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ORBIT DESIGN AND SIMULATION FOR KUFASAT NANO-SATELLITE

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ABSTRACT. Orbit design for KufaSat Nano-satellites is presented. Polar orbit is selected for the KufaSat mission. The orbit was designed with an Inclination which enables the satellite to see every part of the earth. KufaSat has a payload for imaging purposes which require a large amount of power, so the orbit is determined to be sun synchronous in order to provide the power through solar panels. The KufaSat mission is designed for the low earth orbit. The six initial Keplerian Elements of KufaSat are calculated. The orbit design of KufaSat according to the calculated Keplerian elements has been simulated and analyzed by using MATLAB first and then by using General Mission Analysis Tool.
Keywords: Orbit design, Orbit simulation, General Mission Analysis Tool, Nanosatellite, KufaSat.

1. INTRODUCTION

When designing an orbit for a satellite mission, the biggest consideration is the mission requirements, In case of the missions which require a full Earth coverage and can include earth surface imaging, and the measurement of physical parameters such as the gravity field and the radiation environment, the best orbits will have low altitudes and near-polar inclinations (Paskowitz and Scheeres, 2006). Orbit selection can vary in terms of altitude and their orientation and rotation relative to the Earth. The circular orbits are predicted by matching the centripetal acceleration to the gravitational force.

This paper presents the orbit design and simulation of KufaSat Nano-Satellite. KufaSat is the Iraqi student satellite project sponsored by the University of Kufa and it will be the first Iraqi satellite to fly in space. The main tasks for KufaSat will be to imaging purposes and remote sensing. It is a Nano-satellite based on the CubeSat concept.

The General Mission Analysis Tool (GMAT) is used in this paper to design and analysis KufaSat orbit. GMAT is a space trajectory optimization and mission analysis system developed by NASA. It is designed to model, optimize, and estimate spacecraft trajectories in flight regimes ranging from low Earth orbit to lunar applications, interplanetary trajectories, and other deep space missions. Analysts model space missions in GMAT by first creating resources such as spacecraft, propagators, estimators, and optimizers.

Resources can be configured to meet the needs of specific applications and missions. GMAT contains an extensive set of available Resources that can be broken down into physical model Resources and analysis model Resources. Physical Resources include spacecraft, thruster, tank, ground station, formation, impulsive burn, finite burn, planet, comet, asteroid, moon, and barycenter. Analysis model Resources includes differential corrector, propagator, optimizer, estimator, 3-D graphic, x-y plot, report file, ephemeris file. GMAT contains new technology and is a test-bed for future technology development (GMAT Documentation). To satisfy NASA's mandate and maximize technology transfer, GMAT is an open source software system licensed under the NASA Open Source Agreement (http://www.opensource.org/licenses/nasa1.3.php).

2. ORBITAL MECHANICS

Orbital mechanics are the study of the motions of artificial satellites and space vehicles moving under the influence of forces such as gravity, atmospheric drag, thrust, and so on (Barnes-Svarney, 2005). Johannes Kepler developed the first laws of planetary motion to predict the motion of the planets about the sun or the path of a satellite about the earth, and his theories were confirmed when Isaac Newton revealed his universal law of gravitation. These laws provide a good approximation of the path of a body in space mechanics.

A. Orbital elements

There are six Classical Orbital Elements that are necessary for us to know about an orbit and a satellite's place in it. These elements help us describe: Orbit size, Orbit shape, Orbit orientation, and Orbit location. They also specify the part of the Earth the satellite is passing over at any given time and Field of View (FOV) which is the angle that describes the amount of the Earth's surface the spacecraft can see at any given time. These six orbital elements shown in Figure 1 are (Sally Ride EarthKAM):

Semi-major Axis (a): Describe the size of the orbit, which is one-half the distance across the major axis of the orbit.

Eccentricity (e): Specifies the shape of an orbit and is given by the ratio of the distance between the two foci and the length of the major axis. The eccentricity of a circular orbit is zero, and for an ellipse it can range from zero to less than one.

Inclination (i): Angle between the plane of the equator and the orbital plane.
Right Ascension of the Ascending Node (Ω): It is the angle between the Sun and the intersection of the equatorial plane and the orbit on the first day of spring in the Northern Hemisphere. The day is called the vernal equinox. Looking down from above the North Pole, the right ascension of the ascending node is positive counter-clockwise.
Argument of Perigee (ω): Angle between the ascending node and the orbit's point of closest approach to the Earth (perigee).

True Anomaly (v): True Anomaly is one of three angular parameters ("anomalies") that define a position along an orbit, the other two being the eccentric anomaly and the mean anomaly. True Anomaly represents the angle between the perigee and the vehicle in the orbit plane.

