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Satellite low orbits variation due to Atmospheric drag , solar radiation pressure and lunar attraction.

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ABSTRACT

Orbital elements are the parameters required to uniquely identify a specific orbit . Some of these elements semi major axis (a) , Eccentricity (e) , Inclination (i) , Longitude of the ascending node(Ω) study with the important perturbation as atmospheric drag, solar radiation pressure and Moon attraction for different high (hp) or different inclination .The eccentricity are used as: (e=0.012) with hp=95.8 Km. and i=63 degree, (e=0.0015) with hp= 601.8 Km. and i=85 degree or i=63 degree, (e=0.1) with hp=1264.2 Km. The results point to the fact which is the importance perturbation depend on high of perigee for the satellite orbit at low high atmospheric drag is a dominate at increase high the lunar attraction and solar radiation pressure effects are increase , The elliptic orbit first becomes circular as the apogee altitude decreases to the same value of perigee and then rapidly spirals into the dense atmosphere . The summations of all perturbations acceleration components are get before solving the equation of motion , the eccentricity (e) change with time due to all perturbations is plotted the secular decreases at low orbits but (e) is stabile at hp=601Km., and (e) is secular increases at medium and high orbits. where miens the S.R.P. and lunar attraction are more effects from atmospheric drag. Our results are comparison with some other works it is quite agreements.

1-Introduction and Orbit Geometry:

The orbit can be described by the lengths of the semi major axis (a) and the semi minor axis (b). Orbital elements are the parameters required to uniquely identify a specific orbit. In celestial mechanics these elements are generally considered in classical two-body systems, where a Kepler's laws, Newton's laws of motion and Newton's law of universal gravitation are used. There are many different ways to mathematically describe the same orbit, but certain schemes each consisting of a set of six parameters are commonly used in astronomy and orbital mechanics [1].

The main elements Eccentricity (e), Semi major axis (a) that define the shape and size of the ellipse orbit [1].

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the perigee to apogee distances divided by two point to Semi major axis. The elements define the orientation of the orbital plane are the inclination (i) is vertical tilt of the ellipse with respect to the reference plane, measured at the ascending node , Longitude of the ascending node(Ω) is horizontally orients the node of the ellipse . The other two elements are Argument of perigee (w) is defines the orientation of the ellipse in the orbital plane, as an angle measured from the ascending node to the semi major axis and mean anomaly at epoch defines the position of the orbiting body along the ellipse at a specific time [1,2].

The angles of inclination, longitude of the ascending node, and argument of perigee can also be described as the Euler angles defining the orientation of the orbit relative to the reference coordinate system. Satellite orbit perturbations come from the non-spherical Earth with non-homogeneous distribution of mass, atmospheric drag force and the existence of other celestial bodies. With the effect of perturbations, a satellite orbit will not simply be the two-body motion. The magnitude of disturbance varies with the altitude at LEO orbit the atmospheric drag are dominant, but at HEO orbit atmospheric drag could be ignored because of the dramatic drop of the atmospheric density in outer space. Third body perturbations due the Moon increase with high and it's more than the perturbations due the sun. In this research the atmospheric drag , solar radiation pressure and lunar attraction are calculated and studied for satellite orbits at different high and inclination [3].

2-Theory :

The mean angular motion (n) in an elliptic orbit is defined as:

$$n = \sqrt{\frac{\mu}{a^3}} \qquad (1)$$

The period of the orbit (T) is defined as Kepler's third law:

$$T = \frac{2\pi}{\sqrt{\mu}} a^{\frac{3}{2}} = 2\pi / n \qquad (2)$$

The general equation of motion can be written by using Newton's laws in motion with general gravitational law given as[1,2,3]:

$$\ddot{\vec{r}}_i + \frac{\mu}{r^3}\vec{r} = ap \qquad (3)$$

ap : is the perturbation acceleration. The orbits described by this equation are conic sections, having a constant energy (En) and angular momentum vector (h) that can be expressed as [2]

$$En = \frac{v^2}{2} - \frac{\mu}{r} \qquad (4)$$

The radial distance from the central body to the satellite can be related to the true anomaly (f) by [4] :

$$r = \frac{p}{1 + e\cos f} \qquad (5)$$

The true anomaly (f) has a convenient direct geometric interpretation, it is not always mathematically convenient to express the satellite location through this angle.

The mean anomaly (M) is typically used in describing the location of a satellite in an orbit. it does vary linearly with time according to the equation:

$$M = M_0 + n(t - t_0) \qquad (6)$$

Where M_0 is the mean anomaly at time t_0, and n is the mean angular motion of the satellite equation (1) .The eccentric anomaly can be get it as in the following equation [2,3,4,5]

$$M = E - e\sin E \qquad (7)$$

One of the analytical methods is used to solve Kepler's equation by iterative methods only **[5]** . A common way is to start with an approximation of $E_0 = M$ or and employ Newton's method to calculate successive refinements E_i until the result changes by less than a specified amount from one iteration to the next [5].

$$f(E) = E - e\sin E - M \qquad (9)$$

The solution of Kepler's equation is equivalent to finding the root of $f(E)$ for a given value of M . Applying Newton's method for this purpose, an approximate root E_i of f may be improved by computing

$$E_{i+1} = E_i - \frac{f(E_i)}{f'(E_i)} = E_i - \frac{E_i - e\sin(E_i) - M}{1 - e\cos(E_i)} \qquad (10)$$

And return equation (7) ,for $i =$ any value when $\Delta E_i < \varepsilon$ $(\varepsilon \approx 0)$, when ΔE_i give as follows:

$$\Delta E_i = \frac{f(E_i)}{f'(E_i)} \quad \ldots\ldots\ldots(11)$$

the velocity coordinates obtained from derivation the position coordinates as[4,6]:

$$r_w = \frac{\sqrt{\mu a}}{r} e\sin E \qquad (12)$$

$$\dot{x}_w = -\frac{\sqrt{\mu a}}{r}\sin E \qquad (13)$$

$$\dot{y}_w = \frac{\sqrt{(1-e^2)\mu a}}{r}\cos E \qquad (14)$$

$$\tan\frac{f}{2} = \sqrt{\frac{1+e}{1-e}}\tan\frac{E}{2} \qquad (15)$$

$$\vec{h} = \vec{r}\times\vec{v} \quad \text{or} \quad h = r.v\cos\phi$$

where v is the velocity of the object.

