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Evaluation of performance and space operations of a geostationary satellite with electrical propulsion

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Table of contents

Table of contents	II
Table of figures	3
Acronym list.....	4
Symbol list	5
Objective	6
1 Basic principles.....	7
1.1 Geostationary satellite	7
1.2 Perturbation actions on a GEO satellite	9
1.2.1 Attraction of Sun and Moon.....	10
1.2.2 Non spherical Earth.....	12
1.2.3 Solar radiation	14
1.3 Space chemical propulsion.....	15
1.4 Space electrical propulsion.....	17
1.5 Collision avoidance	19
2 Manoeuvring a GEO satellite	20
2.1 Station keeping	20
2.2 Collision avoidance	22
3 Manoeuvring a GEO satellite with electrical propulsion	24
3.1 Definition of the mathematical model.....	24
3.2 Analysis results for station keeping.....	29
3.3 Considerations	35
4 Outcomes	37
Bibliography.....	38

Table of figures

Figure 1 - Geostationary and geosynchronous orbits.....	7
Figure 2 - Forces acting on a GEO satellite	8
Figure 3 - Earth as seen from GEO orbit	8
Figure 4 - Station keeping box (Micheli).....	9
Figure 5 - Sun and Moon attraction force	10
Figure 6 - Velocity on inclined orbit.....	11
Figure 7 – Satellite ground track for a circular, geosynchronous orbit with 45° of inclination (Micheli)	11
Figure 8 – Satellite ground track for geosynchronous orbit with 45° of inclination and 0.1 ellipticity (Micheli).....	12
Figure 9 - Forces generated by non-spherical Earth	13
Figure 10 - Solar radiation on GEO satellite (Micheli)	14
Figure 11 – Rocket engine schematic	15
Figure 12 - Bi-propellant propulsion system	16
Figure 13 - Schematic of Hall effect thruster.....	18
Figure 14 – N-S station keeping maneuver.....	20
Figure 15 – E-W station keeping maneuver schematic.....	21
Figure 16 - E-W station keeping maneuver schematic	21
Figure 17 – Collision avoidance parameters.....	23
Figure 18 – Coordinate frames.....	25
Figure 19 – Propulsion configuration	25
Figure 20 – Schematic of station keeping box (lateral plane).....	28
Figure 21 – Schematic of station keeping box (equatorial plane).....	28
Figure 22- Evolution of latitude for electrical satellite	29
Figure 23 - Comparison on latitude evolution for HET and chemical thrusters satellites.....	30
Figure 24 - Evolution of longitude for electrical satellite	30
Figure 25 - Comparison on longitude evolution for HET and chemical thrusters satellites.....	31
Figure 26 - Evolution of orbit eccentricity for electrical satellite	31
Figure 27 - Comparison on eccentricity evolution for HET and chemical thrusters satellites	32
Figure 28 - Evolution of orbit inclination for electrical satellite	32
Figure 29 – ΔV for NE thruster on RTN frame	33

Acronym list

CBS	Chemical Bi-propellant Satellite
ECI	Earth Centered Inertial (frame)
GEO	Geostationary Orbit
HET	Hall Effect Thruster
OMD	Overall Miss Distance
PCA	Point of Closest Approach
PO	Primary Object
RTN	Radial Tangential Normal (frame)
SO	Secondary Object
TCA	Time of Closest Approach

Symbol list

Latin letters	
C_R	Solar radiation coefficient
F	Force (N)
G	Gravitational constant, $6,67428 * 10^{-11} \frac{m^3}{kg \ s^2}$
g	Gravtational acceleration at Earth surface, $9,81 \frac{m}{s^2}$
I_{sp}	Thrust specific impulse (s)
i	Orbit inclination (deg)
M	Earth mass, $5,9736 * 10^{24} kg$
m	Satellite mass (kg)
r	Orbit radius (km)
T	Orbit period (s); Thrust (N).
V	Satellite velocity
Greek letters	
α	Slew angle
γ	Cant angle
Δ	Variation
ω	Angular velocity (rad/s)
Subscript	
c	Centrifugal
eq	Equatorial
f	Final
ga	Gravitational attraction
GEO	Geostationary
i	Initial
v	Vertical

Objective

The purpose of this work is to characterize flight performances of a geostationary satellite with electric propulsion during station keeping operations. On the basis of these results, the appropriate manoeuvring strategy for the system will then be defined. Such an analysis is necessary considering the use of electric propulsion in lieu of the chemical one: in fact, although the adoption of these engines currently constitutes a strong innovation in the technological field, the performance characteristics and their methods of use require the definition of specific operation plans and adequate manoeuvring strategies for the optimal management of the space system. For this purpose, a mathematical model has been developed in order to describe flight performances of an electrically propelled geostationary satellite and typical parameters have been compared with the equivalents of a chemically propelled satellite.

1 Basic principles

This section gives basic principles for understanding the following of this work and includes the definition of:

- Geostationary satellite;
- Manoeuvring a GEO satellite;
- Space chemical propulsion;
- Space electrical propulsion.

1.1 Geostationary satellite

A geostationary satellite is a generic satellite describing a geostationary orbit around the Earth. A geostationary orbit is a particular geosynchronous orbit where the orbital period of the satellite is the same as the Earth's rotation period, the orbit trajectory is circular and lies on the equatorial plane.

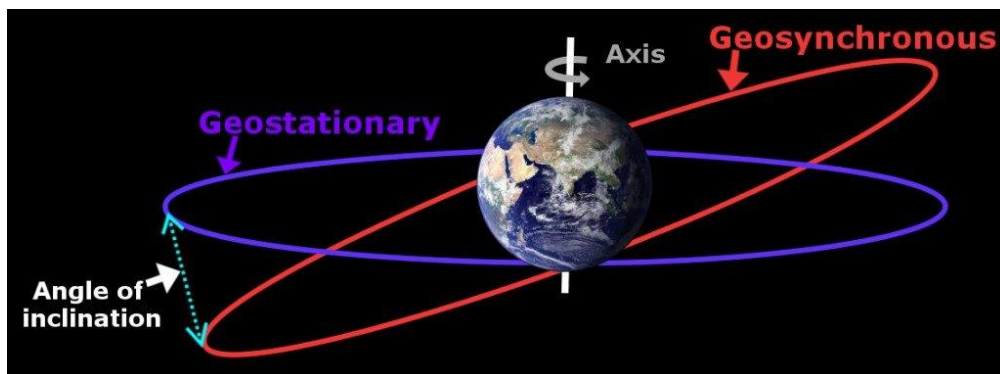


Figure 1 - Geostationary and geosynchronous orbits

Basing on these hypotheses and assuming that either the Earth and the satellite are spherical with an even mass distribution, it is possible to calculate the radius which allows an equatorial, circular orbit to be geosynchronous (and hence geostationary) by equalling centrifugal acceleration to gravitational attraction force.

