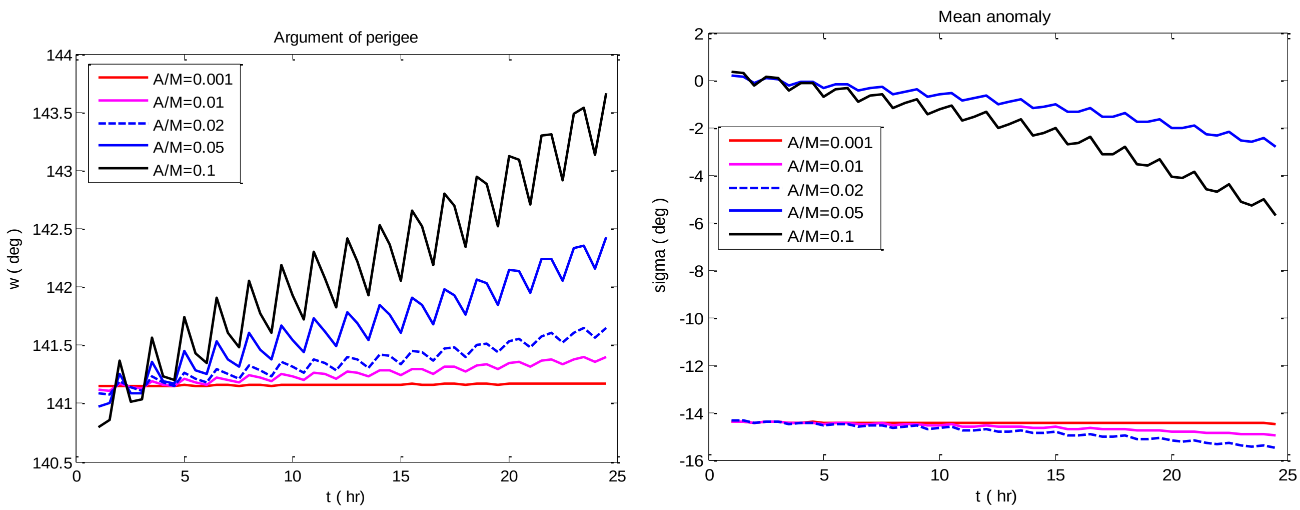
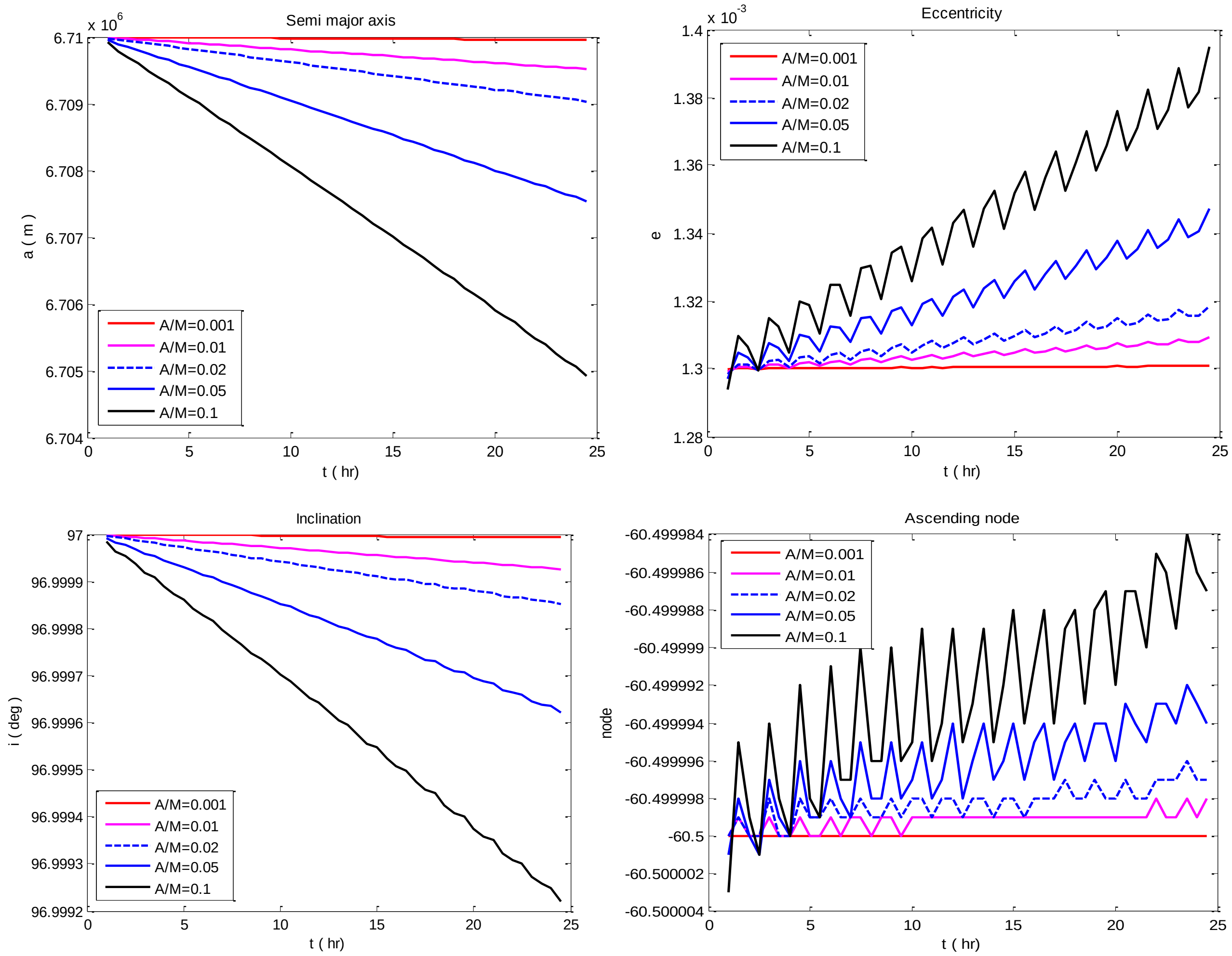


Figure 1. Variations in orbital elements at different values of the area to mass ratio during low solar activity at inclination equal to 51.64 degree.