and ϕ is the angle between a line drawn perpendicular to the position and velocity vectors. The momentum vector $\vec{h}$ can be analysis to components, where

$$\vec{h} = \begin{bmatrix} h_x \\ h_y \\ h_z \end{bmatrix} = \begin{bmatrix} i & j & k \\ x & y & z \\ \dot{x} & \dot{y} & \dot{z} \end{bmatrix} \qquad (16)$$

Euler angles (i,Ω,ω) used in Gauss –Matrix where used to orientation the coordinates from satellite orbit plane to equatorial plane **[1,3]**.

$$\begin{pmatrix} \sin i \sin\Omega \\ \sin i \cos\Omega \\ \cos i \end{pmatrix} = \begin{pmatrix} +h_x/h \\ -h_y/h \\ +h_z/h \end{pmatrix} = \begin{pmatrix} +W_x \\ -W_y \\ +W_z \end{pmatrix} \qquad (17)$$

To Calculation of orbital elements from position and velocity Coordinates there is exactly one set of orbital elements that corresponds to given initial values of r and v. semi major axis a and the eccentricity (e) were calculated as [2]:

$$a = \left(\frac{2}{r} - \frac{v^2}{\mu}\right)^{-1} \qquad (18)$$

$$e = \sqrt{1 - \frac{h^2}{\mu a}} \qquad (19)$$

The inclination and right ascension of the ascending node are given by :-

$$i = \arctan\left(\frac{\sqrt{W_x^2 + W_y^2}}{W_z}\right) \qquad (20)$$

$$\Omega = \arctan\left(\frac{W_x}{-W_y}\right) \qquad 21)$$

3- Perturbation due to Atmospheric drag

Drag is the resistance of the atmosphere to the satellite. Drag force acts in a direction opposite to the direction of the satellites motion, this drag greatest during launch and reentry [**7**]. The acceleration of the satellite due to atmospheric drag can be expressed as **[2, 8]:**

adrag =- 0.5 ρ CD A v^2 iv / m (22)

where The negative sign indicates that the acceleration is in the opposite direction of the velocity vector, **iv** unit vector indicate the direction of movement, ρ is the atmospheric density, *CD* is the drag coefficient, A is the cross sectional area of the satellite perpendicular to the velocity vector, m is the mass of the satellite ,v is the velocity of the satellite relative to the atmosphere .

The density ρ of the upper atmosphere can be modeled by a simple analytical equation (form [2]) when some assumptions are made:

The of the Earth is spherically symmetric;• the scale height is constant over the altitudes of interest; and there is no time variation in the density. As :

ρ = ρ po exp[- (**h** – ***h***po) / H] (23)

Where **ρ** po is the atmospheric density at the initial perigee point, **h** and ***h***po are the altitudes of the evaluation point and the initial perigee point respectively and *H* is the constant scale height.

The following mean values were taken from [2] for an altitude of 450 km:

ρ po =1.585x10^-12 kg/m3 and *H*=62.2 km. The density equal zero at h > 1000Km, so that the magnitude of the drag neglected (or equal zero) at h > 1000Km. H is the scale height (The scale height is that vertical distance in which the density change by a factor exponential and depends upon the altitude, H the height, equations given by **[8, 9** , 10 **]**

The drag coefficient (***CD***) is not as trivial to evaluate as it may seem. Since the density is very low at the altitudes of satellite orbits, The mean value for

CD of 2.2 can be taken with an error (standard deviation) 5 % . In evaluating the velocity of the satellite relative to the atmosphere, the movement of the atmosphere relative to the Earth will be ignored. When the orbit perigee height is below 1000 Km, the atmospheric drag effect can be neglected. other perturbation forces, is a non-conservative force and will continuously take energy away from the orbit. Thus, the orbit semi-major axis and the period are gradually decreasing because of the effect of drag. [**3 , 11**]. The drag force is given by [**2,12**]

4- Solar radiation pressure (SRP) perturbation*:*

The luminosity of the Sun is about 3.86 x 10^{26} watts. This is the total power radiated out into space by the Sun. Most of this radiation is in the visible and infrared part of the electromagnetic spectrum, with less than 1 % emitted in the radio, UV and X-ray spectral bands. The sun's energy is radiated uniformly in all directions. Because the Sun is about 150 million kilometers from the Earth. This still amounts to a massive 1.75 x 10^{17} watts. The amount of power in sunlight passing through a single square meter face to the sun, at the Earth's distance from the Sun. The power of the sun at the earth, per square meter is called the solar constant and is approximately 1370 (W m^{-2}) [2]. Solar radiation pressure is a force on the satellite due to the momentum flux from the sun. For most satellites it acts in a direction radials away from the sun. The magnitude of the resulting acceleration on the satellite due to SRP is given by [7,13] :

asrp **= K P As / m** **(24)**

where *K is* a dimensionless constant between *1 and 2 (K=l: surface perfectly absorbent ;K =2:* surface reflects all light), P is the momentum flux from the sun, As is the cross-sectional area of the satellite perpendicular to the sun-line and m is the mass of the satellite. The value of K is taken as 1.5 for the purpose of this study[12].

The mean value of P is approximately 4.56x10-6 N/m^2 at the distance of the Earth from the Sun is later simulations[3], and corrected with time where the Earth-Sun distance variation through the year . Some times SRP get zero when the satellite pass in the shadow of the Earth, the cross-section area perpendicular to the sun-line was always taken as the maximum cross-sectional area to account .

5- Lunar Attraction Perturbation:

The term *third body* refers to any other body in space besides the Earth which could have a gravitational influence on the satellite. The most significant influences for the LEO satellite come from the sun and the moon.

Planetary gravitational influences are orders of magnitude smaller than these and will be ignored .