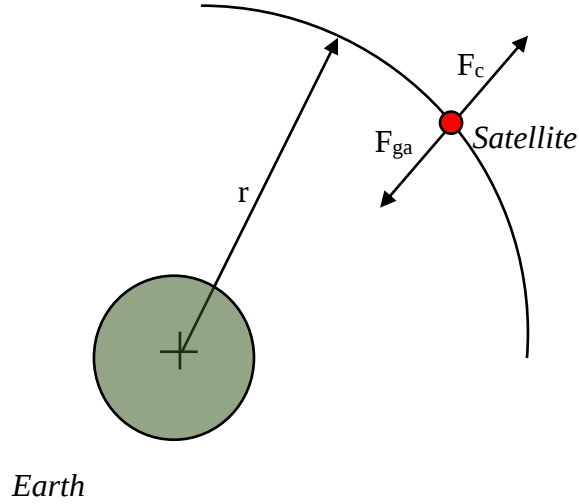


Figure 2 - Forces acting on a GEO satellite

Hence one has

$$\frac{GMm}{r^2} = m\omega^2 r \text{ with } \omega = \frac{2\pi}{T} \quad (1)$$

$$r_{GEO} = \sqrt[3]{\frac{GMT^2}{4\pi^2}} = 42\,168 \text{ km} \quad (2)$$

Where G is the gravitational constant, M is the Earth mass, m is the satellite mass, r is the orbit radius, T is the orbital period (or length of the sidereal day) and ω the angular velocity. Rearranging Equation (1) for satellite velocity, one can find

$$V_{GEO} = \sqrt{\frac{GM}{r}} = 3,075 \text{ km/s} \quad (3)$$

Such a long distance from Earth allows the satellite to see a very wide surface of the Earth itself, as represented in Figure 3. For this reason, GEO satellite are mainly used for telecommunications and meteorological services since a very limited number of satellites can guarantee the full coverage of the Earth (in theory three satellites are sufficient).

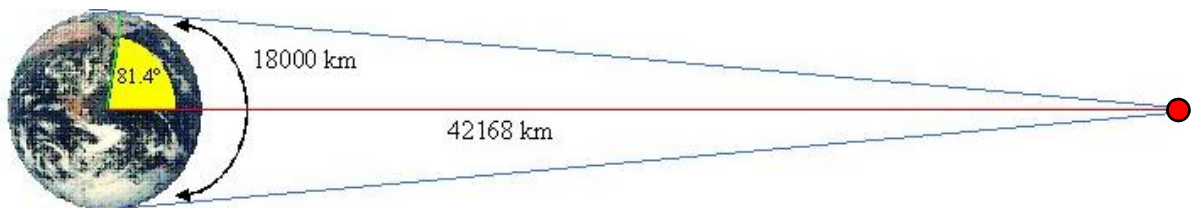


Figure 3 - Earth as seen from GEO orbit

The number of satellites that can occupy simultaneously the GEO orbit is anyway limited because each satellite must not interfere with the others for operational safety and to avoid any potential issue both in terms of spacecraft control and radiofrequency interference. The aspect of assigning a GEO orbital position to a satellite is regulated by international organizations and laws: for this reason, it is crucial to remark that every satellite in GEO orbit has a well-defined orbital slot called “station keeping box” that, in nominal conditions, must not be exceeded. The “station keeping box” can be regarded as a volume within which the satellite is free to move in order to accommodate the perturbations that modify the nominal GEO orbit. The dimensions of the station keeping box are approximately 75 km x 75 km x 35 km, as reported in *Figure 4*, which correspond to a tolerance of $0,1^\circ$ on latitude and longitude and 0,0004 on orbit eccentricity respectively.

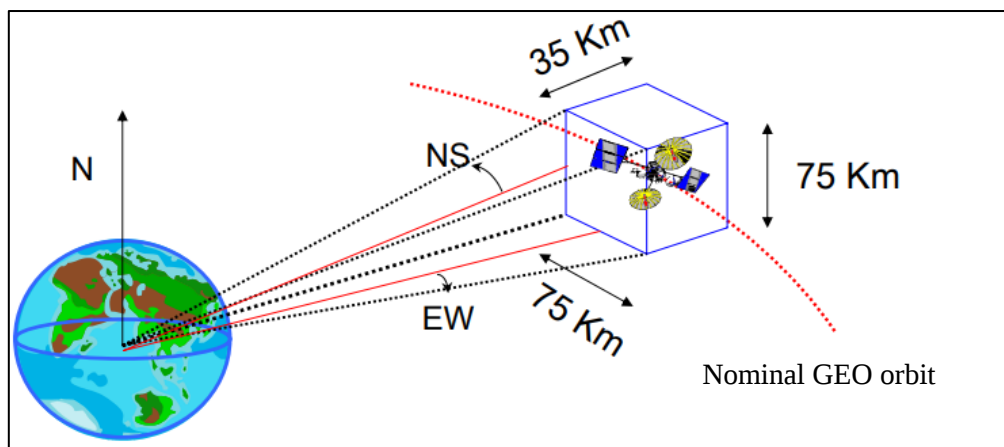


Figure 4 - Station keeping box (Micheli)

1.2 Perturbation actions on a GEO satellite

During its motion on GEO orbit the satellite is subject to several effects that induce modification to the orbit itself. Major sources of perturbations for a satellite in GEO orbit are the attraction of Sun and Moon, the uneven mass distribution and the oblateness of the Earth and the solar radiation. The trajectory of the satellite must then be accurately controlled in order to maintain it within the station keeping box. Additional corrections and controls may be necessary in order to avoid potential collision with satellites located in adjacent orbits or when foreign objects come extremely close to the spacecraft (usually in the range of few kilometres). All these phenomena require the satellite to have a propulsion system and an attitude control system which guarantee the full control of the spacecraft.

1.2.1 Attraction of Sun and Moon

The Sun and the Moon modify the GEO orbit because they exert an attractive force F_a on the satellite, acting as external bodies as represented in *Figure 5*.