28

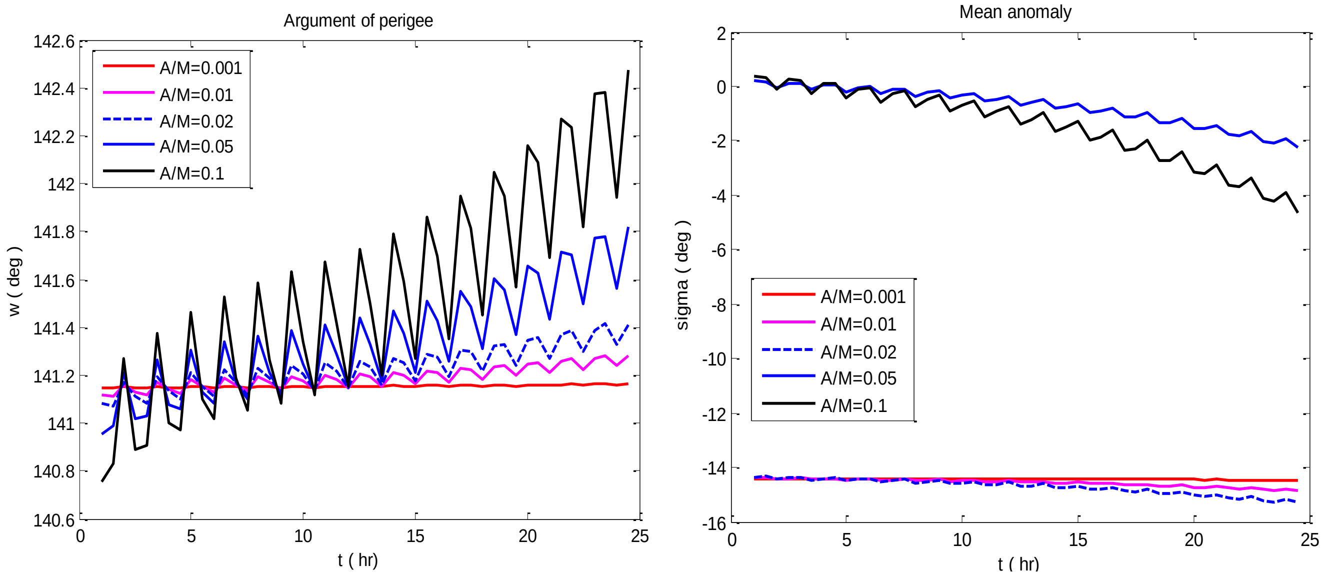
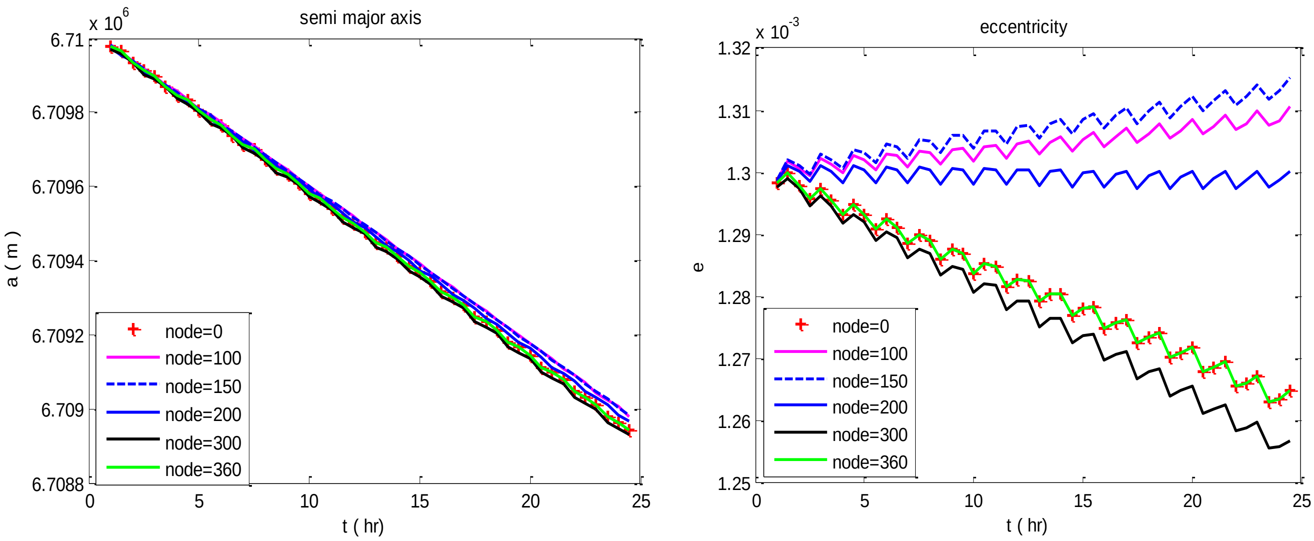


Figure 2. Variations in orbital elements at the different value of area to mass ratio during low solar activity at inclination equal to 97 degree.

2. Ascending Node

Figures (3-7) show the variation in orbital elements at various node values during low solar activity. The size parameter (semi major) elements and inclination show only minor variations with changing node values, whereas the shape parameter (eccentricity) and mean anomaly are affected by changing node values. For all inclinations, the smallest variance in eccentricity was found at i=200°, while the largest variation was recorded at i=300°. While for prograde orbit, the smallest variation in mean anomaly is at i =200°, and for retrograde orbit, the highest variance is at the same above node value. Also observe that in the figures when the inclination parameter is set to (0°,180°), the modification in the inclination parameter has no influence on the deferent value of the node.



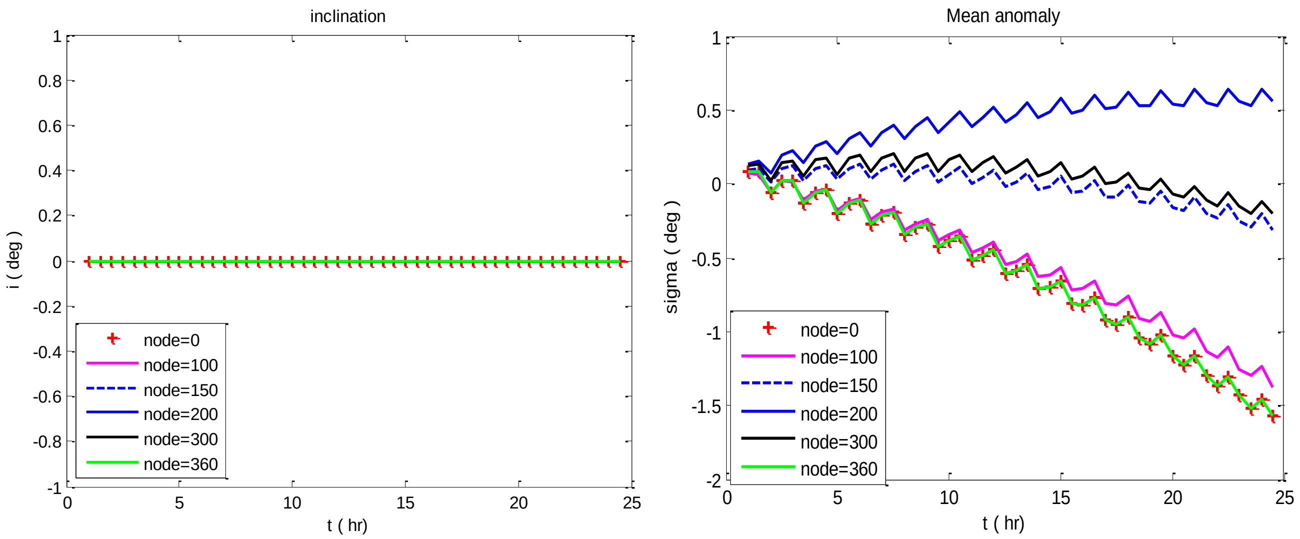


Figure 3. Variations in orbital elements at different value of node during low solar activity at inclination equal to 0 degree and (A/m)=0.02m^2/kg.