The perturbing acceleration due to the gravitational attraction of a third body can be calculated as follows [7, 13]:

ad = - µd (rs + f(q) rd) **(25)**

where µd : is the gravitational parameter of the third body and the definitions of the vectors are given in the following ,The functions f and q are :

$$q = \frac{\mathbf{r}_s \cdot (\mathbf{r}_s - 2\mathbf{r}_d)}{\mathbf{r}_d \cdot \mathbf{r}_d}$$

$$f(q) = q\frac{3+3q+q^{2}}{\quad ^{3}}$$

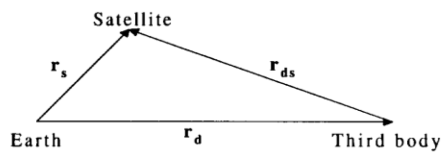

Fig. 1: Vector definitions for third body attractions.

erturbations i... n of the sun and the moon. The sun is modeled as an Earth satellite in a circular orbit with a radius of one astronomical unit and a period of 365.26 days. The inclination of the orbit is 23.439°. The moon is modeled as an Earth satellite in an orbit with an eccentricity of 0.056, semi-major axis of 384000 km .In this work the vector (rd) components were calculated at all days and the others components were calculated at all period of satellites. the effects of the Moon will be treated as a third body acting on the satellite. Although the mass of the Moon is much lower than that of the Sun the reduced distance between perturbing body and satellite makes the Lunar perturbation about equal to the Solar. [14]

$$a_l = \frac{\mu_l r}{r_l^3}\left[3(i_r \cdot i_l)i_l - i_r\right] \quad (26)$$

Where $\mu_l = GM_l = 4.902794 * 10^3 (Km^3/\sec^2)$ i_l The distance between Moon and centre of Earth vector i_r The distance vector between Satellite and centre of Earth .Due to the strong solar and terrestrial perturbations, a larger number of terms are, however, needed to describe the lunar motion in terms of the mean arguments of the lunar and solar orbit. The mean longitude $L_\circ$ of the Moon , The Moon's mean anomaly l, The Sun's mean anomaly l'. The mean angular distance of the Moon from the ascending node F ,and The difference angel D between the mean longitudes of the Sun and the Moon. All are calculated as the following, The longitude of the ascending node Ω is not explicitly employed .It is obtained from the difference

$\Omega = L\circ - F$. $L\circ = 218^\circ.31617 + 481267^\circ.88088.Tu - 1^\circ.3972.Tu$ (27)

$$l = 134^\circ.96292 + 477198^\circ.86753.Tu \quad (28)$$

$$l' = 357^\circ.52543 + 35999^\circ.04944.Tu \quad (29)$$

$$F = 93^\circ.27283 + 483202^\circ.01873.Tu \quad (30)$$

$$D = 297^\circ.85027 + 445267^\circ.11135.Tu \quad (31)$$

the moon ecliptic coordinates λ ,β which are function to Julian century and Using these values to calculate the Moon’s longitude with respect to the equinox and ecliptic of the year 2000, Where the leading term is due to the inclination of Moon's orbit relative to the ecliptic, which amounts to approximately 5.1° . Finally the Moon's distance from the center of the Earth is:

$$\begin{aligned} r_l = [&385000 - 20905.\cos(l) - 3699.\cos(2D - l) \\ &- 2956.\cos(2D) - 570.\cos(2l) + 246.\cos(2l - 2D) \\ &- 205.\cos(l' - 2D) - 171.\cos(l + 2D) \\ &- 152.\cos(l + l' - 2D)] \quad Km \end{aligned} \quad (32)$$

The spherical ecliptic coordinate may again be converted to equatorial Cartesian coordinates [4].

6- Calculation of the relative velocity of satellite

For some satellites, we cannot assume that the mass is constant. Recognize the v_{rel} is not the velocity vector typically found in the sate vector. This velocity vector is relative to the atmosphere.

Actually, the Earth's atmosphere has a mean motion due to the rotation, and the local winds are superimposed on this mean motion. Notice also that the force of drag in opposite to the velocity vector always . We usually call $\frac{m}{(C_D A)}$ the ballistic coefficient. The velocity vector relative to the rotating atmosphere is given by**[1, 15]**:

$$v_{rel} = v + r \times \omega_{Earth} = \begin{bmatrix} v_x + \omega_{Eart} y \\ v_y - \omega_{Eart} x \\ v_z \end{bmatrix} \quad (33)$$

Where ω_{Earth} represent the Earth's rotational velocity, and assumed to be constant equal

$$\omega_{Earth} = \frac{360 \deg}{MSD} = \frac{1}{86164} \ rad/\sec = 7.29115 \times 10^{-5} \ rad/\sec$$

Where MSD represent the Mean Solar Day, which equal to 23hr; 56min; 4sec.

v represent the orbital velocity vector where the components give as:

$$v_{rx} = v_x + \omega_{Earth}\ y\ ,\ v_{ry} = v_y - \omega_{Earth}\ x\ ,\ v_{rz} = v_z$$

$$v_{rel}(mag) = \sqrt{v_{rx}^{\ 2} + v_{ry}^{\ 2} + v_{rz}^{\ 2}} \qquad (34)$$

7- Results and Discussion:

The programs wanted are design to calculate the perturbations acceleration components ,which are used with the equation of motion and solved to get the components of position, velocity, momentum and finally the orbital elements variation . The eccentricity are used as: (e=0.012) with hp=95.8 Km. and i=63 degree, (e=0.0015) with hp= 601.8 Km. and i=85 degree, (e=0.1) with hp=1264.8 Km. and i=63 degree. These values are used to comparable with other works. The orbital elements(a ,e ,i ,Ω) are calculated and plotted with time included one of the three perturbations (atmosphere drag, solar radiation pressure, lunar attraction.) and with all these perturbations .the above elements have more effect on the satellite life time.

The results plot in the figures (1 to 22) for different highs and inclinations Since the drag is greatest at perigee, where the velocity and atmospheric density are greatest, the energy drain is also greatest at this point. Under this dominant negative impulse at perigee, the orbit will become more circular in each revolution. The elliptic orbit first becomes circular as the apogee altitude decreases to the same value of perigee and then rapidly spirals into the dense atmosphere, the meaning the eccentricity decreases for L.E.O. as in figs (14, 20) .

