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The Altitude Attainment and Inclination Alignment for the Satellite Launched from Kourou Site

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ABSTRACT

The geostationary orbit is a circular orbit 35786 kilometers above the Earth's equator and following the direction of the Earth's rotation. Communications satellites are often placed in the geostationary orbit. The rockets are used to place the satellite in geostationary orbit. The main goal of the launching process is to enable the satellite to acquire the desired geostationary orbit space parameters. There are two methods applied for putting satellites in the geostationary orbit, the first one is when a rocket takes the spacecraft to a low Earth circular orbit and then towards geostationary transfer orbit, and by the second method, the low circular orbit is skipped and the satellite goes straight to geostationary transfer orbit. Because of too large distance from the Earth's surface this placement to geostationary transfer orbit is not done at once. The location of Kourou at French Guiana is the launching site of France and also shared with ESA (European Space Agency). The launch process from Kourou site firstly places the satellite at geostationary transfer elliptical orbit, and then the orbit geostationarity is achieved through two phases, the first phase is the altitude attainment at 35786km and the second is the inclination to be aligned with equatorial plane. During the whole process, the needed satellite's propellant mass must be minimized. Three thrusts apogee and nodal thrust method applied for the altitude attainment and then the inclination alignment for the satellite to be consolidated in geostationary orbit, which is launched from Kourou site, is simulated and analyzed within this paper.

Keywords: GEO; satellite; orbit; inclination

1 Introduction

Usually, to attain the geosynchronous or geostationary Earth orbit, the propellant makes up about half of a satellites mass, what directly impacts the launching process and satellite's lifetime. When launching and then consolidating the satellite on its own high circular orbit (ex. geosynchronous), the needed satellite's propellant mass must be minimized.

The Hohmann transfer is well known for the minimum of propellant mass used for satellite transfer into high orbits. The Hohmann transfer orbit is based on two instantaneous velocity changes. The transfer consists of a velocity impulse on an initial circular orbit, in the direction of motion and collinear with velocity vector, which propels the space vehicle into an elliptical transfer orbit. The second velocity impulse also in the direction of motion is applied at apogee of the transfer orbit which propels the space vehicle into a final circular orbit at the final altitude [1].

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After too long efforts and experiments, in January 2013, the Robotic Refueling Mission (RRM) performed its first test on board the ISS (International Space station), demonstrating that remotely controlled robots successfully transfer fuel in space. Lofting a satellite with only a portion of the fuel it would need to complete its mission, with the ability to inject more propellant in the future, creates two advantages: the lower launching cost and more room for instruments to be packed aboard. Robotically refueling satellites in orbits in the near future, will allow satellite service providers to dramatically extend their services and the satellites' lifetime [2].

This paper considers the concept of satellite acquirement of space orbit parameters, followed by main characteristics of launching vehicles and launching methods towards geosynchronous (geostationary) orbits. The launching process toward geostationary orbit from the site in Kourou is analyzed, associated with maneuvers and thrust to be applied in order the geostationary orbit to be attained and consolidated. Maneuvers and applied thrusts are directly correlated with propellant consumption/saving, what is among main goals on future scientific research related to satellite systems.

2 Acquiring Space Orbit Parameters

The main goal of the launching process is to enable the satellite acquire the desired space orbit parameters. The position of the orbital plane in space is specified by means of two parameters - the *inclination i* and the *right ascension of the ascending node* (*Ω*). Inclination *i* represents the angle of the orbital plane with respect to the Earth's equator. The right ascension of the ascending node *Ω* defines the location of the ascending and descending orbital crossing nodes (these two nodes make a *line of nodes*) with respect to a fixed direction in space, known as Vernal equinox. Vernal equinox is direction of line joining the Earth's center and the Sun on the first day of spring [3]-[5].

In order the orbit plane to attain the exact desired position in space it is too important that correct conditions are established at the satellite injection point. Further, to keep the desired orbit position in order to enable the appropriate satellite mission, the various in orbit operations such as orbit stabilization, orbit correction and stations keeping are mandatory.