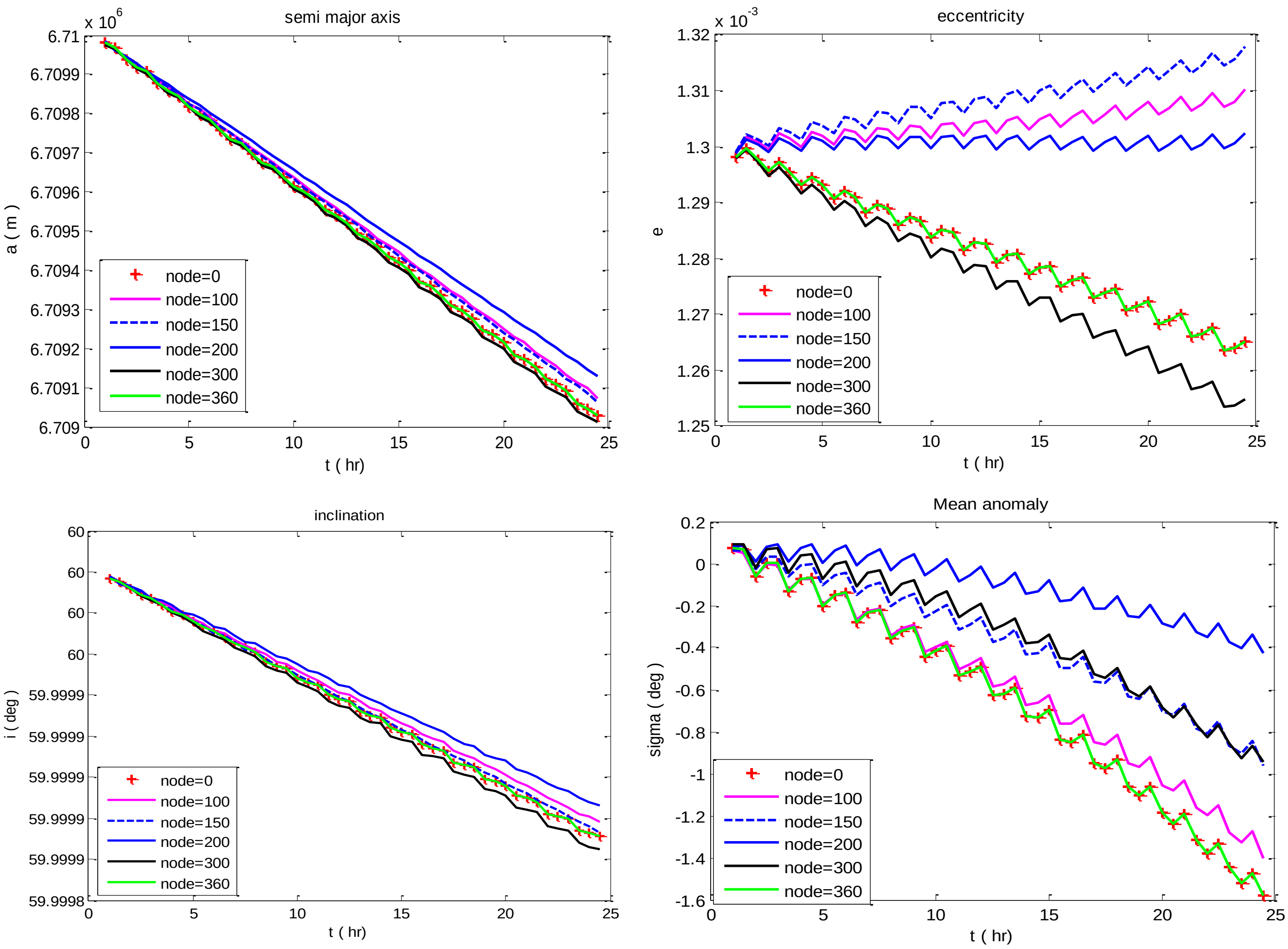


Figure 4. Variations in orbital elements at different value of node during low solar activity at inclination equal to 60 degree at(A/m)=0.02m^2/kg.