For near-circular orbits ,Therefore, aerodynamic design of satellite structure should be taken into account A/m ratio. for low earth Orbits such that the ratio should be minimized for the outer atmosphere. The results are discuses as the following:

1- Fig (1- 4) shows that the semi-major axis (a) is secular decreases with time due to atmospheric drag and S.R.P. at different highs or inclinations but (a) is increase (150 m.) through 20 periods after that (a) is stabile. Due to lunar attraction effect is appear at hp>=1000 Km. as in fig(5).
2- Fig (6,7) shows that the inclination (i) change with time due to atmospheric drag , it's rapidly decrease at hp=95.8 Km. stabile at hp>=600Km. fig(8 ,9) shows that the inclination (i) change with time due to S.R.P. the inclination is small increases at i=63 deg. And stabile at i=95 deg. that miens the S.R.P. is more effect on the equatorial orbits.
3- Fig (10-11) shows that the right ascension of ascending node (Ω) change with time due to atmospheric drag , it's increase at hp=95.8 Km., stabile at hp >=600Km.
4- fig (12,13) shows that Ω variation due to S.R.P. it's small increases at hp= 1264.8 Km. and small decrease at hp=601 Km.
5- Fig (14-16) shows that the eccentricity (e) change with time due to atmospheric drag , it's rapidly decrease rapidly at hp=95.8 Km. and it's more stabile at hp>=600Km. fig(15,16) . fig (17,18) shows that (e) change with time due to S.R.P. the inclination is increases at hp=601,1264 Km. where Δe= 0.002 through 50 periods for all two highs .fig (19) shows the effect of lunar attraction on (e) ,it was appeared stable with time at hp=1264Km.

Figs (20,21,22) are plotted after difficult work that are summations all perturbations acceleration components before solving the equation of motion , the eccentricity (e) change with time due to all perturbations is plotted in fig(20) at hp=95.8Km. is secular decreases that miens the atmospheric drag is the dominates factor but (e)is stabile at hp=601Km.as in fig (21) where balance effect ,(e) is secular increases at hp=1264.8Km. where miens the S.R.P. and lunar attraction are more effects. Our results are comparison with [6,7,14,15,20,21] it is quite agreements

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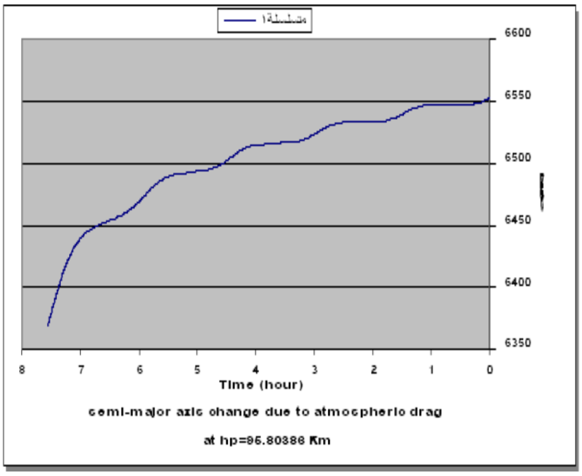


Fig(1): semi major axis variation due to atmospheric drag at hp=95.8 Km. i=63

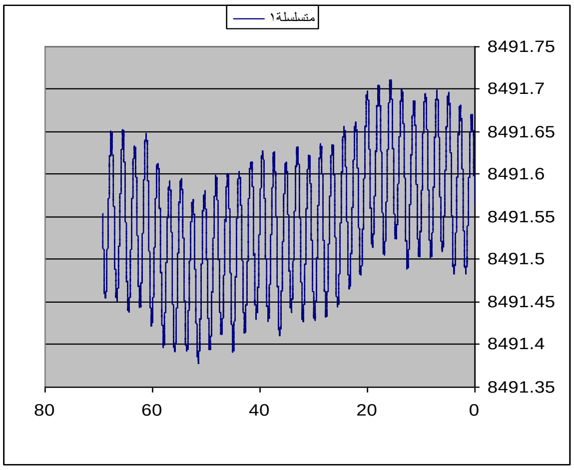
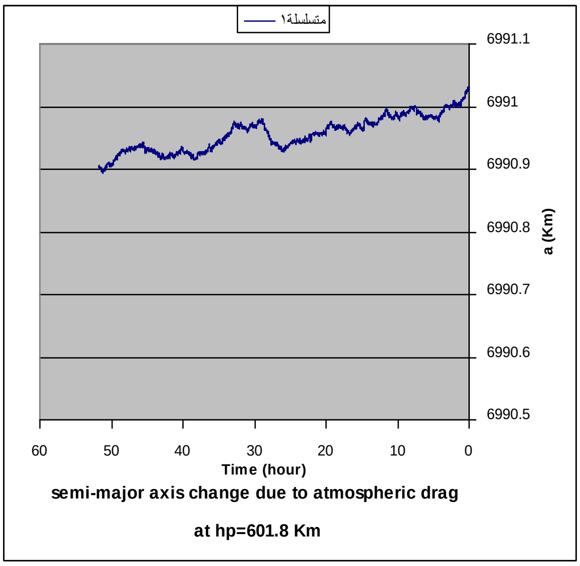


Fig (4):Semi-major axis change due to SRP at hp=1264.2 Km and i=63º.



Fig(1): semi major axis variation due to atmospheric drag at hp=601.8 Km. i=63

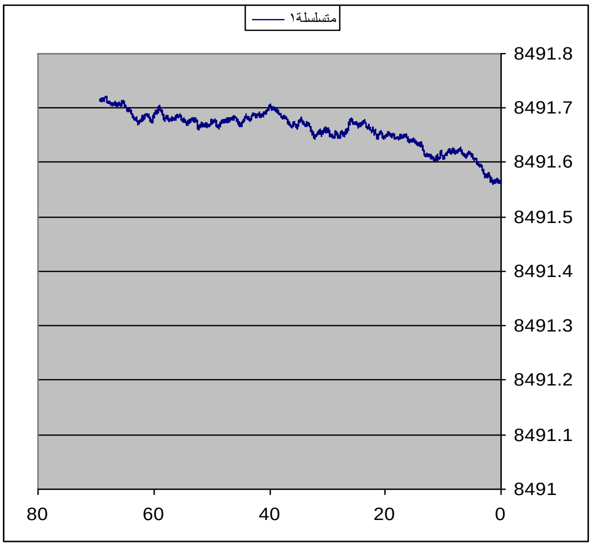
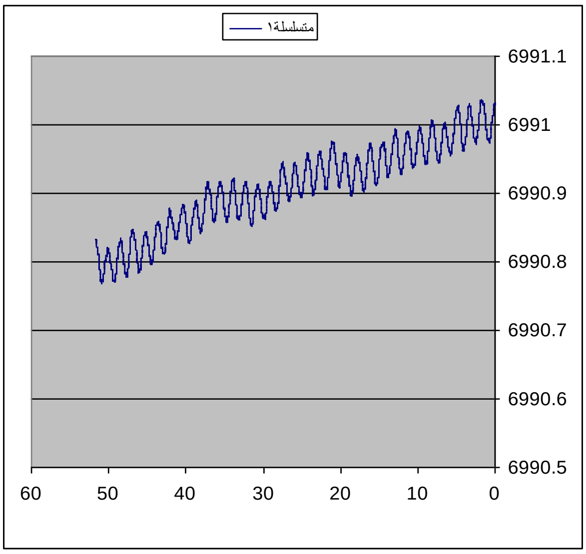