The angle defining the right ascension of the ascending node *Ω*, is basically the difference between two angles, α and β, where α is the angle made by the longitude of the injection point at the time of launch with the direction of vernal equinox and β is the angle made by the longitude of the injection point at the time of launch with the line of nodes, as shown in Fig. 1 [3] and given as (1):

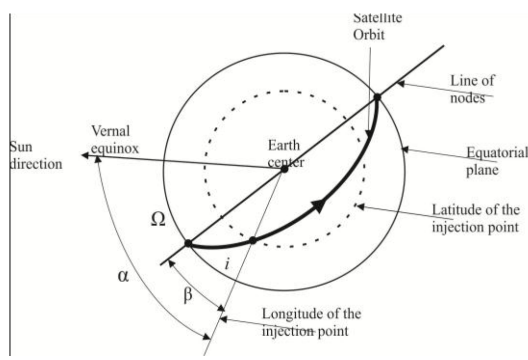

Figure. 1 Two Space orbit parameters.

$$\Omega = \alpha - \beta \quad (1)$$

Considering Fig.1, it is obvious that to ensure the satellite orbits within a given plane, the satellite must be injected at a certain specific time, depending upon the longitude of the injection point, at which the line of nodes makes the required angle with the direction of the vernal equinox. Thus, for a known angle of inclination and the longitude of injection point (launching site) the launching time must be exactly determined in order to acquire the desired right ascension of the ascending node Ω [3]-[4].

The angle of inclination *i* of the orbital plane can be determined from the known values of the azimuth angle (A_Z)and the latitude (*l*) of the injection point, expressed as [3]:

$$cosi = sinA_Z cosl \quad (2)$$

The azimuth angle (A_Z) at a given point in a satellite orbit is the angle made by the projection on the local horizon of the satellite vector velocity at that point with the north.

From Eqn. (2), for the angle of inclination to be zero, the right hand side of (2) should be one, what could happen only if the launching site is at Equator plane, thus the latitude *l=0* and $A_Z = 1$. For launching sites above the Equatorial plane it is $l > 0$ and$sinA_Z < 1$, what will decrease $cosi$ and consequently increase inclination angle *i*, mathematically expressed as [3]:

$$i > l \quad (3)$$

Thus, it can be concluded that the satellite will tend to orbit in a plane which will be inclined to the equatorial plane at the angle equal to or greater than the latitude of the injection point.

3 Launching Methods

Metadata information in the Web Pages and Expansio used to launch a satellite is known as a launch vehicle. There are two types of launch vehicles: expendable rockets such as Ariane of the European Space Agency and Atlas Centaur of the United States which are destroyed while completing their mission, and the other that is employed by a re-usable launch vehicle such as the Space Shuttle of the United States and the Buran of Russia [6].

In order for a satellite to be launched successfully, the launch rocket must be placed in a vertical position initially. This allows the rocket to penetrate the densest layer of the Earth’s atmosphere quickly, which helps reduce fuel consumption. The concept of the launch vehicle is given in Fig. 2 [6].

Expendable rockets for communication satellites have three stages. The first stage raises the satellite at about 50 miles (80 km), and then the second stage raises the satellite to 100 miles above the Earth’s surface, and when the launch vehicle is out of the Earth's atmosphere, the satellite separates from the upper stage and places it into the transfer orbit.

The upper stage of the launch vehicle is connected to the satellite itself, which is enclosed in a metal shield, called a “fairing.” The fairing protects the satellite while it is being launched and makes it easier for the launch vehicle to travel through the resistance of the Earth's atmosphere. Once the rocket reaches an altitude near the satellite's orbit height, the satellite is ejected from the rocket's nose cone and the rocket falls back to Earth, burning up upon reentering our atmosphere. The rockets of the upper stage fire after the satellite is in space and put the satellite in the exact spot where it is needed. The fairing splits apart once the satellite is above the Earth's atmosphere and burns up in the Earth's atmosphere. Once the satellite reaches its desired orbital height, it unfurls its solar panels and communication antennas, which had been stored away during the flight. The satellite then takes its place in orbit with other satellites and is ready to provide services. After the satellite is placed in its transfer orbit, the rocket's mission is complete, and its remnants fall to Earth. When the satellite is already placed in the orbit, then there are mandatory activities in order to consolidate the desired in advance determined orbit space parameters, what is the further subject of this paper [7].

To attain the geostationary orbit, firstly we need to get to the appropriate high altitude and the secondly is needed to change the inclination (most launch pads are not located at the equator) to zero angle. There are two methods applied for putting satellites in the geostationary orbit. One of methods is when rocket takes the spacecraft to a circular orbit at an altitude of 200-300km and then towards geostationary transfer orbit. By the second method, further discussed, the circular orbit is skipped and the satellite goes straight to geostationary transfer orbit. Transfer orbits are often used to test vehicle systems before committing to further action [3]-[4].

