Orbit Simulator of the Alsat-1 First Algerian Microsatellite in Orbit

A.M. Si Mohamed, A. Boudjemai, H. Benzeniar
Centre National des Techniques Spatiales
1, Avenue de la Palestine, Arzew, Algeria

Y. Hashida, H. Steyn, M.N. Sweeting
Surrey Satellite Technology Limited
Guilford, Surrey GU2 7XH, United Kingdoom

Résumé : Cet article présente un simulateur d'orbite du premier microsatellite Algérien Alsat-1 qui donne la position et la vitesse orbitale d'Alsat-1.
Le simulateur tient compte des perturbations principales telles que la traînée atmosphérique et la force non sphérique du corps central.
Les résultats de simulation obtenus montrent une bonne cohérence avec les données orbitales d'Alsat-1.

Abstract : This paper presents an orbit simulator of the Alsat-1 first Algerian microsatellite which propagates the orbital position and velocity of Alsat-1. The simulator is taking into account major perturbations such as atmospheric drag and the non spherical central body force.
The results out of the simulation were compared and tested with real Alsat-1 tracking algorithm; it was shown that the compared program can then generate same track shape.

1. Introduction

Algeria via the National Centre of Space Techniques (CNTS) chooses to develop their technical capabilities through technology transfer by setting a strategy to implement space technology and to meet her known and potential needs. Alsat-1 project translates the awareness and the understanding on the part of the user community of the benefits of space technology.
Algeria's first satellite Alsat-1 was designed and constructed by Surrey Satellite Technology Limited (SSTL) in the United Kingdom within a collaborative program with CNTS.
The simulation of the Alsat-1 orbit runs under C++ code and MATLAB/Simulink environment by returning the position and velocity vectors at corresponding times. The input parameters to the function developed are defined in a vector containing the Kepler elements as well as the Start and End simulation time. The orbits are propagated using the standard Simplified General Perturbations, SGP4 module for near Earth satellites.
The predicted orbit is used to produce scheduling aids which indicate spacecraft environmental conditions (such as sun or shadow, interference regions and altitude) as well as all potential station-to-station contact times. Using this list of environmental conditions, the experimenter can plan scientific data collection, and knowing the station view periods, can select the times needed to meet mission communications requirements.

Fig. 1 Alsat-1: First Algerian Microsatellite in Orbit

2. Theoretical Aspects

2.1 Orbit Dynamics

The nature of this investigation requires that the position of a satellite moving under the combined influence of the earth's (non-ideal) gravitational field and a number of other perturbing forces must be calculated. The movement of a satellite under the influence of the Earth's gravity field as well as other perturbing forces is described by the following two first order differential equations

$$\dot{\mathbf{r}} = \mathbf{v}$$
$$\ddot{\mathbf{r}} = -\mu\frac{\mathbf{r}}{r^3} + \mathbf{a}_p \qquad (1)$$

Where :
$\mathbf{r}$: The satellite position vector;
$\mathbf{v}$: The satellite vector velocity;
μ : The Earth's gravitational constant;
$\mathbf{a}_p$: The vector sum of all perturbation accelerations acting on the satellite.

2.2 Perturbation

2.2.1 The Earth Gravitational Effect

Most solar system bodies are known to have figures which depart from the spherical model of the particle. The non sphericity of the gravitational potential may give rise to a significant perturbation of satellite trajectories. Therefore, accurate orbit generation, or propagation, may require the inclusion of non spherical terms. The gravitational potentials of the Earth and Moon are the best known of all solar bodies, because of extensive tracking and analysis of close Earth and lunar satellites.
The method of representing its potential is classical. The gravitational field of the body is derived from a scalar potential U that satisfies Poisson's Equation

$$\Delta U = -4\pi\rho_k(r,\varphi,\lambda) \qquad (2)$$

Where :
r : The magnitude of the vector from the body's centre of mass to the satellite;
ϕ : The geocentric, selenocentric, or planetocentric latitude;
λ : The geocentric, selenocentric, or planetocentric longitude.