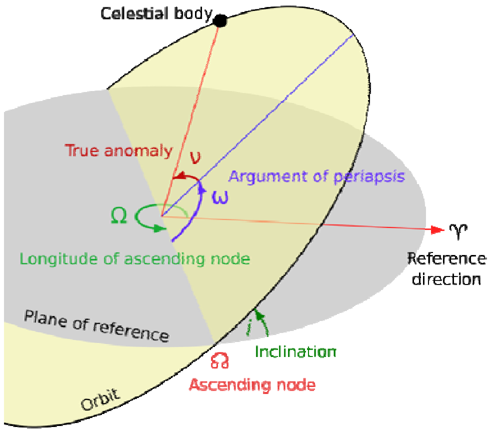

Fig. 1. Graphical Representation of Keplerian elements (Wikipedia)

B. Orbit perturbations

A satellite's orbit in an ideal two-body system describes a conic section, or ellipse. In reality, there are several factors that cause the conic section to continually change. These deviations from the ideal Kepler's orbit are called perturbations.

Perturbations of the orbit are the results of various forces which are acting on a satellite that perturb it away from the nominal orbit .These perturbations, or variations in the orbital elements, can be classified based on how they affect the Keplerian elements. The principal sources of perturbations are Earth gravity harmonics (deviations from a perfect sphere), the lunisolar gravitational attractions, atmospheric drag, solar radiation pressure, and Earth tides. (Eshagh and Alamdari, 2007).

The general form of equations of motion, including perturbations, can be expressed as follows:

$$\frac{d^2\mathbf{r}}{dt^2} = -\frac{\mu}{r^3}\mathbf{r} + a_p \tag{1}$$

where r is the position vector of the satellite measured from the center of the primary body, μ is the gravitational constant of $k^2(m_1 + m_2)$, and a_p is the sum of all the perturbing accelerations and its magnitude for all the satellite orbits is at least 10 times smaller than the central force or two-body accelerations in the solar system. a_p may consist of the following types of perturbing accelerations. (Chobotov, 2002).

- Gravitational, when considering third-body (sun/moon) attractions and the non-spherical Earth
- Non-gravitational like atmospheric drag, solar-radiation pressure, outgassing (fuel tank leaks on the spacecraft), and tidal friction effect.

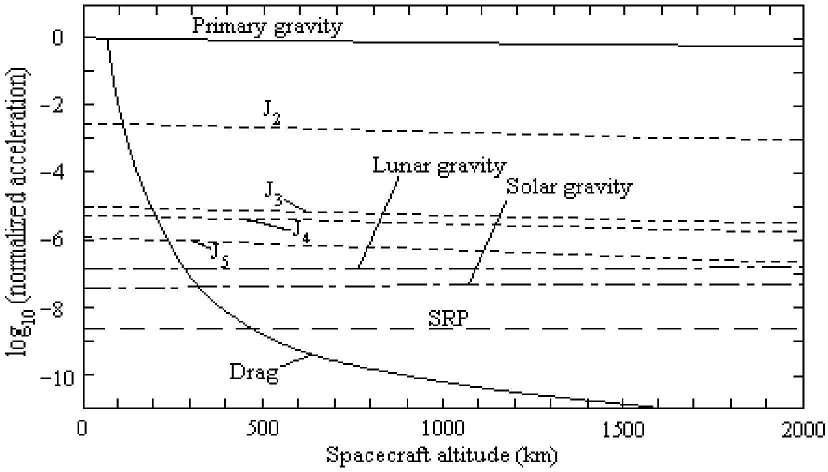

Fig. 2. Comparison of the perturbation accelerations for Earth orbits (Fortescue and Stark, 1995)

3. ORBIT DESIGN AND KEPLERIAN ELEMENTS OF KUFASAT

For the KufaSat mission a nearly circular orbit with Altitude of 600 km and inclination of 97 degrees is selected.

Using equation 2 which represents the Newton's form of Kepler's third law to calculate orbital period T:

$$T = \sqrt{\frac{4\pi^2 R^3}{GM}} \tag{2}$$

where R is the average radius of orbit for the satellite = R_{earth} + height, G is universal gravitational constant = 6.6726 x 10^{-11} Nm^2/kg^2. M_{earth} = 5.98x1024 kg (Mass of earth), R_{earth} = 6.37 x 10^6 m

T =5588 sec = 96.46 minutes or 15 orbits per day which translate to ground track speed of 7.566 km/sec (Chessab, 2013). However, only three orbits out of the fifteen will be in the ground station field of view

Keplerian elements shown in Figure 1 describe the shape and size of the orbit of satellite and they are the basics for development and implementation of orbit propagator. The six initial Keplerian Elements of KufaSat are calculated using the equations as shown below:

Semi major axis (a): Defines the size of the orbit. It is the distance between apogee and perigee divided by two.

$$\mathrm{a} = \sqrt[3]{\frac{P^2 GM}{4\pi^2}} \tag{3}$$

where P is the orbital period, G is the gravitational constant, M is the combined mass of primary and secondary bodies.