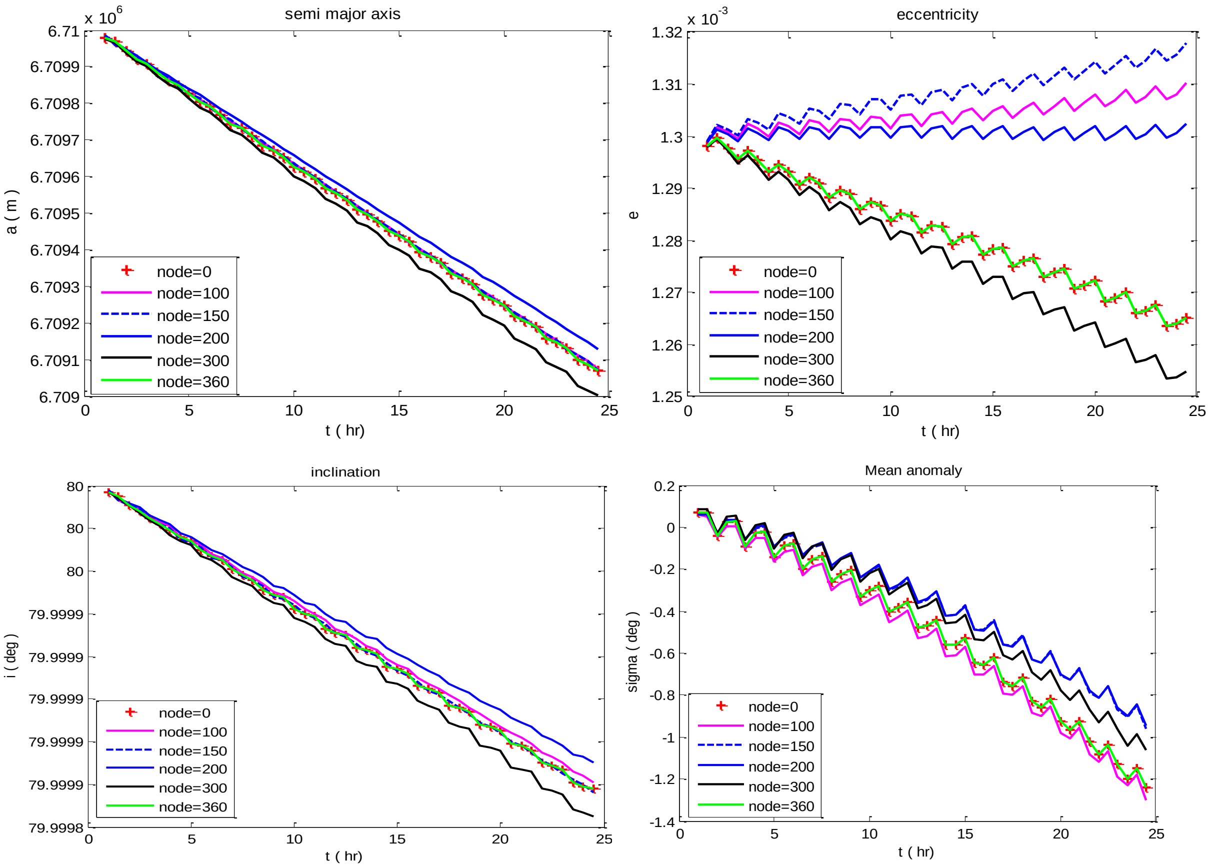


Figure 5. Variations in orbital elements at different value of node during low solar activity at inclination equal to 80 degree at(A/m)=0.02m^2/kg

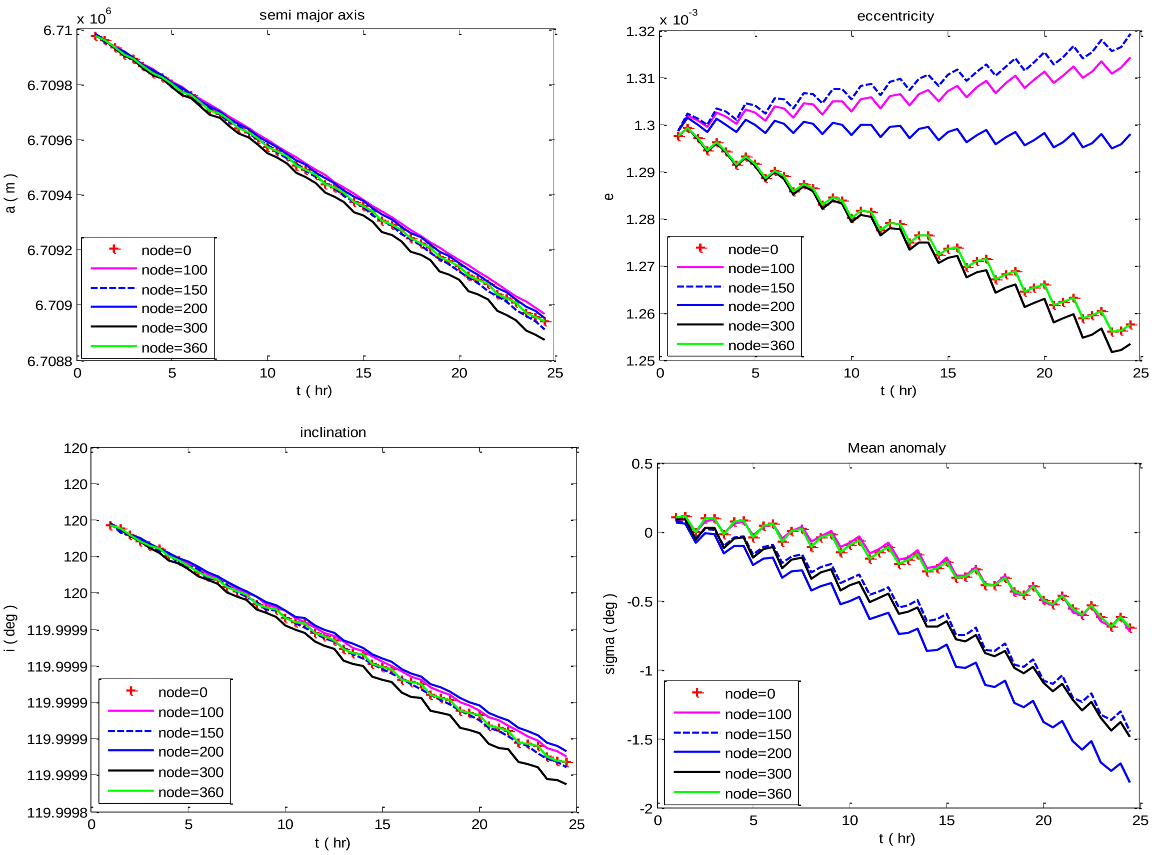


Figure 6. Variations in orbital elements at different value of node during low solar activity at inclination equal to 120 degree at(A/m)=0.02 m^2/kg