Above the surface of the perturbing body, the mass density ρ, is assumed zero, consequently, Equation (2) reduces to the Laplacian, $\Delta U = 0$. Standard separation of variables techniques yield to the following solution :

$$U = \frac{\mu}{r} + \mu\sum_{n=2}^{N}\sum_{m=0}^{n}\frac{a_e^n}{r^{n+1}}P_{nm}(\sin\phi)(C_{nm}\cos m\lambda + S_{nm}\sin m\lambda) \qquad (3)$$

Where :
U : The gravity potential;
r : The magnitude of radius vector ;
a : The Earth's equatorial radius;
n,m : The degree and order of spherical harmonics;
P_{nm} : The Legendre functions;
C_{nm}, S_{nm} : The coefficients of spherical harmoncs;
λ : The geocentric east longitude of the satellite;
ϕ : The geocentric latitude of the satellite.

In the equation above, the first term is the point mass part. The other terms are an expansion of spherical harmonics to represent the non spherical effects of the Earth mass, i.e. the Earth gravitational perturbation.

2.2.2 The Atmospheric Drag

Satellites below 1000 km orbit height are strongly affected by drag forces. Although the air density is extremely low at such altitudes, the high velocity of a satellite leads to significant acceleration, obtained by the following equation

$$\ddot{\mathbf{r}} = -\frac{C_D A}{m}\frac{\rho}{2}|\dot{\mathbf{r}}_a|\dot{\mathbf{r}}_a \qquad (4)$$

Where :
C_D : The drag coefficient;
A : The cross-sectional area of the vehicle perpendicular to the direction of motion;
m : The mass of the vehicle;
ρ : The atmospheric density at the vehicle's altitude;
$|\dot{\mathbf{r}}_a|$: The speed of vehicle relative to the rotating atmosphere;
$\dot{\theta}$: The rate of rotation of the Earth.

2.3 Orbit Predictions

Orbit prediction involves two steps – (1) Orbit Propagation, (2) Calculation of position and velocity vectors. Orbit Propagation takes into account the various perturbations some which may be secular, short periodic (changing very fast within an orbit) and long periodic (slowly changing with a frequency of many orbits or a few days).
Once the orbital elements are propagated to the new time, the position and velocity vectors in the inertial frame are computed using a set of transformation equations.
The simulator calculates the satellite's acceleration due to the Earth's gravity field with a vector equation derived from the gradient of the potential function expressed as :

$$a_g(\mathbf{r},t) = \nabla U(\mathbf{r},t) \qquad (5)$$

This acceleration vector is a combination of pure two-body or point mass gravity acceleration and the gravitational acceleration due to higher order nonspherical terms in the Earth's geopotential. In terms of the Earth's geopotential U, the inertial rectangular Cartesian components of the satellite's acceleration vector are as follows:

$$\ddot{x} = \left(\frac{1}{r}\frac{\partial U}{\partial r} - \frac{z}{r^2\sqrt{x^2+y^2}}\frac{\partial U}{\partial \phi} \right)x - \left(\frac{1}{x^2+y^2}\frac{\partial U}{\partial \lambda} \right)y \quad (6.a)$$

$$\ddot{y} = \left(\frac{1}{r}\frac{\partial U}{\partial r} - \frac{z}{r^2\sqrt{x^2+y^2}}\frac{\partial U}{\partial \phi} \right)y + \left(\frac{1}{x^2+y^2}\frac{\partial U}{\partial \lambda} \right)x \quad (6.b)$$

$$\ddot{z} = \left(\frac{1}{r}\frac{\partial U}{\partial r} \right)z + \left(\frac{\sqrt{x^2+y^2}}{r^2}\frac{\partial U}{\partial \phi} \right)y \quad (6.c)$$

Where $r = \|\mathbf{r}\| = \sqrt{x^2+y^2+z^2}$ is the magnitude of radius vector of the satellite and (x,y,z) its coordinate position.