Eccentricity e: Defines the shape of the orbit. Its value ranges from 0 when the orbit is a perfect circle to 1 when the orbit is very flat

$$\vec{e} = \frac{\vec{v} \times \vec{h}}{\mu} - \frac{\vec{r}}{\mathrm{r}} \tag{4}$$

where v is orbital speed, $\vec{h}$ is angular momentum vector, μ is standard gravitational parameter, and r is orbit radius .

Inclination angle (i): Defines the orientation of the orbit with respect to the Earth's equator. It is the angle between $\vec{Z}$ and angular momentum vector, $\vec{h}$. The inclination ranges from 0 to 180 degrees.

$$\cos(i) = \frac{\hat{Z} \cdot \vec{h}}{|\hat{Z}||\vec{h}|} \tag{5}$$

Right Ascension of Ascending Node (Ω): The angle from the vernal equinox to the ascending node. The ascending node is the point where the satellite passes through the equatorial plane moving from south to north. Right ascension is measured as a right handed rotation about the pole $\vec{Z}$.

$$\cos(\Omega) = \frac{\hat{I} \cdot \vec{n}}{|\hat{I}||\vec{n}|} \tag{6}$$

If $n_j \le 0$ then $\Omega = 360 - \Omega$

Argument of Perigee (ω): The angle from the ascending node to the eccentricity vector, e⃗ $\vec{e}$, measured in the direction of the satellite's motion. The eccentricity vector points from the center of the Earth to perigee with a magnitude equal to the eccentricity of the orbit.

$$\cos(\omega) = \frac{\vec{n} \cdot \vec{e}}{|\vec{n}||\vec{e}|} \tag{7}$$

If $e_k \le 0$ then $\omega = 360 - \omega$

where:

$\vec{n}$ is a vector pointing towards the ascending node (i.e. the z-component of n is zero),

$\vec{e}$ is the eccentricity vector (a vector pointing towards the perigee).

True mean anomaly (M): The fraction of an orbit period which has elapsed since perigee, expressed as an angle. The mean anomaly equals the true anomaly for a circular orbit.

$$\cos(\nu) = \frac{\vec{e} \cdot \vec{r}}{|\vec{e}||\vec{r}|} \tag{8a}$$

If $\vec{r}\vec{v} < 0$ then $\nu = 360 - \nu$

$$\tan\left(\frac{E}{2}\right) = \sqrt{\frac{1-e}{1+e}} \tan\left(\frac{\nu}{2}\right) \tag{8b}$$

$$M = E - e\sin(E) \tag{8c}$$

where E is eccentric anomaly.

The Keplerian elements found using equations (3-8) for simple orbit design can be tabulated as shown in Table 1.

Table 1. Calculated Keplerian elements

Field	Value	Unit
Semi-Major Axis (SMA)	6978	km
Eccentricity (ECC)	0.00001715	
Inclination (INC)	97	deg
Argument of Perigee (AOP)	150	deg
Right Ascension of Ascending Node (RAAN)	0	deg
True Anomaly (TA)	10	deg

Low Earth orbiting satellites have physical lifetimes determined almost entirely by their interaction with the atmosphere. Prediction of such lifetimes depends upon knowledge of the initial satellite orbital parameters, satellite mass, cross-sectional area (in the direction of travel), drag coefficient, atmosphere model and solar flux. (Afful, 2013).

Using NASA Debris Assessment Software (DAS) to calculate orbit lifetime of KufaSat it is found that the orbit lifetime of KufaSat Nanosatellite is 17.435 years.

4. ORBIT SIMULATION AND ANALYSIS

The physical characteristics and on-board instruments of KufaSat are listed in Table 2 (Chessab et al., 2014).