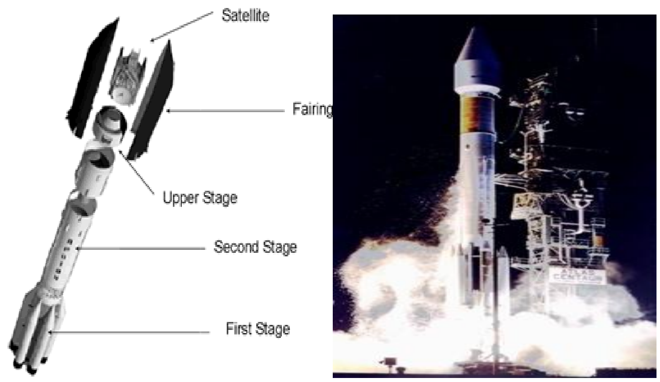

Figure 2. The launch vehicle concept (left) and Atlas expendable satellite launch vehicle (right) [6].

The Geostationary Transfer Orbit (GTO) is elliptical in shape with its perigee at an altitude between 180 km and 200 km and its apogee at the geostationary altitude. The spacecraft has an engine that is called the Apogee Kick Motor (AKM). That motor is fired on while the satellite is at the apogee of its elliptical transfer orbit aiming to circularize the orbit. But it usually is not all done at once. Usually the AKM is used three times respectively by three burns. Further is discussed the satellite launch process from the launch site Kourou in French Guiana toward final geostationary orbit.

4 Kourou launching site

The location of Kourou at French Guiana was selected in 1964 to become the spaceport of France and it is operational since 1968. In 1975, France offered to share Kourou with ESA (European Space Agency) [8].

The satellite launch site at Kourou in French Guyana is located at coordinates of 5° 9′ 34.92″ N, 52° 39′ 1.08″ W and it is particularly suitable as a location for a spaceport as it fulfills the two major geographical requirements of such a site: it is quite close to the equator, and it has uninhabited territory, so that lower stages of rockets and debris from launch failures cannot fall on human habitations.

The structure of Kourou center is presented in Fig. 4, where it is obvious that the control center is located around 12km far from the launching zone because of safety reasons [6].

5 Launching Process and Parameters Acquisition

The entire satellite launching process by Ariane vehicle toward geostationary orbit, launched from Kourou site at French Guiana is further described by activity steps.

The orbit circularization and the geostationarity are achieved through two phases, the first one is the altitude attainment at 35786km (geosynchronous orbit altitude) and the second is the inclination alignment (zero inclination) with Earth's equatorial plane.

5.1 Circularization and Altitude Attainment

The launch vehicle takes the satellite to a point that is intended to be the perigee of the transfer orbit, at a height of about 200km above the Earth's surface. At this point the satellite is ejected from the rocket's nose cone (see Fig. 2); respectively it is injected in the transfer orbit. The satellite along with its apogee boost motor is injected before the launch vehicle crosses the equatorial plane.

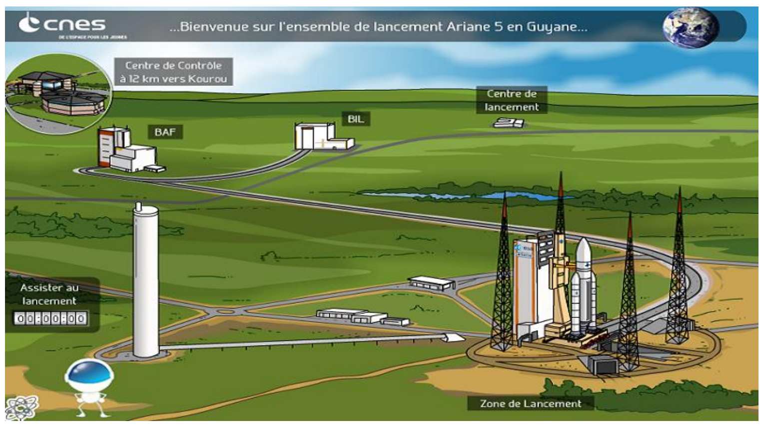

Figure 4. Satellite launch site at Kourou [6]