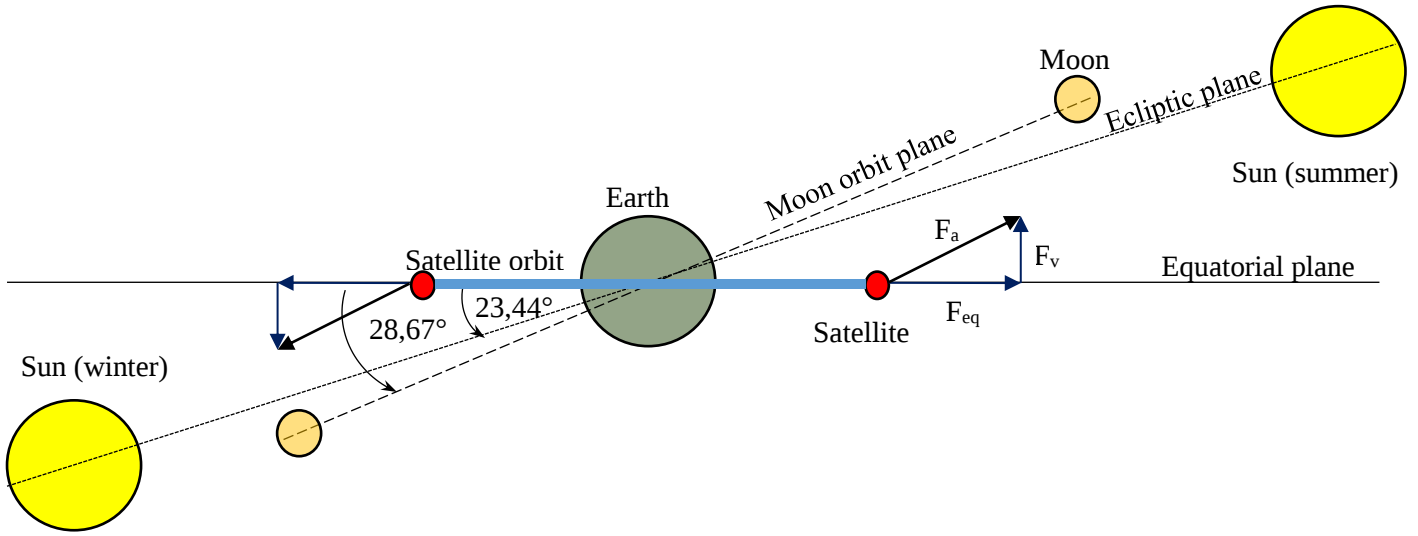


Figure 5 - Sun and Moon attraction force

The Sun and the Moon generate on the GEO satellite an attractive force F_a which can be decomposed in F_v (perpendicular to the equatorial plane) and F_h (lying on the equatorial plane). These forces depend on the mutual position of Earth and Moon on their orbital planes and, consequently, can vary over time. In particular, since the ecliptic plane has an inclination of 23,44° relative to the Equatorial plane, the Sun contribution to the vertical force F_v is maximum when the Earth is at perihelion and aphelion and null when Earth and Sun are aligned on the ascending and descending node. A similar contribution is given by the Moon with the addition of a fluctuating contribution, in fact the inclination of the Moon orbit plane is not constant and varies between $23,45^\circ \pm 5,12^\circ$ every 18,6 years. Globally, the Sun and the Moon attraction generate a vertical force F_v which causes the satellite to drift along the North-South direction. This force modifies the inclination of the GEO orbit and causes the satellite to approach the bottom and the upper faces of the station keeping box. If uncorrected, orbit inclination reaches its maximum of 15° in 27,5 years and returns to 0° after subsequent 27,5 years (Larson & Wertz, 1999). The in-plane force F_{eq} results in the drift of the satellite in the radial direction, modifying orbit eccentricity and hence causing the satellite to approach the “radial” faces of the station keeping box.

When the vertical force F_v is applied to the satellite, the orbit is no more geostationary because the plane of the orbit is not equatorial. Due to orbit inclination (*Figure 6*), the satellite velocity referred to a point on Earth is smaller than V_{GEO} and results

$$V = V_{GEO} \cos(i) \quad (4)$$

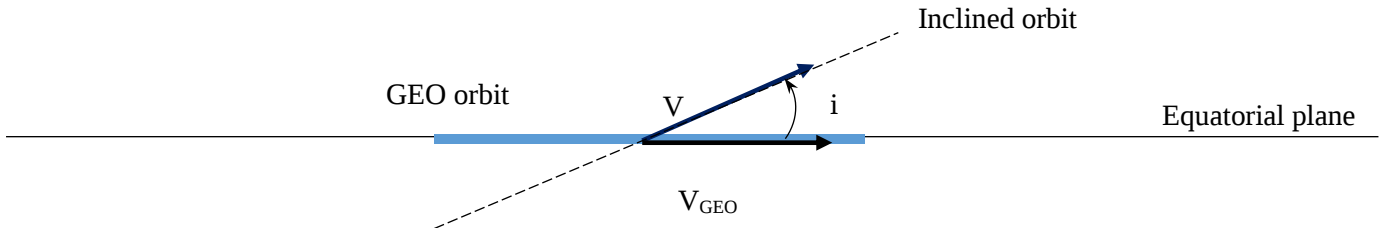


Figure 6 - Velocity on inclined orbit

For an Earth observer, the satellite on the inclined orbit is slower respect to a GEO satellite and increases its latitude when travelling on the ascending portion of the orbit; on the opposite, when travelling on the descending portion of the orbit, it is faster and decreases its latitude. From the Earth observer, this motion appears as represented in *Figure 7*.

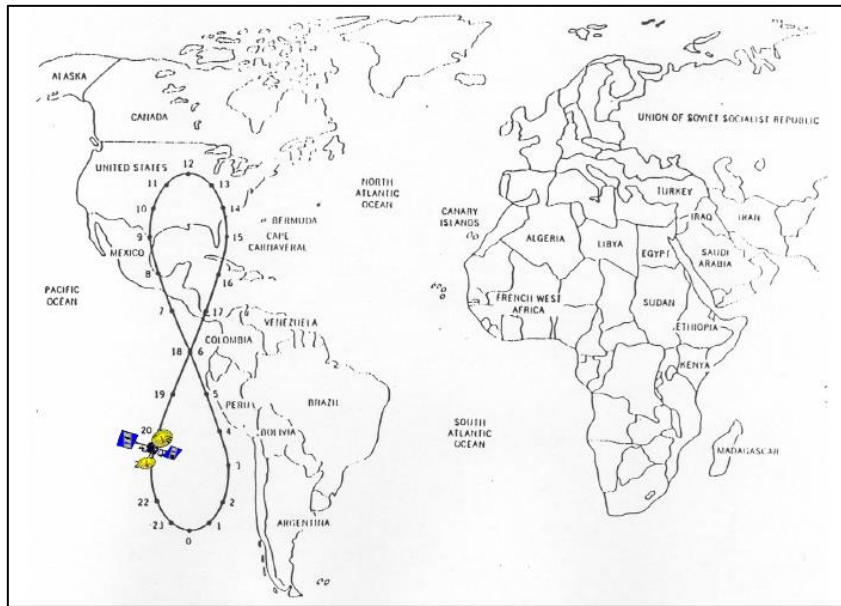


Figure 7 – Satellite ground track for a circular, geosynchronous orbit with 45° of inclination (Micheli)

When the force F_{eq} is applied to the satellite, the orbit is no more geostationary because eccentricity is non-zero and hence the orbit is not circular. According to Kepler's second law, the elliptic orbit forces the satellite to slow down in proximity of the apogee and accelerate in proximity of perigee. Similarly to the previous case, for an Earth observer the satellite appears shifting in longitude because

of the change in velocity. When considering orbit inclination and ellipticity, satellite ground track appears as reported in the following figure.

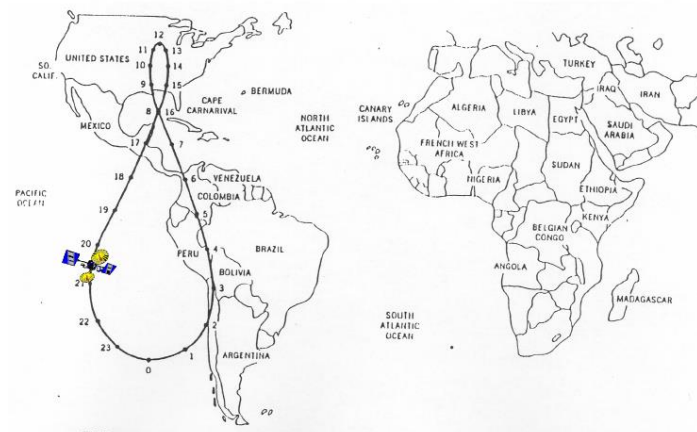


Figure 8 – Satellite ground track for geosynchronous orbit with 45° of inclination and 0,1 ellipticity (Micheli)

1.2.2 Non spherical Earth

Equations (1), (2) and (3) have been obtained assuming that either the satellite and the Earth were spherical with an even mass distribution. However Earth is not exactly spherical and its mass is not uniformly distributed, in fact there is a difference of about 21 km between mean equatorial radius and polar radius, and the equator has an elliptic shape with a difference between major and minor axes of about 150 m. This particular shape generates two main effects:

- A non-uniform geopotential, which results in a gravitational attraction force higher for GEO orbits;
- The gravitational attraction force is not aligned with the Earth center and gives a force contribution f_θ tangent to the orbit (Figure 9).

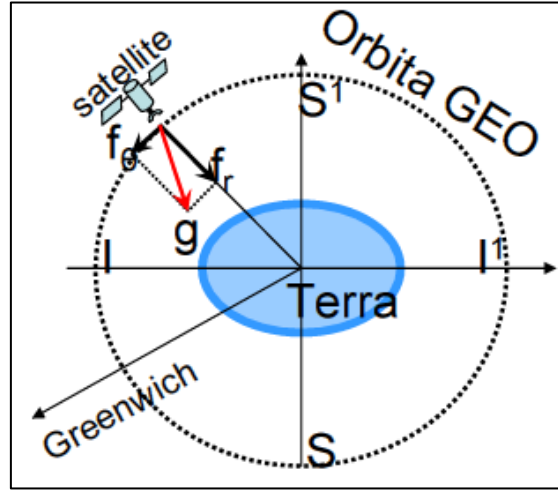


Figure 9 - Forces generated by non-spherical Earth (Micheli)

The higher gravitational attraction force, according to Equation (3), gives to the GEO satellite a velocity higher than V_{GEO} resulting in the loss of synchronization with a fixed point on Earth. Consequently the orbit radius shall be increased, moving the satellite in the radial direction within the station keeping box, in order to recover the loss of synchronization.