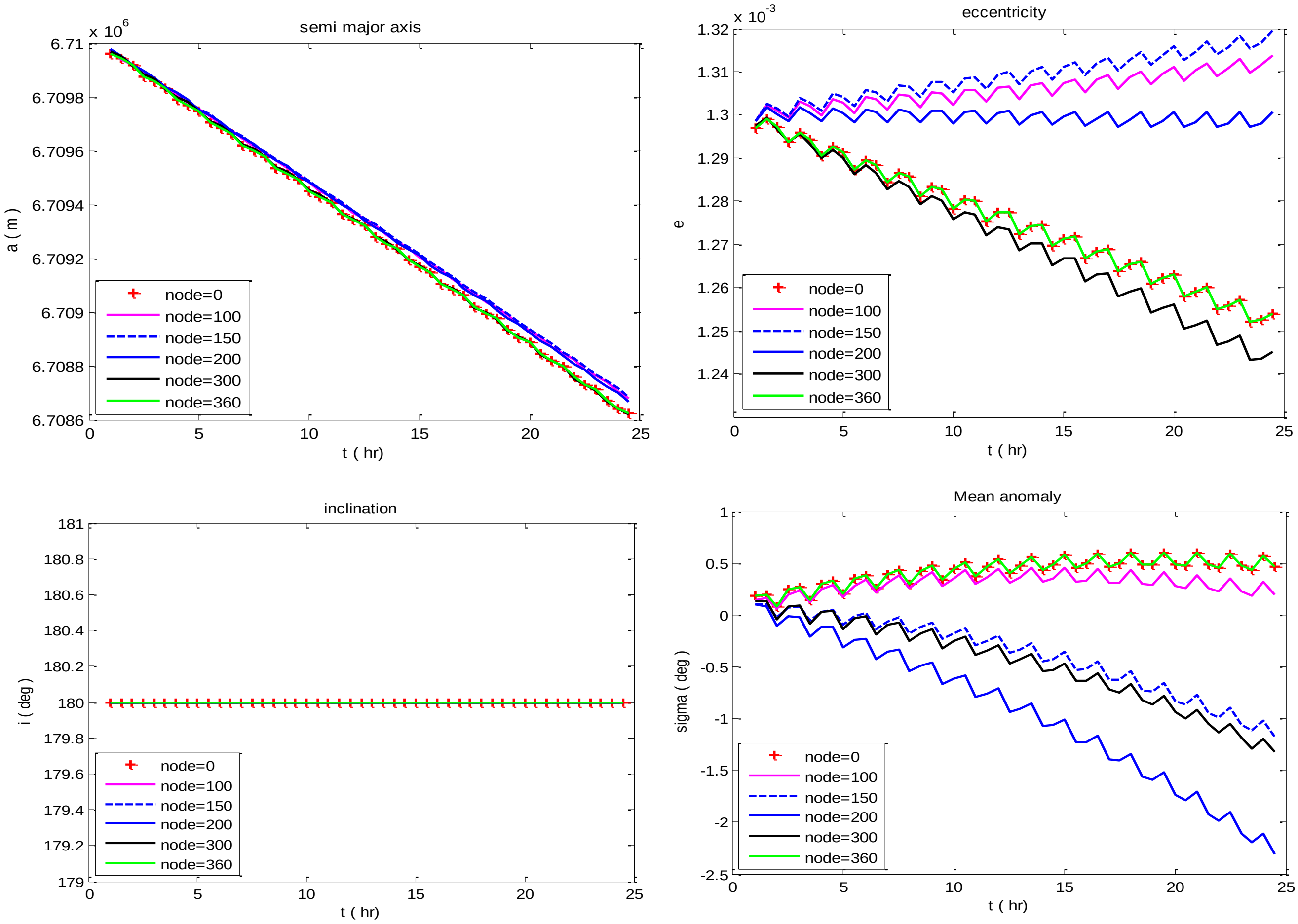
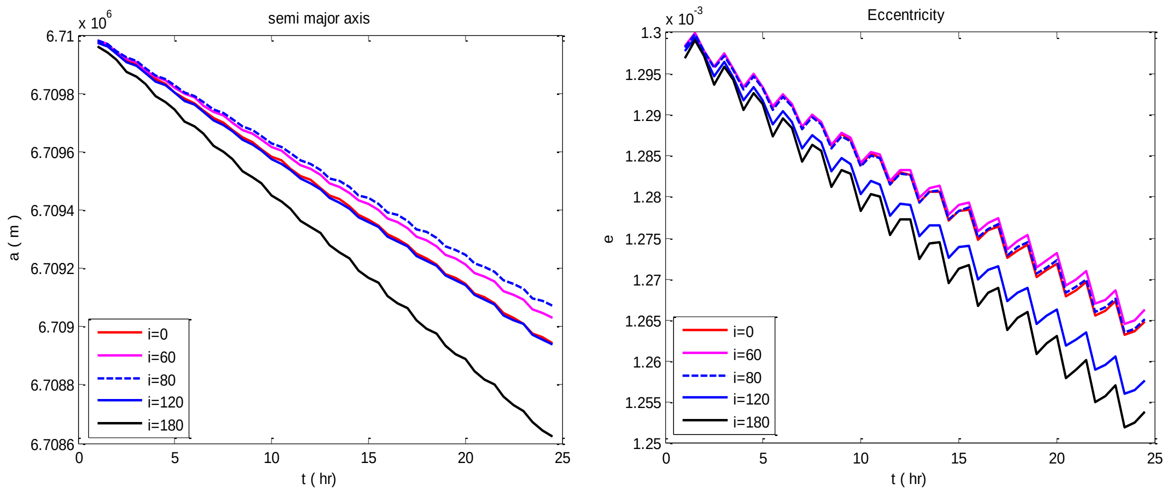


Figure 7. Variations in orbital elements at different value of node during low solar activity at inclination equal to 180 degree at(A/m)=0.02 m^2/kg.

3. Inclination

Figures (8-13) show the variation in orbital elements at different inclinations during low solar activity, with different values of ascending node at area to mass ratio equal to 0.02 m^2/kg. It is notable that the minimum variation in semi major occurs at i=60° and 80°, and the maximum variation occurs at i=180°. Also, at all node's values, the highest variation in eccentricity occurs at an inclination of 180°, whereas the lowest changes are at i=0°. The mean anomaly shows the smallest changes at i=180° for nodes (0°, 100°, 360°) and i=0° for nodes (150°, 200°, 300°).



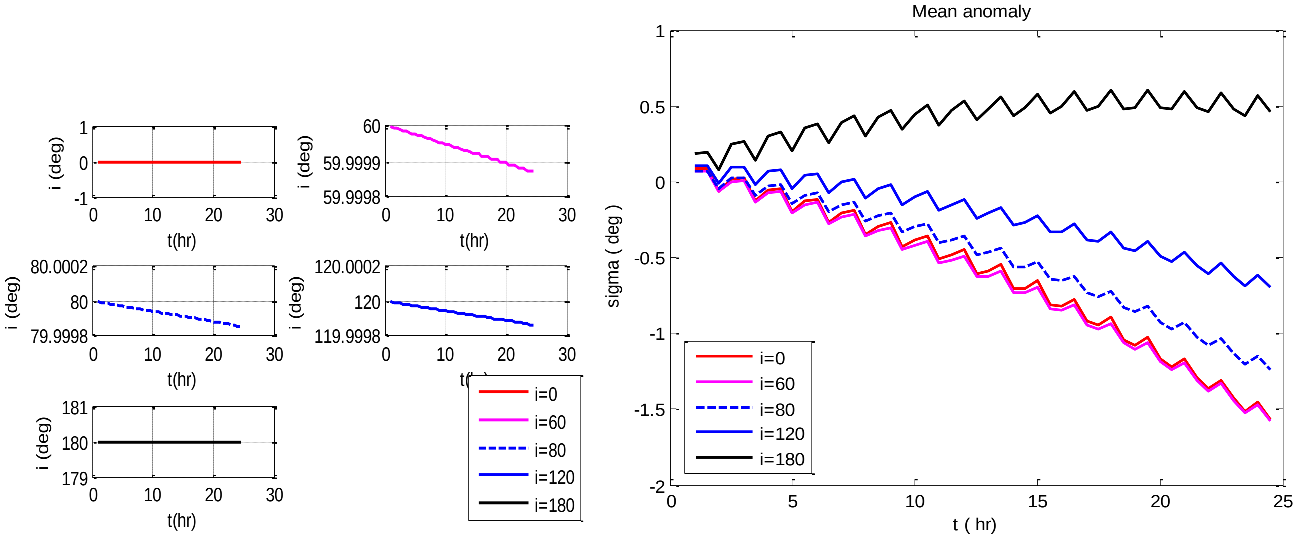


Figure 8. Variations in orbital elements at different value of inclination during low solar activity at Ω=0°