Fig (5) :Semi-major axis change due to lunar attraction at hp=1264.2 Km ,i=63 º.



Fig(3):Semi-major axis change due to SRP at hp=601.8Km and i=85º

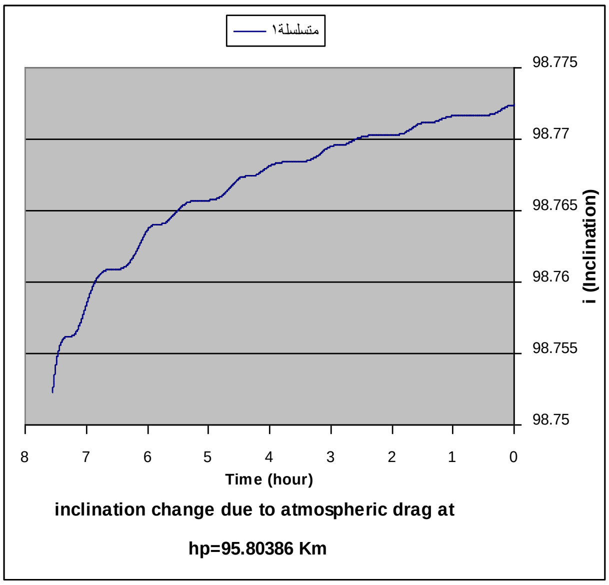


Fig (6) :Inclination change due to lunar attraction at hp=95.8 Km.

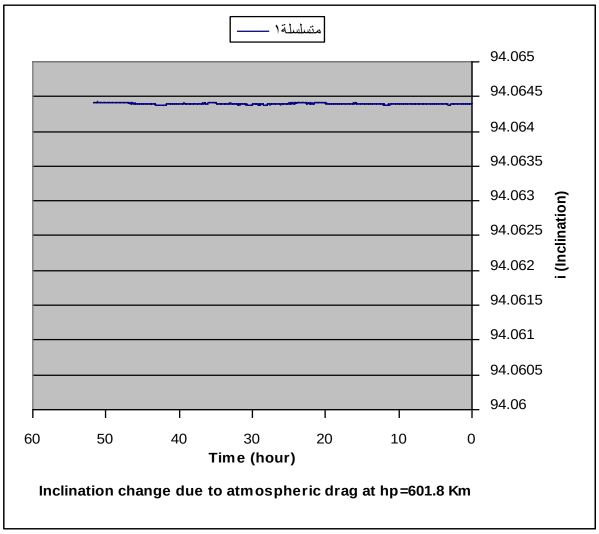
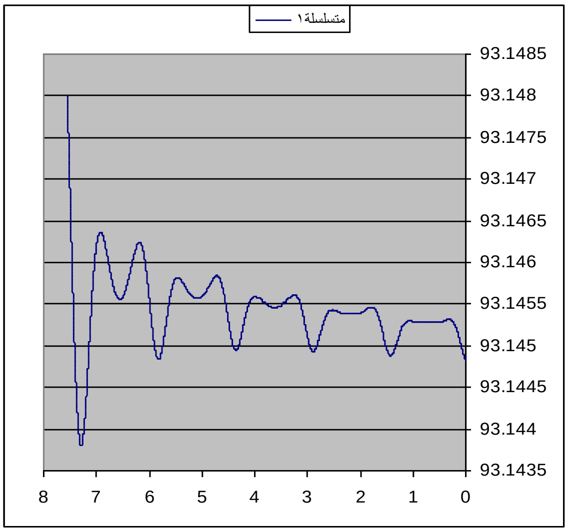
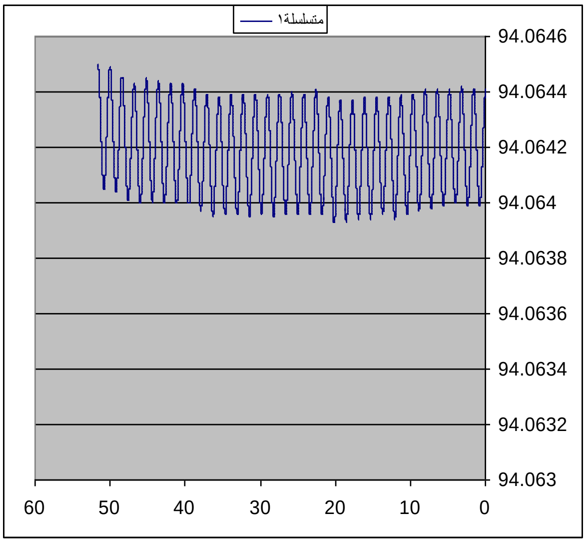


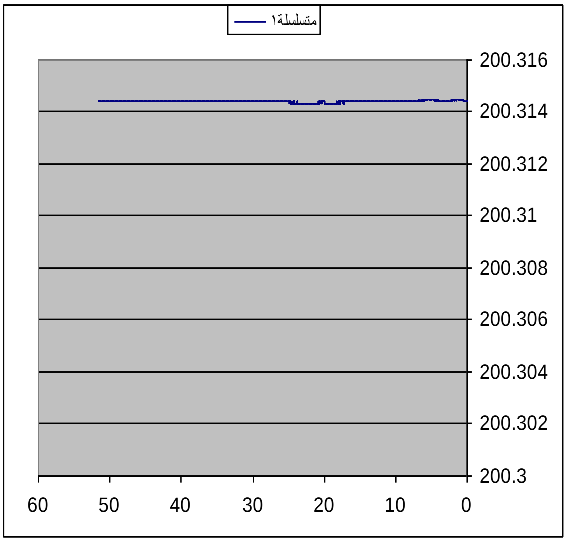
Fig (7) :Inclination change due to lunar attraction at hp=1264.2 Km.