The force contribution f_θ causes the satellite to accelerate or decelerate along the orbit, resulting in a satellite drift in East/West direction within the station keeping box. Additionally, if the force f_θ is equiverse with satellite velocity, the acceleration is associated with an increase of orbit radius because the energy of the system is increased; on the opposite, if the force f_θ has opposite verse with satellite velocity, the deceleration is associated with a reduction of orbit radius because the energy of the system is decreased. Similarly to the previous case, the change of satellite velocity generates the loss of synchronization with a fixed point on Earth.

1.2.3 Solar radiation

Solar radiation generates a force F_{ps} which perturbs satellite orbit. This radiation force depends on satellite characteristics such as cross sectional area, mass and material of invested surfaces. The way solar radiation affects satellite orbit varies with the interactions of the solar photon flux over the satellite (*Figure 10*).

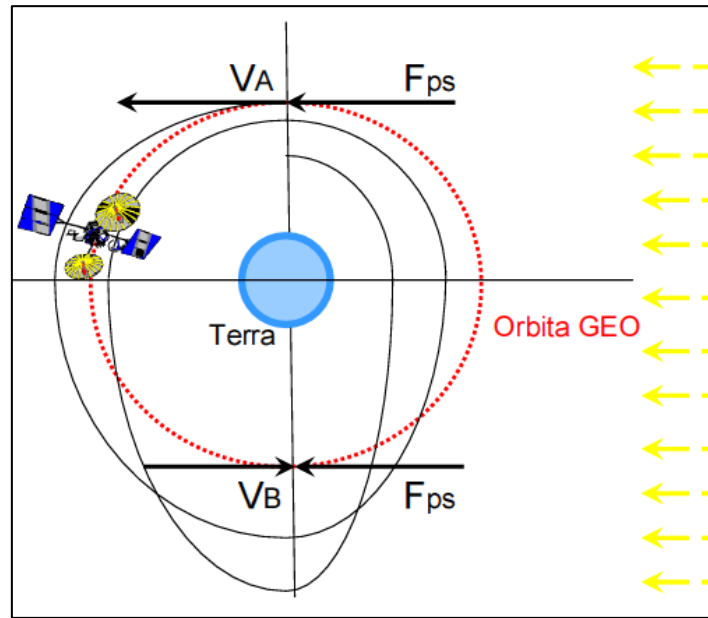


Figure 10 - Solar radiation on GEO satellite (Micheli)

Similarly to the effects given by non-spherical Earth, the force F_{ps} may accelerate or decelerate the satellite causing an increase or a decrease of the orbit radius and hence ellipticity.

1.3 Space chemical propulsion

All the phenomena described in the previous sections perturb the GEO stationary orbit of a generic satellite. In order to compensate these effects and maintain the satellite within the station keeping box, spacecrafts are equipped with onboard propulsion system.

Currently, the majority of satellites is equipped with chemical propulsion system, that is a rocket engine where exhaust gases are generated by reacting one or two chemical components and are subsequently ejected through the nozzle (*Figure 11*).

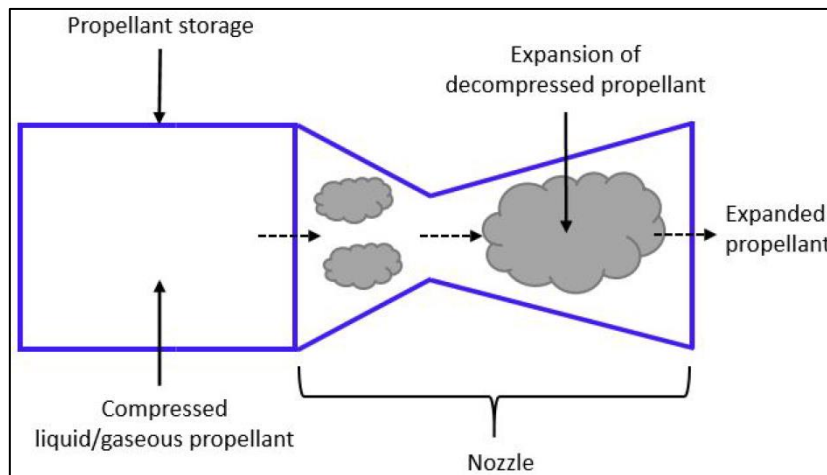


Figure 11 – Rocket engine schematic

Typically mono-propellant engine utilize hydrazine or hydrogen peroxide which decompose spontaneously in presence of a catalyst or at very high temperatures. Typical bi-propellant engines for satellite applications utilize fuel and oxidizer which are burned inside the combustion chamber. Chemical components are stored in pressurized tanks and then distributed to nozzles located over the surface of the spacecraft. This architectural concept is represented in the following figure.

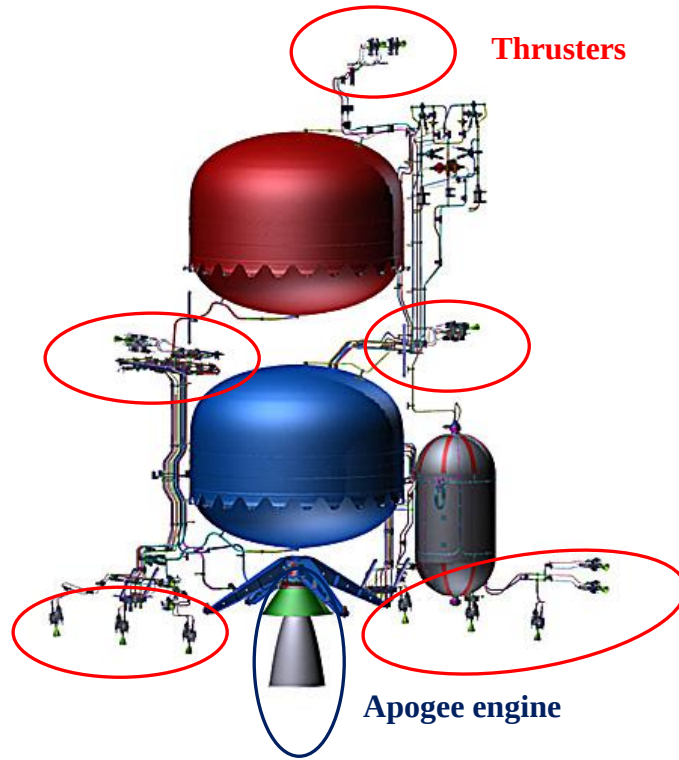


Figure 12 - Bi-propellant propulsion system

Thrusters are usually located on the external surface of the satellite and aligned along roll, pitch and yaw axis, in order to guarantee the full control of the system and to cooperate with the attitude control system. Additional components of the chemical propulsion system include valves, filters, tubing and sensors in order to fully monitor and control the system.

In terms of performance, the most relevant parameters which are normally considered are:

- Thrust (N) – the amount of force that the engine can generate;
- Specific impulse (s) – the time an engine can generate thrust for a given mass flow of fuel and oxidizer;

Specific impulse is a crucial parameter either for the design of the spacecraft and for its operating life, in fact it is directly linked to the variation of speed of the spacecraft and necessary fuel mass by the relationship

$$\Delta V = g I_{sp} \log \left(\frac{m_i}{m_f} \right) \quad (5)$$

Where ΔV is the speed variation of the spacecraft, g is the gravitational acceleration at Earth surface, I_{sp} is the specific impulse and m_i and m_f indicate the mass of the spacecraft at the beginning and at the end of the engine burn. A higher specific impulse requires less fuel mass for the same ΔV and hence

allows for an overall mass saving on the spacecraft. The following values can be considered as reference for the presented parameters for a bi-propellant thruster installed on a GEO satellite.