The injection velocity at perigee v_p expressed as

$$v_p = \sqrt{\left(\frac{2\mu}{r_p}\right) - \left(\frac{2\mu}{r_a + r_p}\right)} \tag{4}$$

is such that the injected satellite attains an eccentric elliptical orbit with an apogee altitude at about 35786km where the velocity is v_a expressed by:

$$v_a = \sqrt{\left(\frac{2\mu}{r_a}\right) - \left(\frac{2\mu}{r_a + r_p}\right)} \tag{5}$$

r_p and r_a are respectively perigee and apogee distances (from the Earth's center), $\mu = m \cdot G = 3.986 \cdot 10^5 \, km^3 / s^2$, G is the Earth's gravitational constant and m Earth's mass. The correlations in between apogee (r_a), perigee (r_p) distances with respective attitudes at apogee (h_a) and at perigee (h_p) is:

$$r_{a,p} = R_E + h_{a,p} \tag{6}$$

For launching conditions from the site at Kourou toward geostationary orbit, and considering Earth's radius R_E= 6371km, stem out that the perigee and apogee distances, respectively are r_p=6571km and $r_a = 42159$km [1]. Applying these parameters in (4) and (5), yields out that the velocity at the injection point (perigee) in order to create a geostationary transfer orbit, is:

$$v_p = 10.248 \text{ [km/s]} \tag{7}$$

and the velocity at apogee of the geostationary transfer orbit is

$$v_a = 1.589 \text{ [km/s]} \tag{8}$$

The main goal is to circularize the orbit, but usually it is not all done at once. For this purpose, usually the Apogee Kick Motor (AKM) is used three times respectively by three burns, generating three velocity thrusts as presented in Fig. 5.

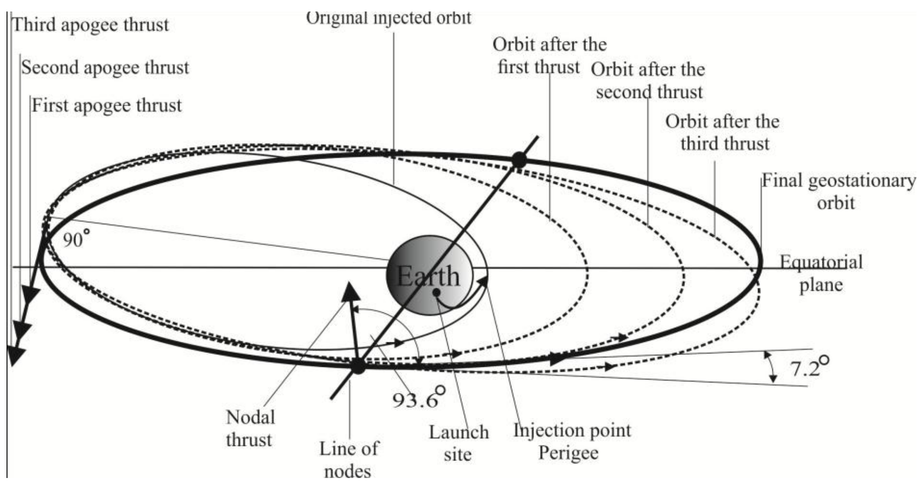

Figure 5. Three thrust at apogee and nodal thrust.

Considering that the initial altitude at the injection point is at 200km, then attempting by the first burn (thrust) to attain an altitude of about 12000km, by the second one an altitude of about 24000km, and finally the geostationary altitude of 35786km by the third thrust, further applying (4), (5), and (6) the respective velocities and altitudes attained up to the final orbit circularization are given in Table 1.

Table 1. Velocities and respective altitudes

	Velocity at perigee (v_p) [km/s]	Altitude at perigee (h_p) [km]	Velocity at apogee (v_a) [km/s]	Altitude at apogee (h_a) [km]
Injection phase	10.248	200	1.589	35786
Under thrust 1	5.501	12000	2.383	35786
Under thrust 2	3.910	24000	2.801	35786
Under thrust 3 (Final stage)	3.067	35786	3.067	35786

Table1 confirms that under the third thrust at apogee point it is attained the geostationary altitude and the orbit is already circularized. The variation of velocities after each thrust is presented in Fig, x, where it is obvious the convergence of both velocities, at apogee and perigee after the third thrust, when the orbit is circularized and consequently the satellite has constant velocity at each point of the orbit.