Table 2. KufaSat characteristics

Spacecraft size	10 cm x 10 cm x10 cm
Spacecraft mass	1.3 kg
Total power	1.456 -3.323 W (2.4 W average)
Battery type, capacity	Lithium Polymer , 1.25 Ah
Bus voltage	3.3V , 5V , unregulated
Solar cells	Azur Space TJ (3G30C)+ Spectrolab UTJ TASC
Orbit	LEO, altitude of 600 km, inclination =97°, period=96.684minutes
Attitude Control System	3-axis stabilized control using 3 magnetic coils , 1.5m long gravity gradient boom
Attitude Determination System	3-axis magnetometer, six single axis sun sensors, and 3 gyroscopes.
Mission lifetime	3 years
Orbit lifetime	17.435 years
Payload	CMOS Camera
RF communications Downlink Uplink	S-band for camera image UHF-band for Health of KufaSat, Beacon VHF-band for Telecomand and Software Update

The orbit design of KufaSat according to the calculated Keplerian elements shown in Table 1 has been simulated using two methods, MATLAB, and General Mission Analysis Tool (GMAT).

A. Simulation using MATLAB:

The orbit design has been simulated in MATLAB using Keplerian elements shown in Table 1. MATLAB graphical user interface (GUI) is created to simulate a satellite orbiting the Earth in three dimensions. Figures 3, 4, 5, 6, 7 show KufaSat Orbit Simulation in MATLAB.

Figure 3 shows three dimensional view of KufaSat orbit. Figure 4 shows a ground track of KufaSat projected onto a two-dimensional world map over one day (fifteen revolutions of the satellite around earth). This figure illustrates the orbital track for KufaSat as a sun-synchronous satellite in near-polar orbit. The orbital track relative to the Earth's surface is due to a combination of the orbital plane of the satellite coupled with the rotation of the Earth beneath the satellite, so it gives a good indication of the daily coverage of a satellite in sun-synchronous orbit. Figure 5 shows satellite position, figure 6 shows satellite velocity, and figure 7 shows Orbital elements.