Reference values of typical parameters for spacecraft with bi-propellant thruster	
Thrust	10 N
Specific impulse	300 s

Table 1 - Reference values for bi-propellant thruster

1.4 Space electrical propulsion

Electrical space engines represent a major technological innovation and it is estimated that in 2019 over 500 spacecrafts used electric propulsion for station keeping, orbit raising or primary propulsion. An additional study stated that in 2020 half of all new satellites adopted full electric propulsion. An electrical propulsion system uses electro-magnetic fields to accelerate exhaust gas and thus change the velocity of the spacecraft. Electric thrusters typically use much less propellant than chemical rockets because they operate at a higher specific impulse, but due to limited electric power, the overall thrust is much weaker compared to chemical rockets. Anyway electric propulsion can provide a small thrust for a long duration of time.

In the following, the discussion will be focused on Hall effect thruster, which is one of the most largely used technology in the sector and is adopted as reference thruster for the aim of this work.

Hall effect thrusters utilize an electric field to accelerate the ionized propellant and a magnetic field to ionize the propellant and counter act some side effects related to its functioning. A schematic of Hall thruster is reported in the figure below.

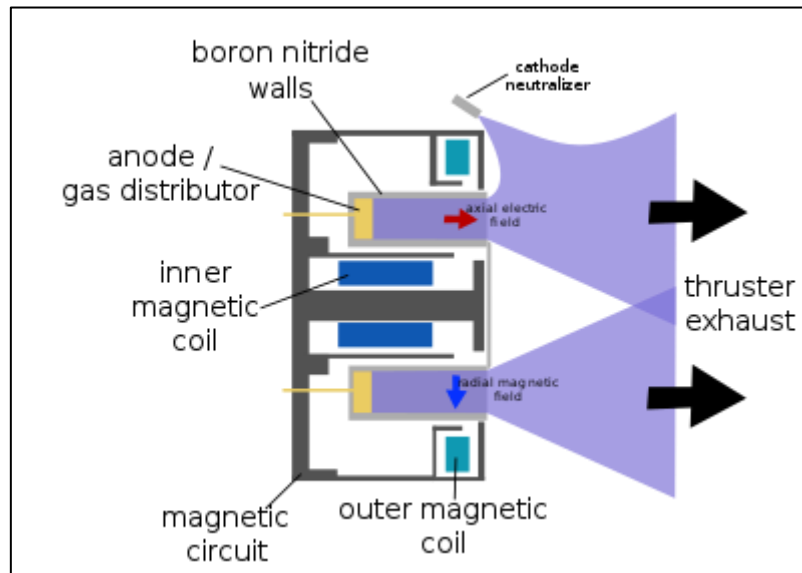


Figure 13 - Schematic of Hall effect thruster

The propellant is fed through the anode, which has numerous small holes in it to act as a gas distributor. Xenon and krypton are commonly used as propellant because of high atomic weight and low energy of ionization, other propellants of interest include argon, bismuth, iodine, magnesium and zinc. As the neutral propellant atoms diffuse into the channel of the thruster, they are ionized by collisions with circulating high-energy electrons; the ionization allows the electric field between the anode and the cathode to accelerate the propellant ions and hence generate thrust. Upon exiting, however, the ions pull an equal number of electrons with them, creating a plasma plume with no net charge. The radial magnetic field is then placed at the exit section and is designed to be strong enough to substantially deflect the low-mass electrons, but not the high-mass ions. The majority of electrons are thus stuck orbiting in the region of high radial magnetic field near the thruster exit plane. This rotation of the electrons is a circulating Hall current, and it is from this that the Hall thruster gets its name. Collisions with other particles and walls, as well as plasma instabilities, allow some of the electrons to escape from the magnetic field, and they drift towards the anode ionizing the injected propellant.

The thrust produced depends on the electrical power utilized to generate the electrical and the magnetic field. Typically, devices operating at few kW produce thrust of few hundreds mN. Due to the large amount of the electrical power required, Hall effect thruster require accessory equipment for the correct functioning of the system, such as dedicated solar array and electrical power units. For instance, these equipment have to be considered during the design phase and when defining satellite budgets.

Following tables present typical engine parameters for a Hall thruster and compare them to the equivalent of a bi-propellant chemical thruster.

Reference values of typical parameters for spacecraft with Hall thruster	
Thrust	0.275 N
Specific impulse	1800 s

Table 2 - Reference values for Hall thruster

	Bi-propellant thruster	Hall thruster
Thrust	10 N	0.275 N
Specific impulse	300 s	1800 s

Table 3 - Comparison between chemical bi-propellant and Hall thrusters

1.5 Collision avoidance

Collision avoidance include all necessary activities to prevent impacts between two space objects. Space environment (from low Earth orbit to beyond geostationary orbits) is largely crowded due to the high number of satellites and to frequent space missions, consequently it is not uncommon that two objects may get closer to each other, creating a potential issue of collision. Additionally, space environment include also an extremely large number of debris such as asteroids or fragments generated by pasted impacts; the size of these debris ranges from few millimetres to few tens of centimetres and it is estimated that the overall number amounts approximately to 129 million. The largest effort in collision avoidance is performed to prevent that these debris impact against spacecrafts. Along the decades, the United States' Department of Defence has developed the Space Surveillance Network (SSN), that is a network of ground based radars and optical systems and some space based sensors able to track orbiting objects larger than 10 cm. Orbit parameters of tracked object are made public and potential risk collision are communicated by United States' Combined Space Operation Center (CSpOC) to satellite operators in order to actuate remedial actions to avoid the collision. Potential collision between spacecraft and debris is much more frequent, but there may be cases where two spacecrafts may be involved in the process.

When a potential collision is identified, operators decide whether a collision avoidance manoeuvre should be performed or not. Typical parameters considered for the decision include the probability of collision, minimum distance between the objects, conflict with mission objectives, and expected time before occurring of the minimum distance between objects. In addition, major part of satellites are equipped with optical or radar sensors which can guarantee a sufficient field of view along specific directions in order to detect autonomously potential risk of collision with objects approaching the spacecraft.

2 Manoeuvring a GEO satellite

As described in *paragraph 1.1*, a geostationary orbit can be achieved only at an altitude very close to 35789 km, directly above the equator and with a circular trajectory of the satellite. These requirements, combined with the disturbing actions mentioned in *paragraph 1.2* and the necessity to avoid impacts with space objects ask for a continuous monitoring of the satellite orbit and the actuation of remedial actions on the system when certain conditions appear, as for example when the satellite is very close to the boundaries of the station keeping box or it may collide with an external object.

2.1 Station keeping

It is referred as station keeping all the satellite operations needed to maintain the satellite within a defined orbital slot. These activities are necessary because the combination of lunar gravity, solar gravity and the flattening of the Earth at its poles tend to modify orbit inclination, causing it to increase with a gradient of about $0,85^\circ$ per year (with a maximum of 15° after 27 years). To correct this perturbation, orbital station keeping maneuvers are performed in order to control the North-South movement of the satellite. Spacecraft thrusters are hence activated and eject propellant in direction orthogonal to the orbital plane. This part of the GEO station-keeping is called North-South control.