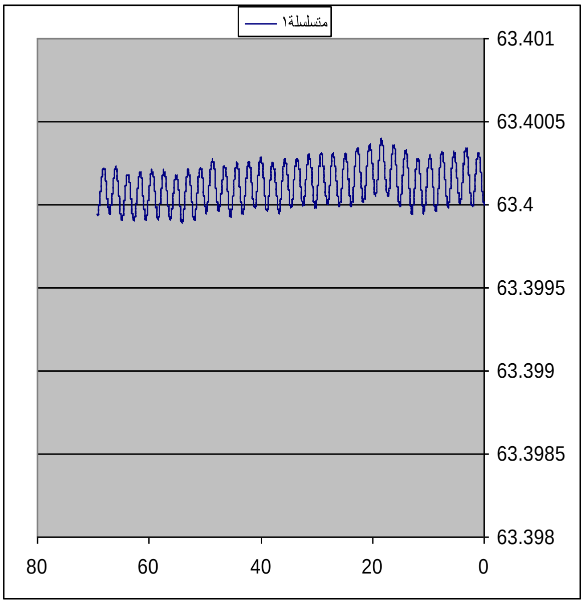
Fig(10): Ω change due to atmospheric drag at hp=95.8 Km and i=63º



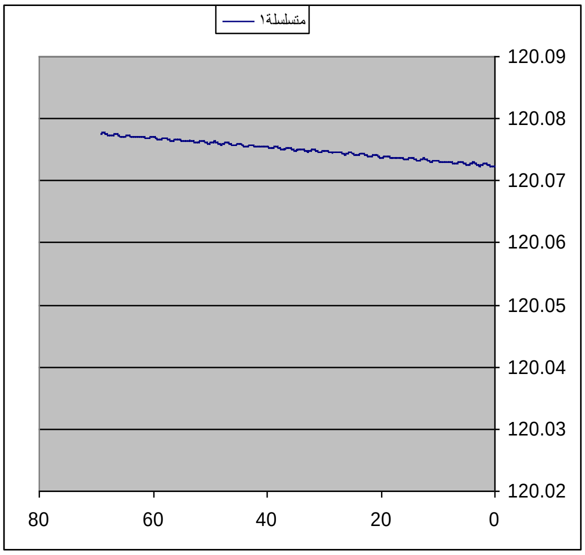
Fig(8): Inclination change due SRP at hp=601.8 Km .



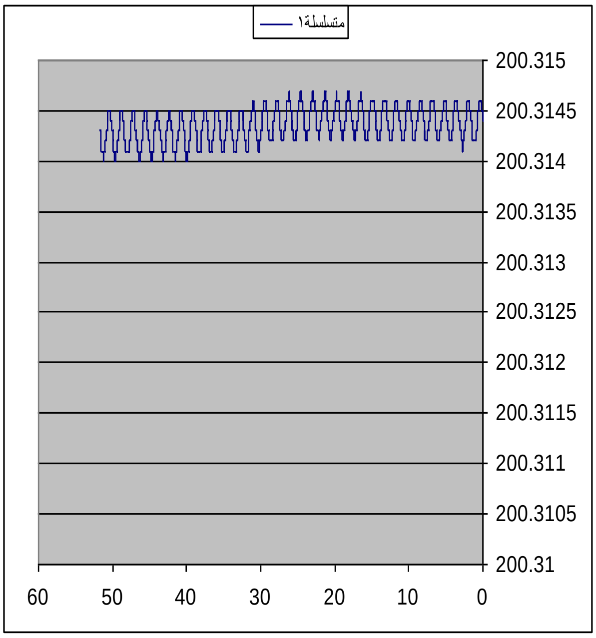
Fig(11): Ω change due to atmospheric drag at hp=601.8 Km, i=85º



Fig(9): Inclination change due SRP at hp=1264.2 Km.