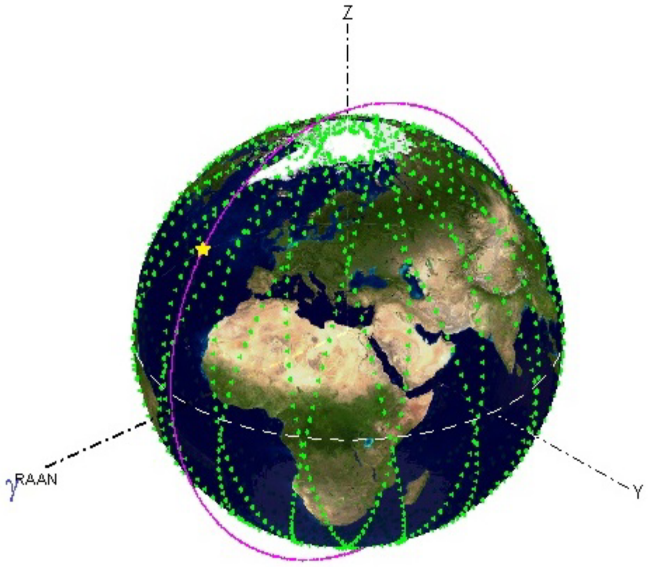

Fig. 3. Three dimensional view of orbit

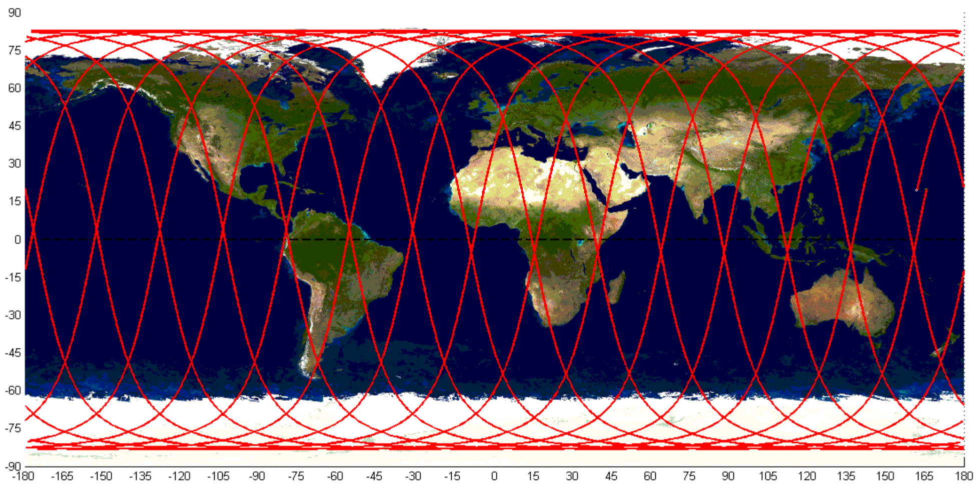

Fig. 4. Ground track over one day

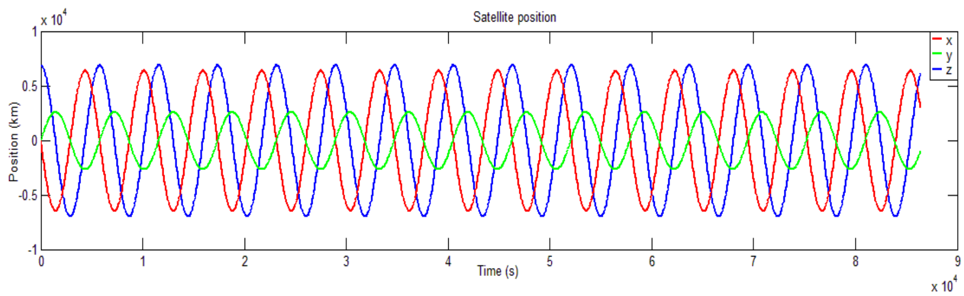

Fig. 5. Satellite position

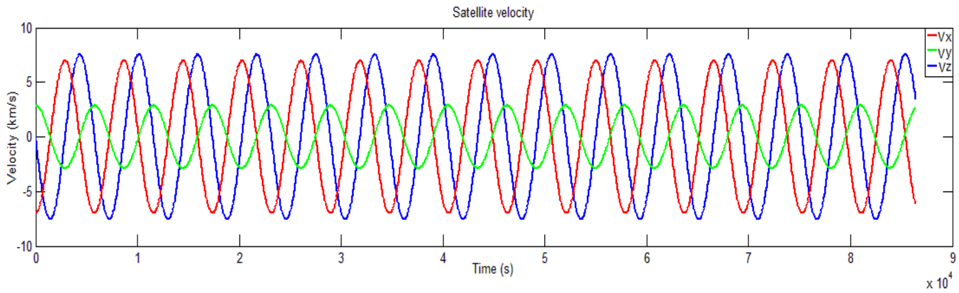

Fig. 6. Satellite velocity

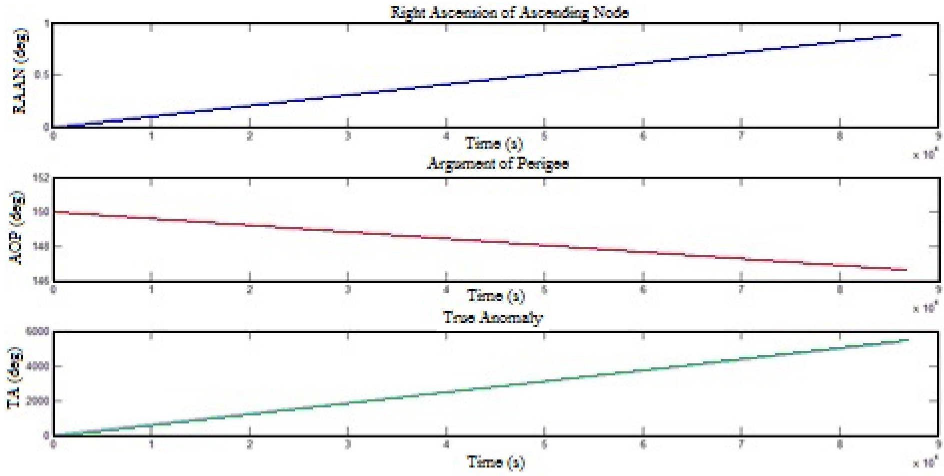

Fig. 7. Orbital elements

B. Simulation using GMAT:

General Mission Analysis Tool (GMAT) is used to simulate and analysis the orbit design of KufaSat. UTC Gregorian Epoch format and EarthMJ2000Eq coordinate system are used in simulation. Force model with RSSStep error control, Earth as a central body, JachiaRoberts atmosphere model and JGM-2 gravity model were used as a propagator setup to evaluate the drag on KufaSat. The Gravitational bodies include the Sun and Moon. Many of other bodies have such a small effect on the orbit of KufaSat so they can be neglected. Other non-gravitational forces included in the propagation, these are atmospheric drag and solar radiation pressure (SRP).RungeKutta89 with 60 sec initial step size, 0.001 sec mini step size, 2700 sec max step size, and 9.99e-12 accuracy are used as integrator in simulation. The analysis includes orbit determination and prediction of satellite position and velocity, satellite tracking, and command summary. Figure 8 shows three dimensional view of KufaSat orbit.