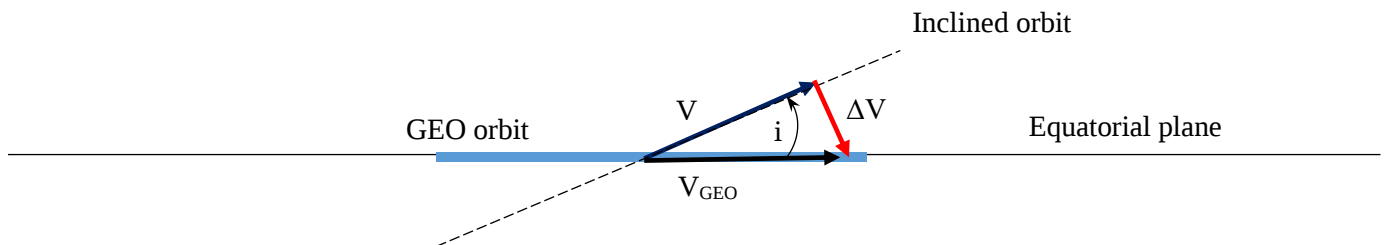


Figure 14 – N-S station keeping maneuver

Referring to *Figure 14*, the maneuver is performed by adding an additional velocity component (ΔV) to the actual satellite velocity (V) when it reaches the orbit node (i.e. the intersection between the GEO orbit and the inclined orbit). This results in a final component velocity (V_{GEO}) which lies on the equatorial plane. The ΔV needed to compensate for this perturbation amounts approximately to 50 m/s per year.

Other effects to take into account are the modification of the eccentricity and the longitudinal drift along the orbit, caused respectively by the asymmetry of the Earth along the equator and by the presence of two stable equilibrium points at $75,3^\circ\text{E}$ and 108°W . The corrections needed to compensate these effects are called East-West control.

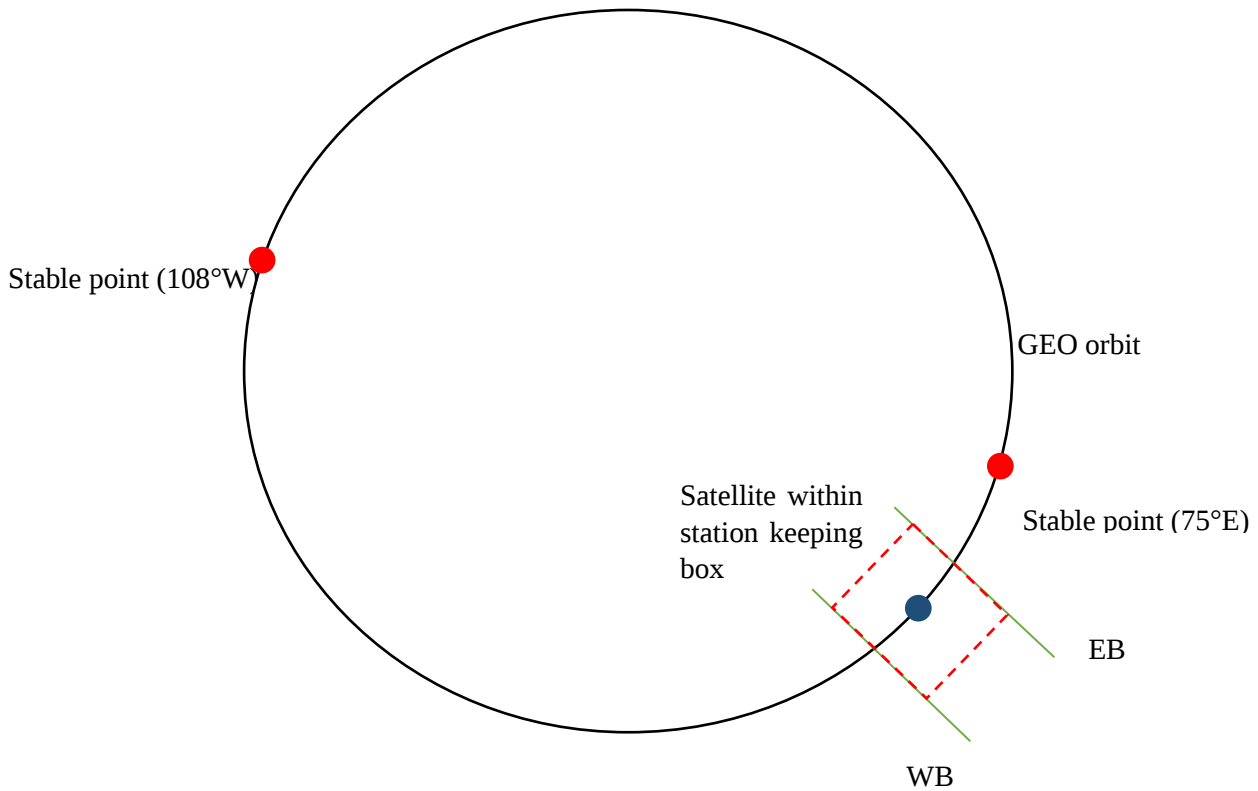


Figure 15 – E-W station keeping maneuver schematic

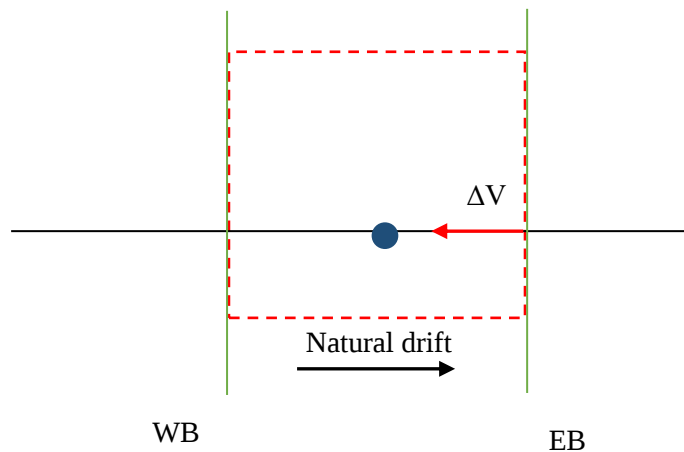


Figure 16 - E-W station keeping maneuver schematic

Referring to *Figure 15* and *Figure 16*, any geostationary satellite placed westward of an equilibrium point would be slowly accelerated, without any action, towards the closest stable equilibrium position, causing a longitude drift. The E-W control maneuver utilizes this natural motion of the satellite: that is the spacecraft is left drifting eastward until it reaches the east bound (EB) of the station keeping box; at this place thrusters are activated in order to give to the satellite a sufficient change of velocity (ΔV) to reach the western bound (WB) of the box with zero velocity. Having no residual velocity at WB, satellite starts again to drift eastward. East-West control is also used to correct the eccentricity of the orbit, which may be altered by the asymmetry of the Earth along the equator and solar radiation.

In fact, when the satellite accelerates and the semi major axis increases, the eccentricity has to be corrected with a thruster burn adding a ΔV opposite to satellite velocity. On the opposite, if the semi major axis decreases, the eccentricity has to be corrected with a thruster burn adding a ΔV to satellite velocity. All these maneuvers are performed by making thruster burns tangential to the orbit and require an overall ΔV of about 2 m/s per year, also dependent on the longitude of the satellite.

From an operative point of view, to extend the life-time of ageing geostationary spacecraft with a small quantity of fuel left, one sometimes discontinues the North-South control focusing only on the East-West control.

Generally speaking, the propulsion system deeply influences station keeping because maneuvers are defined (among other parameters) basing on its characteristics.

2.2 Collision avoidance

As mentioned at *paragraph 1.5*, collision avoidance activities begins with the analysis of orbit trajectory of space objects which may collide. It is common to identify the following parameters when considering these operations (refer to *Figure 17*):

- Primary object (PO): the space object to be protected from the impact (a satellite, for example);
- Secondary object (SO): the space object which threatens the primary object (an asteroid, for example);
- Overall miss distance (OMD): the distance between the primary and secondary objects when they move along their orbit;
- Point of closest approach (PCA): in the orbit of each object, the point where the overall miss distance is reached;
- Time of closest approach (TCA): the time at which the overall miss distance is reached.