Fig(12): Ω change due SRP at hp=1264.2 Km , i=63º



Fig(13): Ω change due SRP at hp=601.8 Km ., i=85°

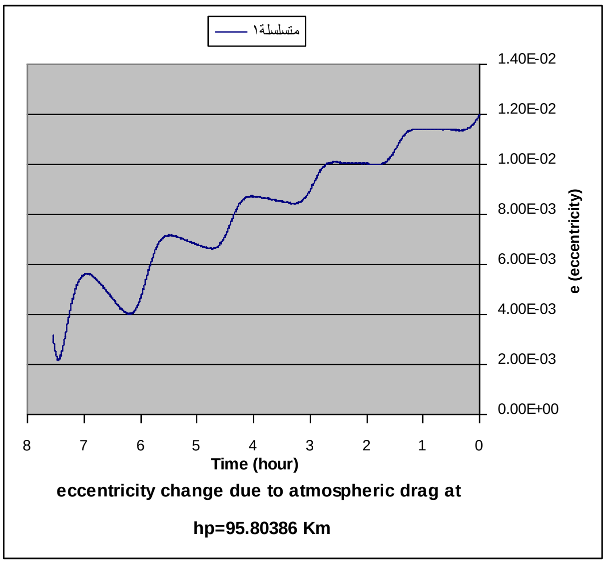
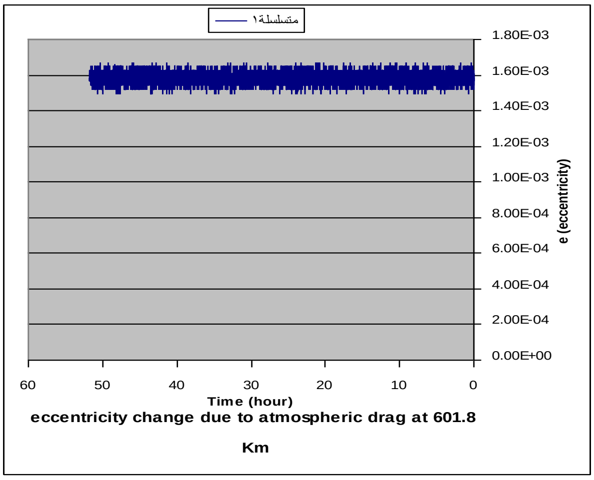
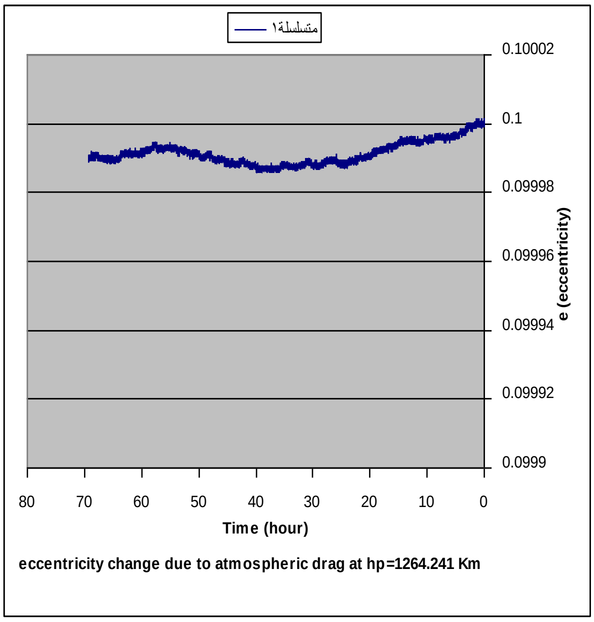


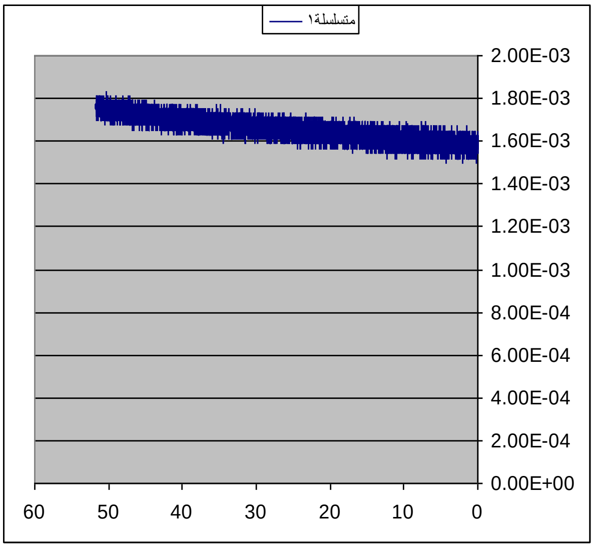
Fig (14): eccentricity change with atmospheric drag at hp=95.8km, i=63 °



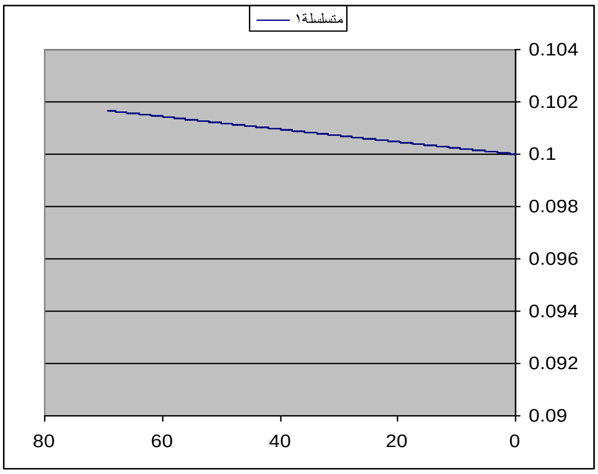
Fig(15) : eccentricity change with atmospheric drag at hp=601.8 km, i=85 °



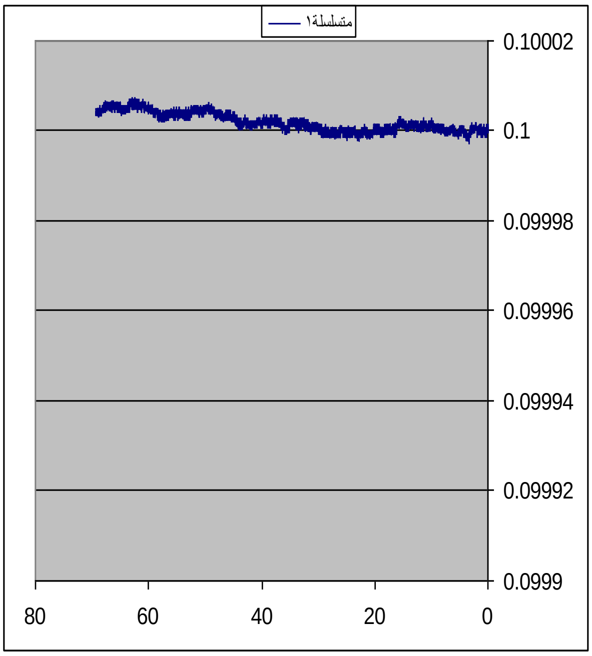
Fig(16): eccentricity change with atmospheric drag at hp=1264.2km, i=63°



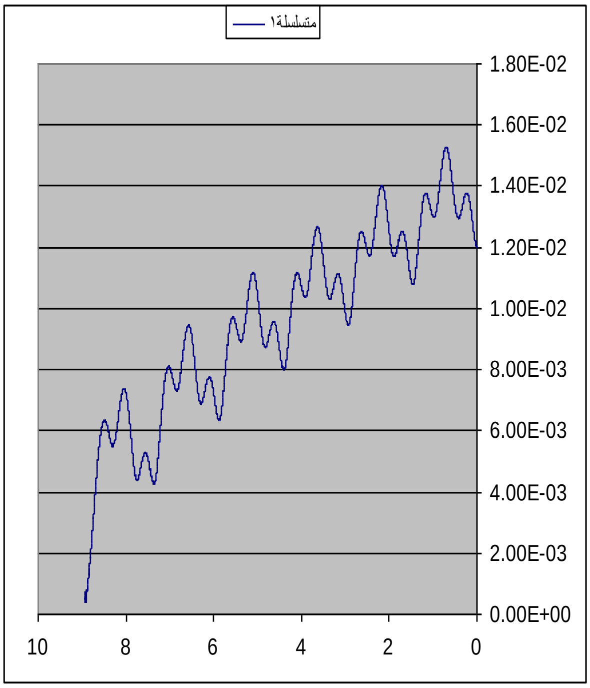
Fig(17): Eccentricity change with SRP at hp=601.8 Km ,i=63°