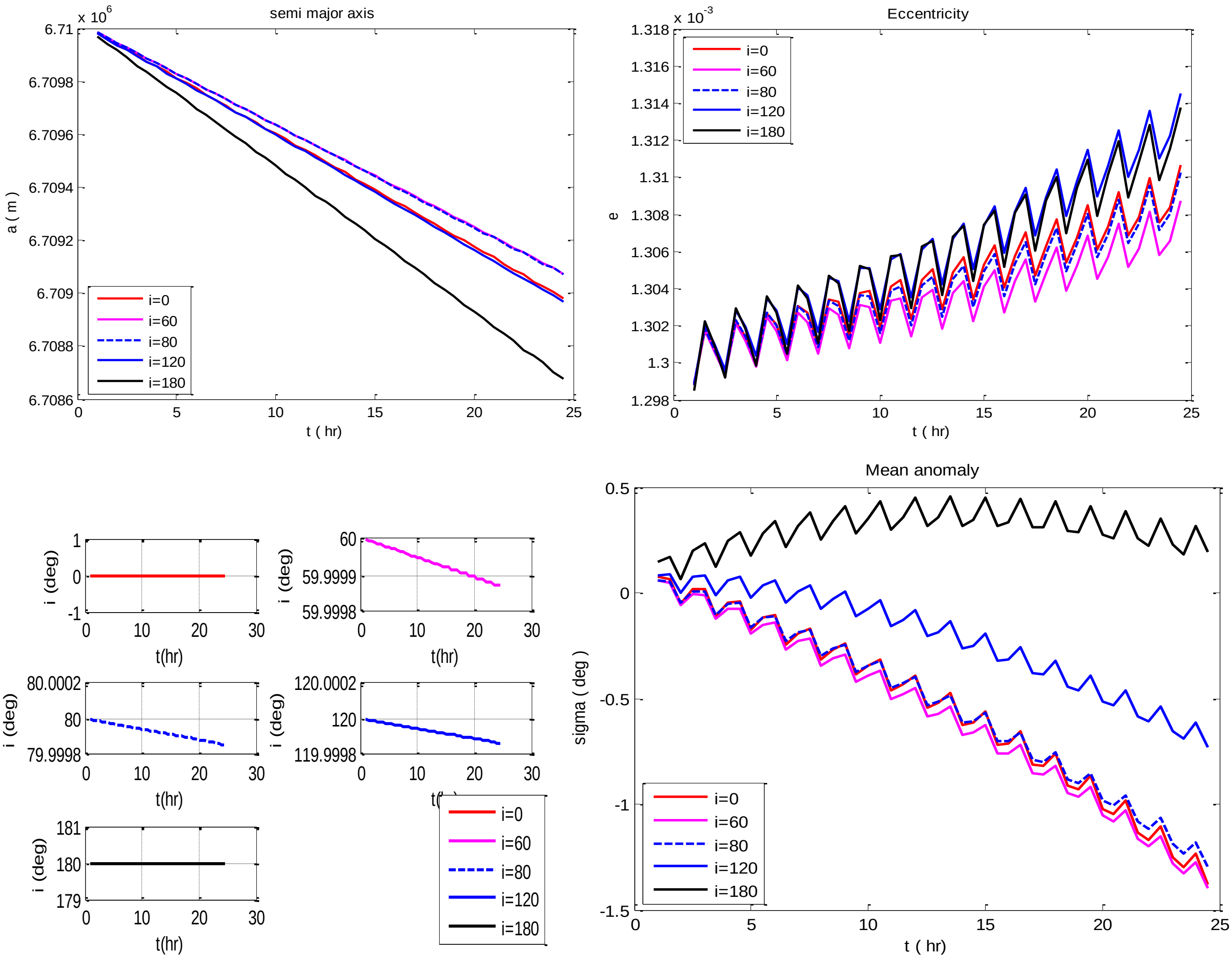


Figure 9. Variations in orbital elements at different value of inclination during low solar activity at Ω=100°

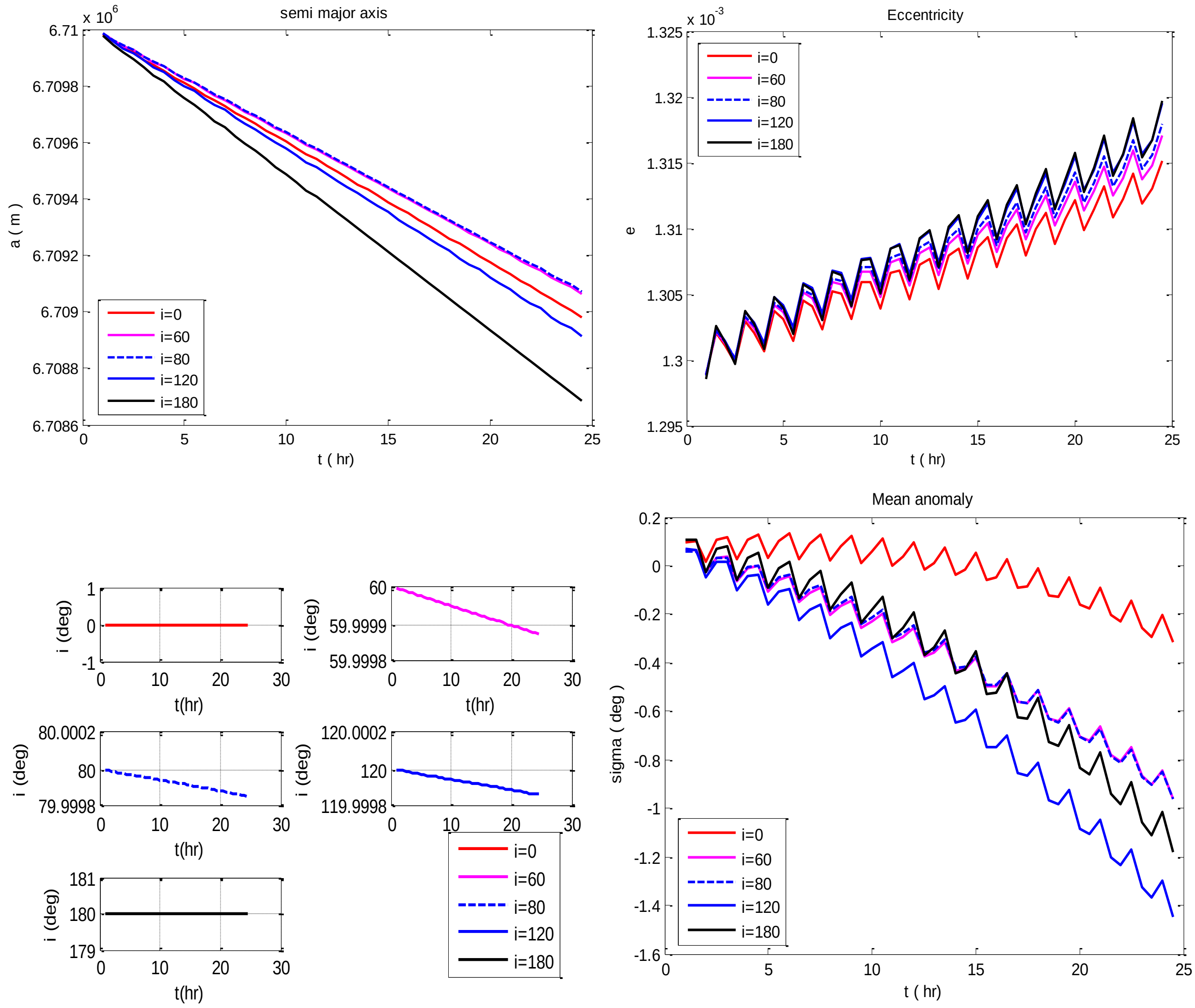
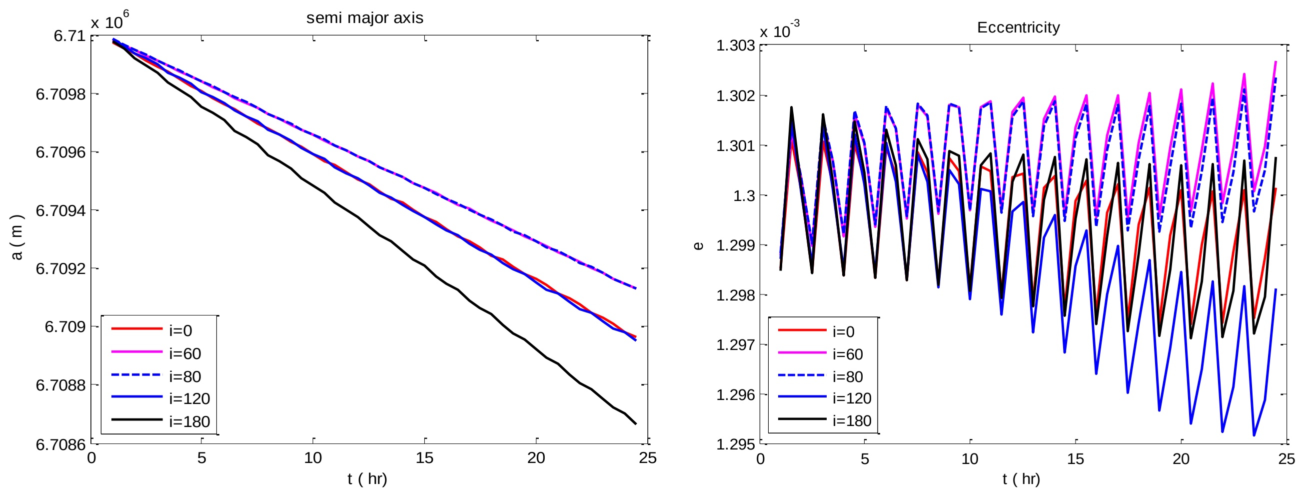