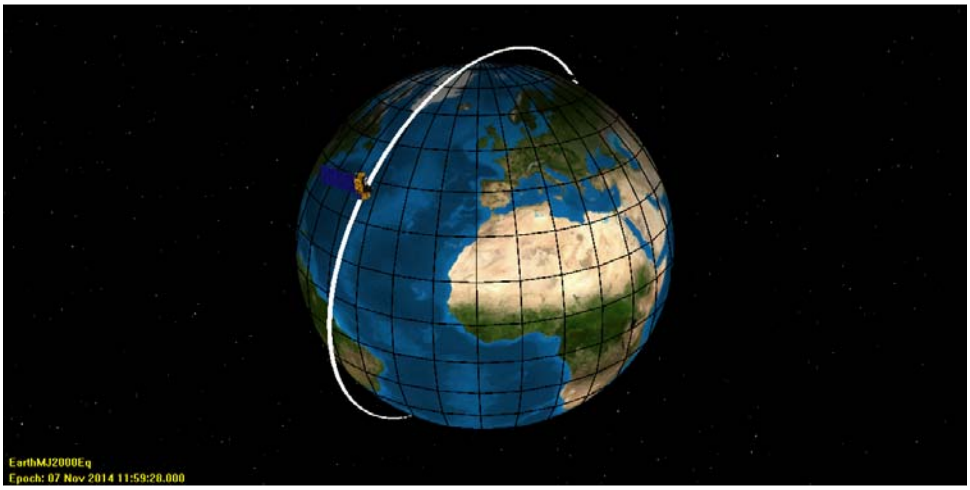

Fig. 8. Three dimensional view of KufaSat orbit

To support the CubeSat, the ground system must be able to track orbital position. The ground track obtained for the computed Keplerian and simple orbit design is simulated in GMAT, figure 9 shows a ground track of KufaSat projected onto a two-dimensional world map over one day (fifteen revolutions of the satellite around earth). As a result of the Earth's rotation the LEO satellite, can be observed by a ground stations for three consecutive passes, after which the satellite will be out of reach of the ground station for many hours.

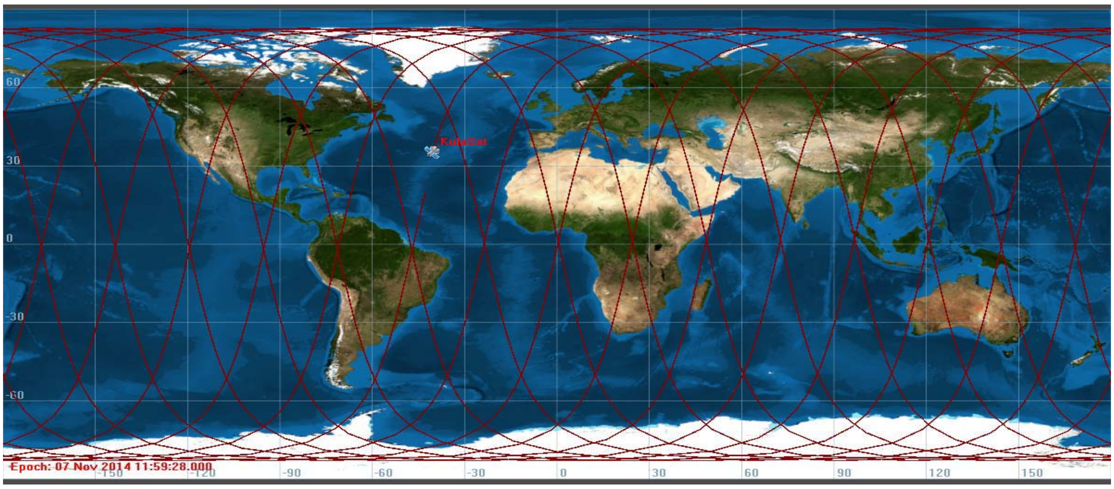

Fig. 9. The ground track over one day

Figures (10, 11) show position vector and velocity vector in EarthMJ2000Eq coordinate system from GMAT.