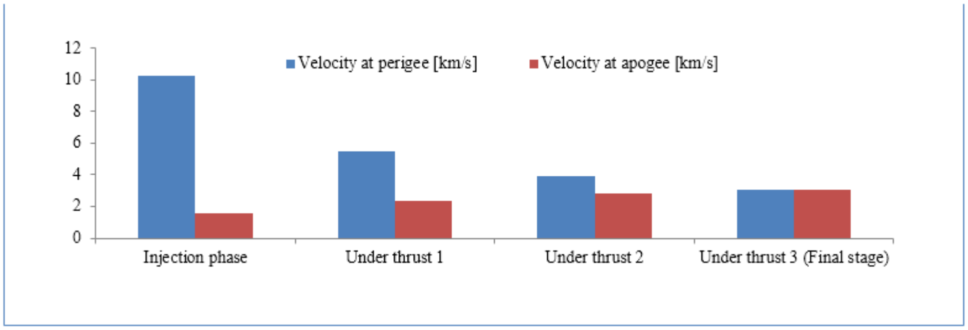

Figure 6. Velocities variation.

Under the consideration of only the orbit circularization respectively altitude attainment, the applied thrust velocity vectors at apogee point in order to increase the perigee distances are coplanar at each burn (thrust) and having the same direction of the satellite's motion. Each velocity vector is tangential at apogee point of the orbit (see Fig. 5).

The satellite is always in move with an actual velocity at apogee, thus in order to increase the perigee, the increment on apogee velocity vector by the apogee thrust has to be applied, expressed as:

$$\Delta v_{a(i)} = v_{a(i)} - v_{a(i-1)} \quad (9)$$

where i indicates burning-thrust step, and $i - 1 = 0$ indicates injection phase. Finally, based on Table 1 and (9) the intensities of coplanar thrust vectors applied three times at apogee for orbit circularization are:

$$\Delta v_{a1} = 794[\text{m/s}] \quad \Delta v_{a2} = 418[\text{m/s}] \quad \Delta v_{a3} = 266[\text{m/s}] \quad (10)$$

This orbit circularization is achieved by three appropriate thrust maneuvers without affecting any change to the inclination. For the attained circularization and attitude of 35786km the satellite velocity in the circularized orbit is

$$v_a = 3.067\ [\text{km/s}] \quad (11)$$

5.2 Inclination Alignment

The projection of the injection velocity vector in the local horizon plane at launching site of Kourou makes an azimuth angle of 85°. The latitude of the launch site is 5° 9′ 34.92″. The inclination angle attained by the geostationary transfer orbit is calculated from (2), and it is:

$$i = 7.2° \quad (12)$$

Next action is another thrust to be applied in order to bring the orbit inclination to 0◦, respectively the already circularized orbit to be aligned with equatorial plane, and finally the geostatinarization to be completed.

The change in inclination Δican be externally affected by applying a thrust velocity vector Δv_i at angle of $(90° + \Delta i/2)$ at one of nodes as illustrated in Fig. 5. The thrust is given by [3]:

$$\Delta v_i = 2v sin\left(\frac{\Delta i}{2}\right) \quad (13)$$

where *v* is the satellite's velocity at circularized orbit, at (11) and Δi is the inclination range that should be corrected. Since, the already circularized orbit has inclination of 7.2° which should be brought at 0°, then $\Delta i/2$ is $3.6°$. Finally the velocity thrust to be applied for orbit alignment with equatorial plane, and making the orbit as fully geostationary one, is:

$$\Delta v_i = 385.15[\mathrm{m/s}] \tag{14}$$

This thrust is applied at either of the two nodes (ascending or descending), as in Fig. 5. The last step is fine tuning to attain the exact inclination, attitude and longitude of the satellite.

6 Conclusion

The satellite launching process from the Kourou site is given. It is confirmed that the geostationarity is achieved by three thrusts applied at apogee and a single thrust applied at the nodal point. By three apogee thrusts it is attained the geostationary orbit altitude and circularity, then by the nodal thrust the satellite orbit is aligned with Earth's equatorial plane. Apogee thrust vectors are tangential with orbit and the nodal vector is normal with orbit plane. Mathematical thrust velocity vectors calculation is also given.

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