3. Simulation Results

The default inputs of the Alsat-1 orbit generator are given as follows:

Table 1. Orbital Inputs

Semi-major axis [metre]	7024263.281
Eccentricity	0.001344
Inclination [degree]	98.1253
Ascending node [degree]	38.3367
Argument of perigee [degree]	95.5909
Mean anomaly [degree]	264.6702

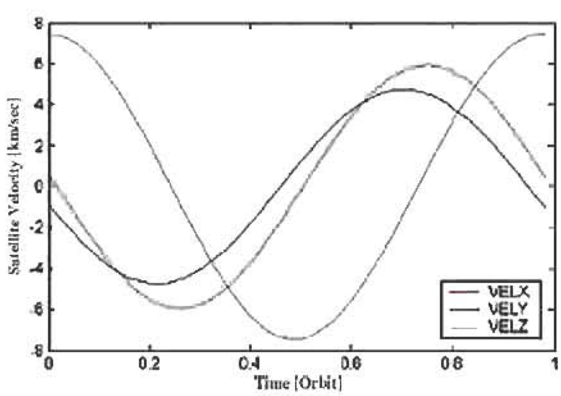

Fig. 2 Satellite Position

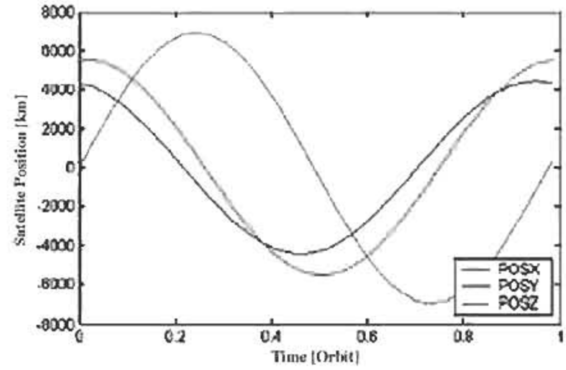

Fig. 3 Satellite Velocity

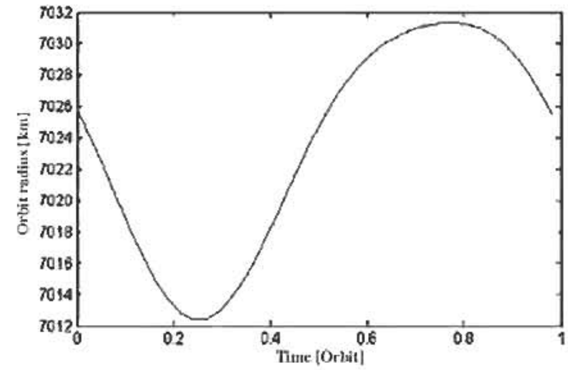

Fig. 4 Radius of orbit satellite

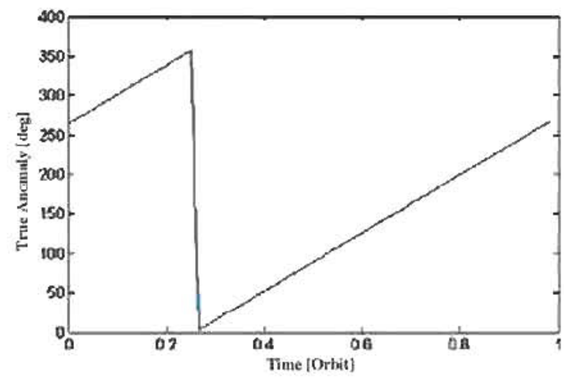

Fig. 5 True anomaly satellite

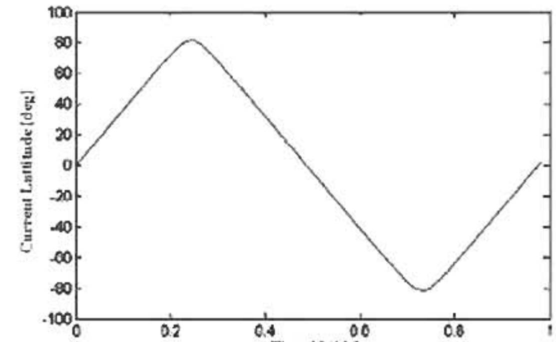

Fig. 6 Current latitude satellite

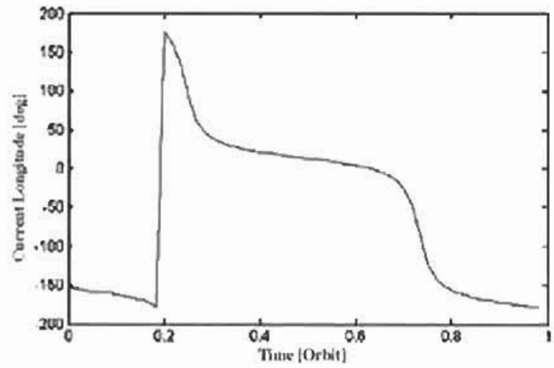

Fig. 7 Current longitude satellite

The orbital elements are no longer constant, they vary with time. Three types of variations

- Secular - linear with time;
- Short period - vary with orbital period;
- Long period - varies with period from days to months.

The atmospheric drag has a significant effect for satellites below 1000 km.

The deviation of the Earth from a perfect sphere is mainly responsible for the changes seen in the Right Ascension of the Ascending Node, Argument of Perigee, Mean Anomaly,...

The Luni-Solar gravitational forces takes into account the effects on an orbit produced by the gravitational pull of the Moon and Sun. These forces contribute to the long-term changes in the Right Ascension of the Ascending Node and the Argument of Perigee, and also produce long-term changes in inclination and eccentricity.

4. Conclusion

The aim of the work presented is to give a real time simulation, which provides support for model development and integration, simulation execution and analysis of Alsat-1 results.