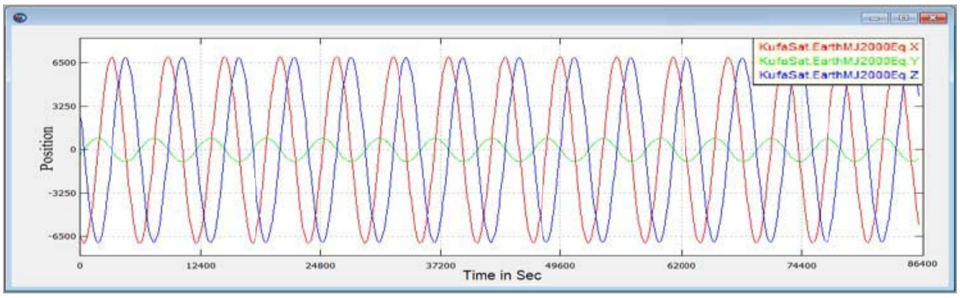

Fig. 10. Position Vector

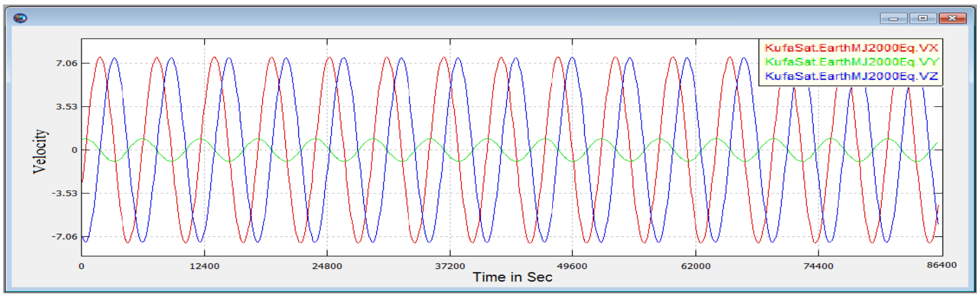

Fig. 11. Velocity Vector

Figures 12, 13, 14, and 15 show orbit stability represented by Keplerian Elements: Argument of Perigee, Inclination, Eccentricity, and Semi-major Axis respectively in EarthMJ2000Eq coordinate system for 86400 sec (one day).