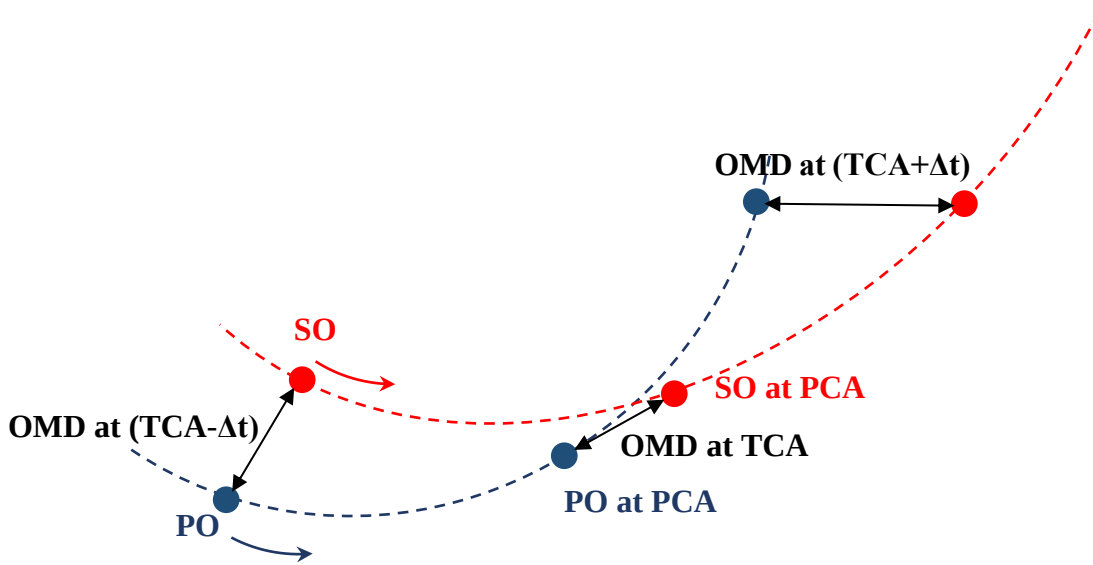


Figure 17 – Collision avoidance parameters

Once the secondary object has been detected, orbit parameters of primary and secondary object are propagated in order to define (to a certain accuracy) the risk of collision, time and position of closest approach. These parameters are then utilized to establish if a collision avoidance maneuver should be executed or not. In common operations, detection of secondary objects is limited to uncontrolled ones about 1000 km off the satellite and collision avoidance maneuvers take place if the overall miss distance is lower than few kilometers. The time interval before the OMD occurs is a critical parameter because the execution of a collision avoidance maneuver is not without impacts on satellite mission: it should be considered that the mission might be suspended or degraded and that the maneuver itself has a cost in terms of satellite fuel, which in turns may decrease system life time. Consequently, if a close approach is predicted in advance, these aspects may be optimized and managed in the optimal way.

3 Manoeuvring a GEO satellite with electrical propulsion

Conclusions of *paragraph 1.2* and *paragraph 1.5* are always valid discarding the type of propulsion considered for the satellite (chemical bi-propellant or Hall effect). However, the difference in characteristic performances such as thrust and specific impulse has implications on the station keeping both on control philosophy and from an operating point of view. As an example, it can be considered that chemical thrusters are typically fired once every two weeks during a time interval of few tens of minutes, providing forces of few tens of Newton. With electrical thrusters, in order to achieve the same station keeping objectives, it is necessary to fire the propulsion system for some hours every day, since this type of engine are capable of providing thrust in the order of hundreds of milliNewton. For this reason, the aim of this chapter is to evaluate and compare the station keeping behaviour of a reference satellite equipped, in the first case, with chemical bi-propellant thruster and, in the second case, with Hall effect thruster. A Software Tool Kit (STK) scenario has been developed in order to generate the results.

3.1 Definition of the mathematical model

A Software Tool Kit (STK) scenario with a reference GEO satellite has been developed. The considered satellite is generic and its parameters have been derived considering a typical geostationary telecommunication mission system.

Two reference system are considered, an XYZ Earth Centered Inertial (ECI) frame and a RTN frame centered in the spacecraft center of mass. In the ECI frame, Z axis is aligned to the North pole, X axis lays on the equatorial plane and Y axis gives a right handed frame. This frame is adopted to define the position of the satellite in terms of latitude and longitude and all the orbital parameters. In the RTN frame, R is orthogonal to the orbit plane, T lays in the orbit plane and is aligned to satellite movement and N is normal to the orbit plane with direction of the satellite angular momentum. A schematic of both frames is reported in *Figure 18*.

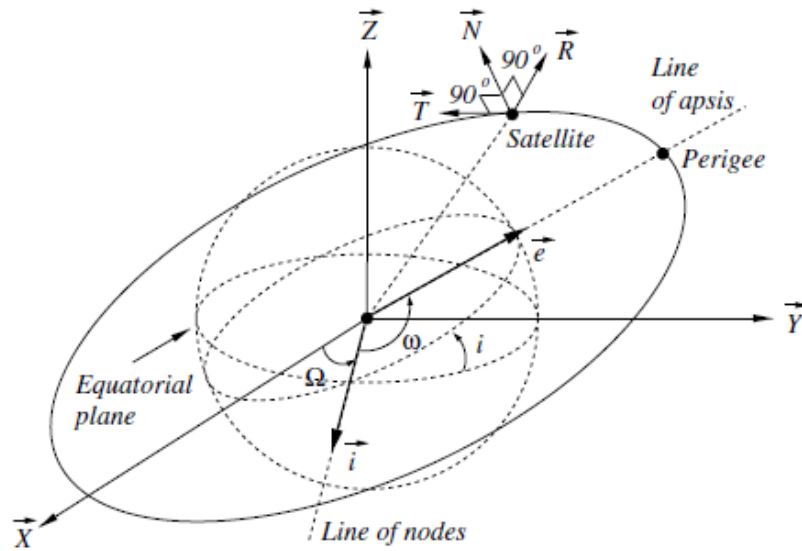


Figure 18 – Coordinate frames (Damiana Losa, 2005)

The reference satellite considered in the simulation scenario is equipped with four thrusters located on its anti-nadir face and the directions of thrust pass through the satellite center of mass, as illustrated in Figure 19.

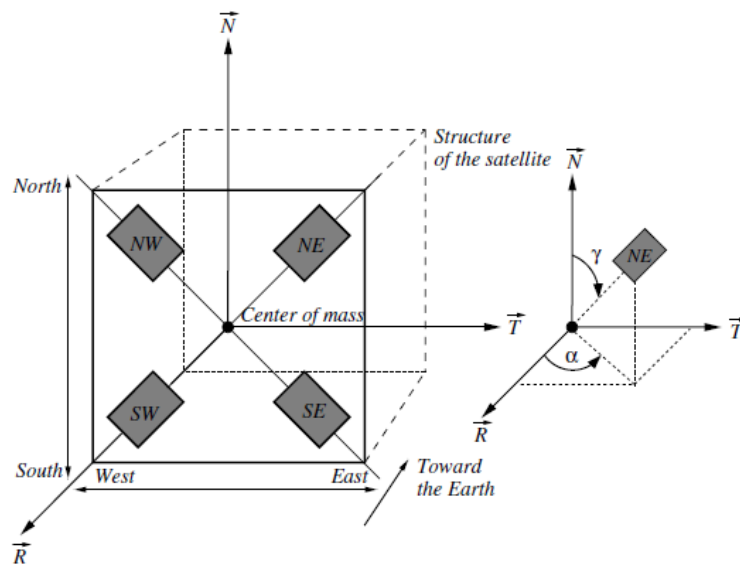


Figure 19 – Propulsion configuration (Damiana Losa, 2005)

The position of each engine is defined by the slew angle (α) and the cant angle (γ), whose values are resumed in the table below.