The development presented constitutes a preliminary version of the work whose accent was put especially on the modelling of the Alsat-1 orbit on the one hand, on the other hand this work allows also to :

- Get use for training program and education to spaces techniques;
- It will help the country Algeria to acquire satellite and approach and learn spacecraft technology;

The extension of the work to the application of Alsat-2 which is an Earth observation Algerian microsatellite with high resolution imagery.

References

Alsat-1 project documentation, 2003, delivered by SSTL to CNTS under the Know How Transfer Program.

Battin, R.H., 1987, *"An Introduction to the Mathematics and methods of Astrodynamics"* AIAA Educational Seriers, New York, USA.

Escobal, P. R., 1976, *"Methods of Orbit Determination"* Krieger Publishing Company, USA.

King-Hele, D., 1987, *"Satellite Orbits in an Atmosphere: Theory and Applications"* Blackie and Son Ltd., London, UK.

Schlarmann, B. K.I., 1994, *"Orbit Determination during the BREM-SAT Mission* ,"45th Congress of the International Astronautics Federation, IAA-94-IAA.11.3.769, Jerusalem, Israel, October 9-14.

Sellers, Jerry Jon., 1994 *"Understanding space: an introduction to astronautics"* Jerry Jon Sellers, New York; London: McGraw Hill.

Sommer, Widjaja, Widle, 1989, *"Study of on board Orbit Determination"*, ESA Contact Number 7463/78/D/IM (SC), Bremen 1989

Valadimir A.Chobotv, 1996 *"Orbital Mechanics"* Second Edition, Alexander Bell Drive, Reston, Virginia.

Vincent L. Pisacane and Robert C., 1994 , *"Fundamentals of space systems"* New York; Oxford: Oxford University Press - The Johns Hopkins University/Applied Physics Laboratory.

Y. Hashida, 2003, *"AODCS Design for Future UoSAT Standard Platform"*, Surrey Space Centre, Guilford, UK.