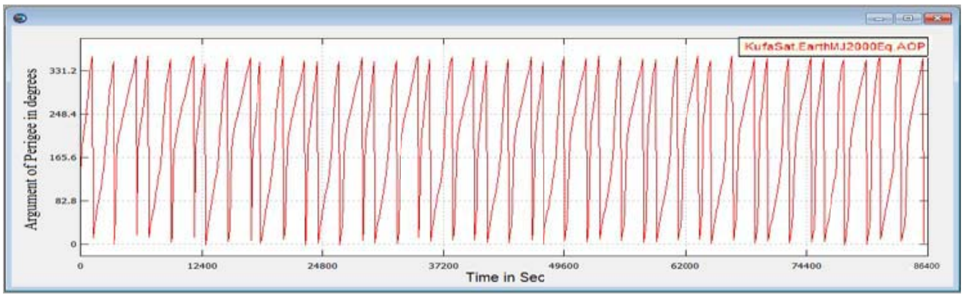

Fig. 12. Argument of Perigee

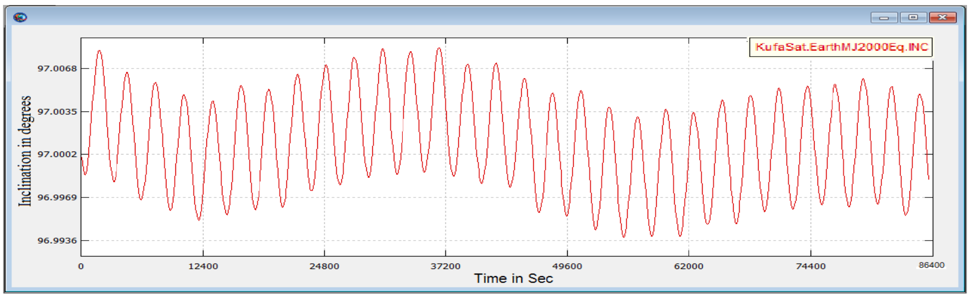

Fig. 13. Inclination

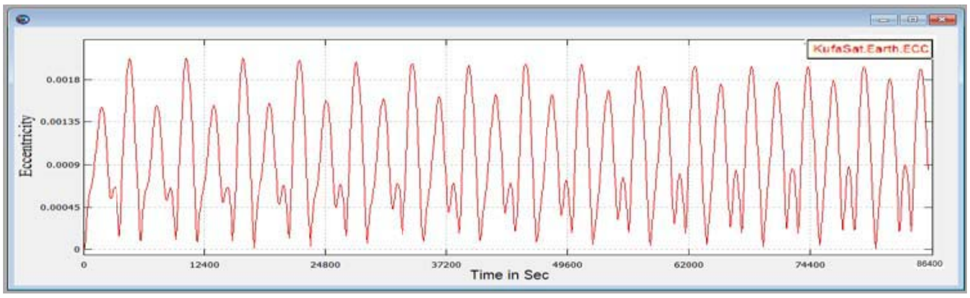

Fig. 14. Eccentricity

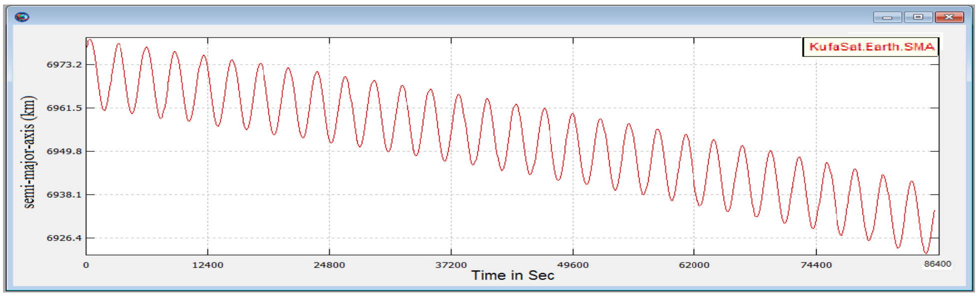

Fig. 15. Semi-major Axis

The results out of simulations using MATLAB and General Mission Analysis Tool show that the two programs can generate same orbit shape and same ground track when the simulations performed according to same orbital elements, atmosphere model, and gravitational bodies.

5. CONCLUSION

It is important to predict and determine an accurate orbit of the satellite in order to planning a successful satellite mission. Satellite orbits are described using a set of orbit elements. This work investigated and calculated the KufaSat orbit elements. KufaSat is the first Iraqi student satellite project sponsored by University of Kufa for imaging purposes and remote sensing. Eccentricity with a very small value which implies a near-circular orbit that allow the satellite to view all areas of the Earth from approximately the same altitude, and thus with the same resolution and sensitivity. Inclination of 97 degrees allows the solar cells to be in a constant solar illumination which mean a large amount of generated power. The orbit design of KufaSat according to the calculated Keplerian elements has been simulated by using MATLAB first and then by using General Mission Analysis Tool.

This simulation provides support for model development and integration, gives ability to observe the effects on shape and position of the orbit. The results indicate the position and the velocity vector in EarthMJ2000Eq coordinate system in addition to orbit stability represented by state of four Keplerian Elements: Argument of Perigee, Inclination, Eccentricity, and Semi-major Axis in EarthMJ2000Eq coordinate system for 86400 sec (one day). The difference between each component of position and velocity out of simulations using MATLAB and General Mission Analysis Tool was very small. The two dimensions representation of the satellite ground track and three dimensional view of orbit out of simulation using Matlab are validated by comparing it with General Mission Analysis Tool. The results out of simulations using MATLAB and General Mission Analysis Tool were identical in orbit shape and ground track when the simulations performed according to same orbital elements, atmosphere model, and gravitational bodies.

REFERENCES

Afful M. A., 2013 "Orbital Lifetime Predictions of Low Earth Orbit Satellites and the effect of a DeOrbitSail", Master Thesis, Stellenbosch University.

Barnes-Svarney P., 2005,"The Handy Math Answer Book", Visible Ink Press. Sally Ride EarthKAM https://earthkam.ucsd.edu/resources/students_orbital_mechanics

Chessab M. M., 2013, "Gravity Gradient Stabilization of Nano Satellite Using Fuzzy Logic Controller", Master Thesis University of Kufa.

Chessab M. M., Shehab Abd-AL-Razak, 2014, "Direct Fuzzy Logic Controller for Nano-Satellite", Journal of Control Engineering and Technology JCET, Vol. 4 Issue. 3. DOI: 10.14511/jcet.

Chobotov V. A. 2002, Orbital Mechanics Third Edition, AIAA Education Series.

Eshagh M. and N. Alamdari, 2007, "Perturbations in orbital elements of a low earth orbiting satellite", Journal of the Earth & Space Physics. Vol. 33, No. 1, P. 1-12.

Fortescue P. W. and John P. W. Stark, 1995, "Spacecraft Systems Engineering" Third Edition.

GMAT, "Mathematical Specification Documentation", published on the web at http://gmat.gsfc.nasa.gov/downloads/documentation.html

Paskowitz M. E. and D. J. Scheeres, 2006," Design of Science Orbits About Planetary Satellites", JOURNAL OF GUIDANCE, CONTROL, AND DYNAMICS, Vol. 29, No. 5.

Wikipedia, The free encyclopedia, https://en.wikipedia.org/wiki/Orbital_elements

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