Figure 10. Variations in orbital elements at different value of inclination during low solar activity at Ω=150°



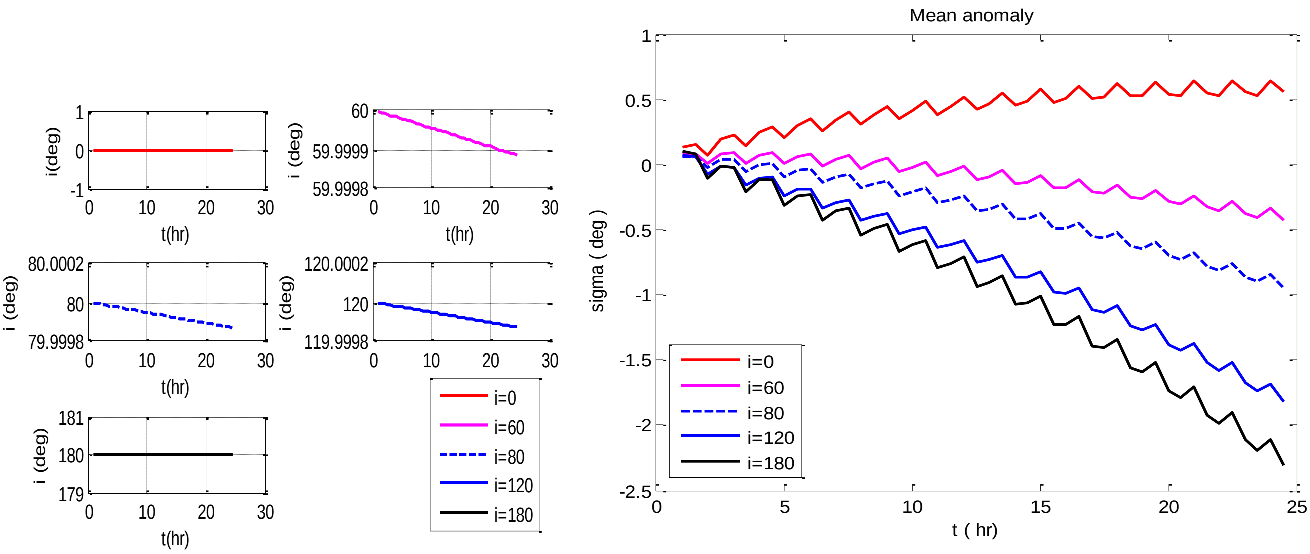


Figure 11. Variations in orbital elements at different value of inclination during low solar activity at Ω=200°