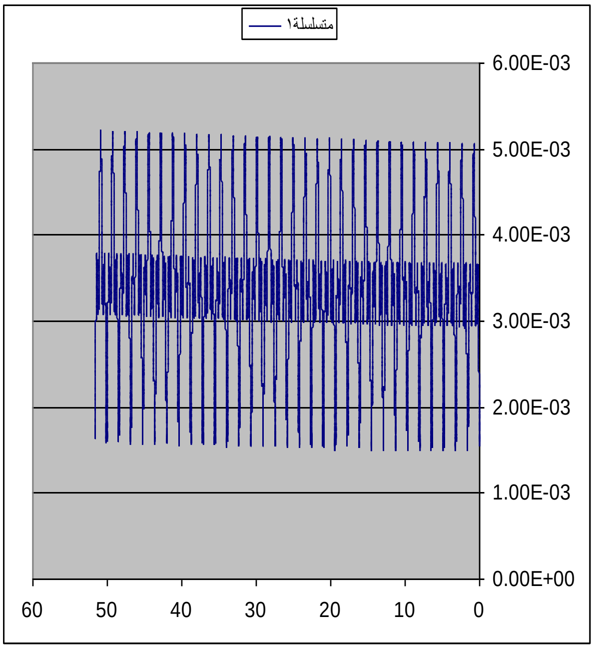
fig(18): Eccentricity change with SRP at hp=1264.2 Km , i=63°.



Fig(19): Eccentricity change due to lunar attraction at hp=1264.2Km ,i=63º.



Fig(20): Eccentricity change due to all perturbations at hp=95.8 Km. ,i=63 º



Fig(21): Eccentricity change due to all perturbations at hp= 601.8 Km. ,i=85º

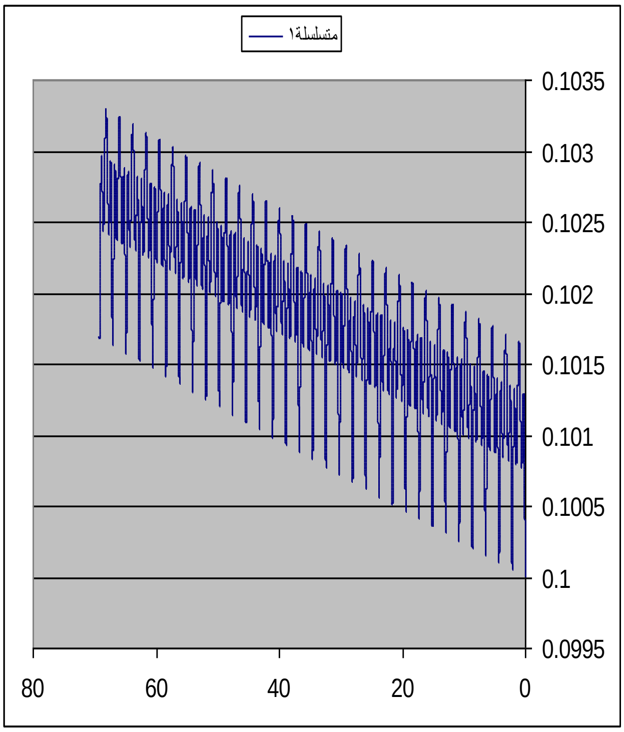


Fig (22): Eccentricity change due to all perturbations at hp=1264.2 Km , i=63º.

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تغير مدارات الأقمار الاصطناعية الواطئة بسبب كبح الغلاف الجوي وضغط الاشعاع الشمسي وجذب القمر

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الخلاصة:

العناصر المدارية هي ضرورية لمعرفة شكل وحجم مدار القمر الصناعي ، وأهم العناصر هي نصف المحور الكبير ، واللامركزية ، والميل ، وبعد العقدة الصاعدة عن الاتجاه المرجعي التي حسبت ودرست بوجود الاضطرابات كبح الغلاف الجوي للأرض وضغط الإشعاع الشمسي وجذب القمر، لارتفاعات وميول متعددة هي: (e=0.012) مع hp=95.8 Km. ، (e=0.0015) ، i=63 مع hp= 601.8 Km. ، (e=0.1), i=85 مع hp=1264.2 Km.، i=63,85 ، النتائج تشير إلى شيء مهم هو أن أهمية الاضطرابات تعتمد على ارتفاع الحضيض ، فلمدارات الأقمار الصناعية الواطئة يكون كبح الغلاف الجوي هو المهيمن وبزيادة الارتفاع يزداد تأثير جذب القمر وضغط الإشعاع الشمسي ، ومدار القطع الناقص يتحول إلى الشكل الدائري بنقصان الأوج وثبات الحضيض وينهار المدار الواطئ بسرعة بزيادة كثافة الهواء أي يقل كل من اللامركزية ونصف المحور الكبير. أما المدارات القطع الناقص العالية فتنتهي بسبب الاضطرابات بزيادة اللامركزية ونصف المحور الكبير فيخرج القمر الصناعي من جاذبية الأرض، تم جمع مركبات التعجيل بوجود جميع الاضطرابات الثلاثة أعلاه قبل حل معادلة الاضطراب ثم حسبت اللامركزية عند الارتفاعات الثلاثة فوجدت أنها تقل عند الارتفاعات الواطئة وتستقر عند الارتفاع 601 كم. وتزداد عند الارتفاعات المتوسطة والعالية حيث يزداد تأثير جذب القمر وضغط الإشعاع الشمسي. تمت مقارنة نتائجنا مع نتائج آخرين فوجدنا توافقا جيدا