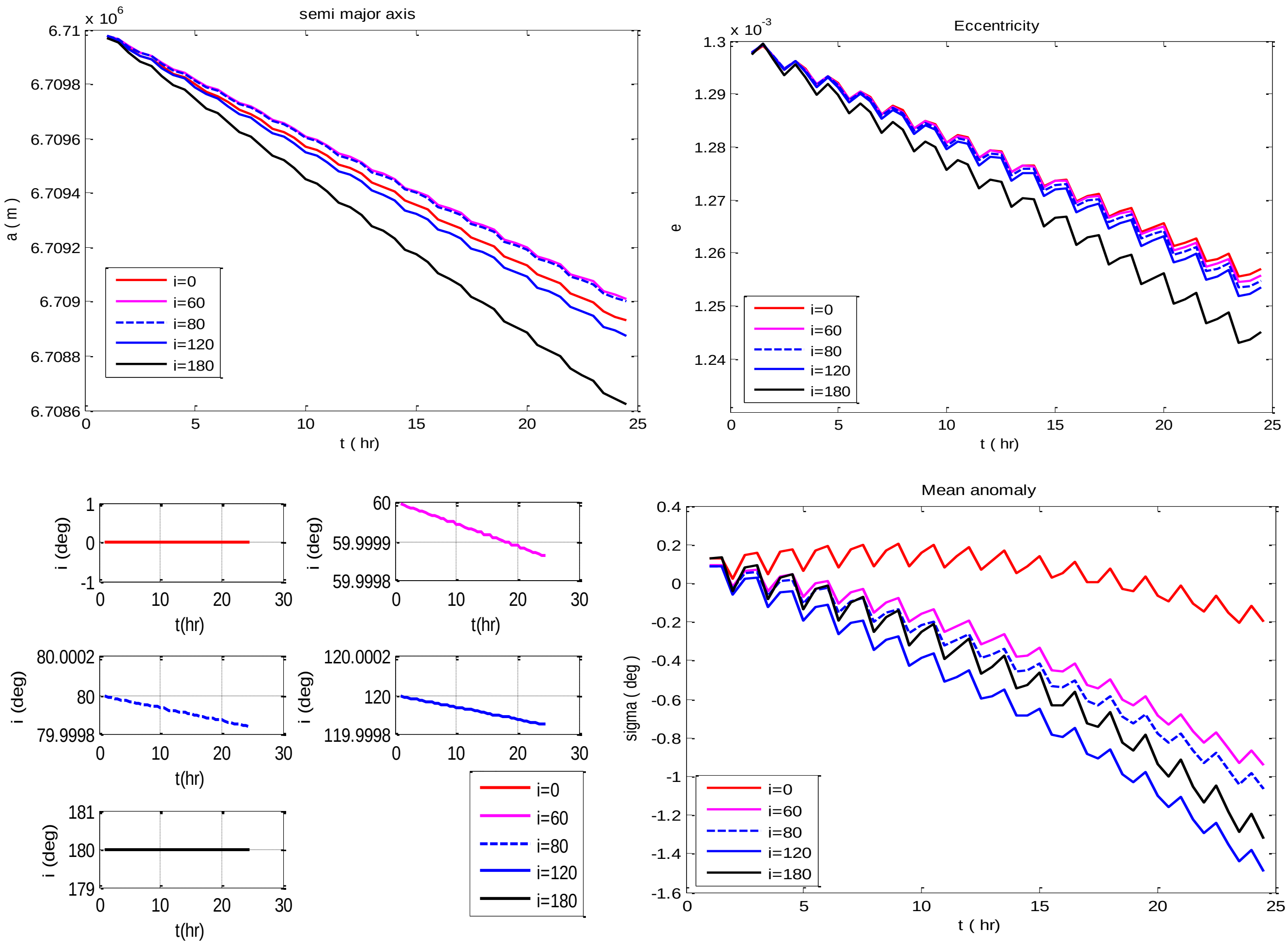


Figure 12. Variations in orbital elements at different value of inclination during low solar activity at Ω=300°

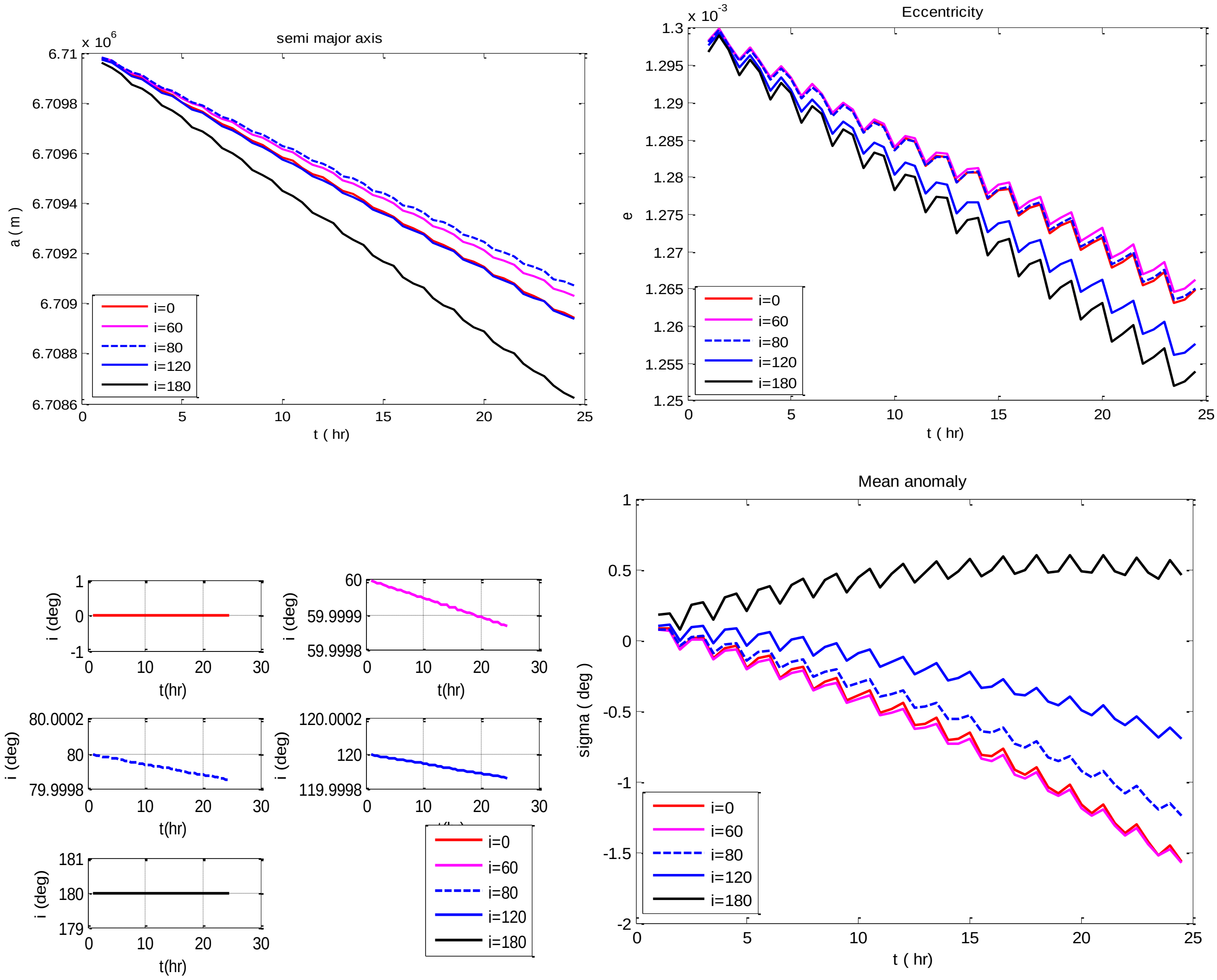


Figure 13. Variations in orbital elements at different value of inclination during low solar activity at Ω=360°

Conclusion

In the present study, the variations of the orbital elements of the satellite in Low Earth Orbital (LEO) environment due to the effect of the atmospheric drag force during minimum solar activity are addressed. And it has been concluded that,

1. The variations of the orbital elements during the epoch of the minimum solar activity are affected in general and show secular changes, when the ratio of area to mass increasing.
2. The variation in orbital elements affected by changing node values. It is found that some elements shows a small variation with changing node values.
3. The variation in orbital elements affected by changing inclinations values.
4. The effect was greater at retrograde orbit than that shown at prograde orbit.

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