Engine	Slew angle (α)	Cant angle (γ)
NW	-15°	50°
NE	15°	50°
SW	-15°	130°
SE	15°	130°

Table 4 – Engine angles

These values are realistic for the current technological level.

The reference mass of the spacecraft (m_{ref}) is assumed to be 2400 kg at the beginning of the simulation and is decreased by an amount Δm given by

$$\Delta m = m_{ref} \left(1 - e^{-\frac{\Delta V}{g I_{sp}}} \right) \quad (6)$$

Where ΔV is the change in velocity after each burn of the thrusters and I_{sp} is the specific impulse of the engine. As mentioned in *paragraph 1.3* and *1.4* the specific impulse depends on the type of propulsion, whose reference values for chemical bi-propellant and Hall effect type are reported in the table below.

	Bi-propellant thruster	Hall thruster
Thrust	10 N	0.275 N
Specific impulse	300 s	1800 s

Table 5 – Reference performance for chemical and electrical propulsion

Consequently, the mass of the satellite is updated after each burn (i) according to the following equation

$$m = m_{ref} - \sum_{i=1}^n \Delta m_i \quad (7)$$

In order to take into account perturbing forces on the satellite generated by solar radiation, following parameters are considered:

- C_R – solar radiation coefficient;
- S – wet satellite surface for solar radiation (m^2);
- ϵ - mean reflectivity.

Parameter	Reference value
C_R	1.5
S	65 m^2
ϵ	0.3

Table 6 – Reference value for acceleration given by solar radiation

These parameters allow for a complete characterization of the satellite for the purpose of this work. The remaining components of the satellite such as payload and other subsystems are not considered for the development of the scenario. Although this may appear as a simplifying hypothesis, they do not contribute significantly to the characterization of the satellite motion considering the objective of this work.

Once the satellite is set up, the motion of the satellite is simulated within the STK scenario. In order to do this, the software propagates the satellite motion using an orbital propagator pre-built within the tool kit. For the chemical propulsion satellite, the orbital propagator describes the acceleration of the satellite according to a “modified” two-body equation given by

$$\ddot{\vec{r}} = -\left(\frac{GM}{r^3}\right)\vec{r} + \vec{\delta} \quad (8)$$

Where $\vec{\delta}$ accounts for the accelerations induced on the satellite by following disturbs:

- the non-spherically and inhomogeneous mass distribution within Earth;
- other celestial bodies (Sun, Moon and planets);
- Earth and oceanic tides;
- direct and Earth-reflected solar radiation pressure.

In this case propulsion forces can be considered as impulsive because their application time is much smaller than the simulation time and their contribution can be neglected in *Equation (8)*. On the opposite, this hypothesis is not realistic for the electrical propulsion satellite because the presence of a small force for a long time interval modifies the dynamic of the satellite. Consequently, for the electrical propulsion satellite the “modified” two-body equation is

$$\ddot{\vec{r}} = -\left(\frac{GM}{r^3}\right)\vec{r} + \vec{\delta} + \vec{\tau} \quad (9)$$

Where $\vec{\tau}$ denotes the acceleration induced by the thrust vector.

If the satellite was uncontrolled, it would freely propagate, losing the GEO orbit. For this reason, the scenario has been implemented with a controller that autonomously manage the position of the satellite within the station keeping box. As first step, a reference geostationary position corresponding to a longitude of 14°E has been assigned to the satellite. Subsequently, the dimension of the station keeping box have been defined, assuming a tolerance of $\pm 0,05^\circ$ either on longitude and inclination and a maximum of $\pm 0,0004$ for the eccentricity value. The reference station keeping box results as per the following schemes.

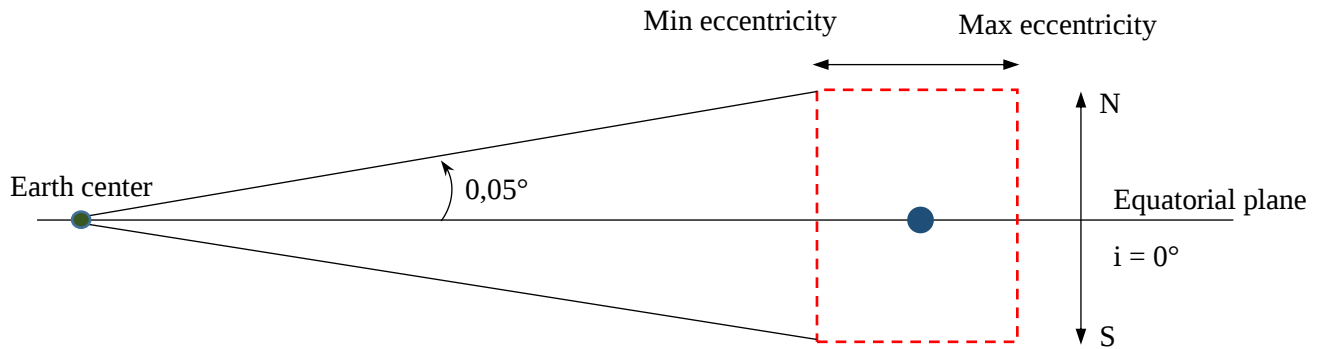


Figure 20 – Schematic of station keeping box (lateral plane)

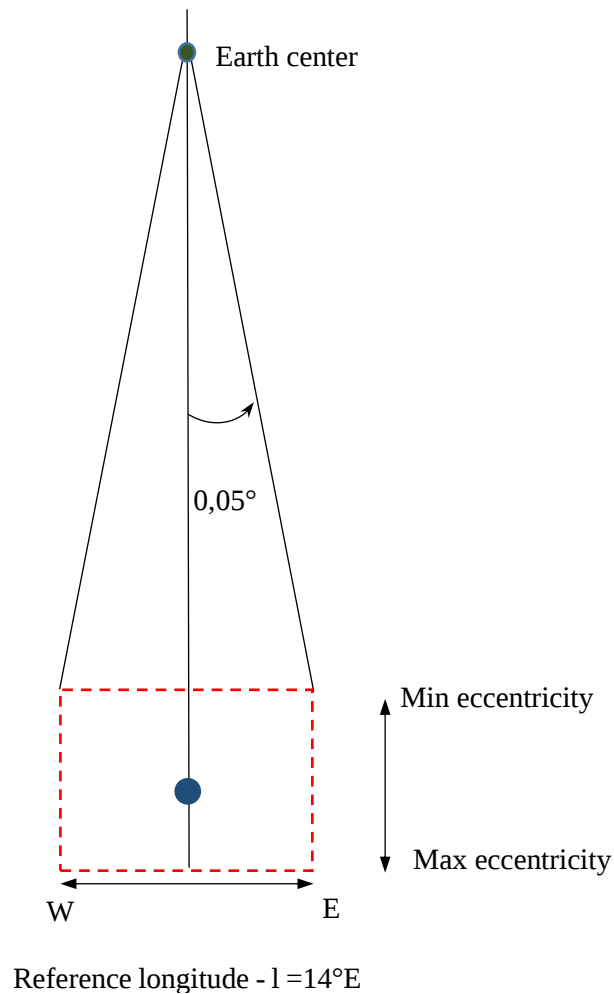


Figure 21 – Schematic of station keeping box (equatorial plane)

Parameter	Reference value	Minimum value	Maximum value	Associated side of station keeping box
Longitude	14° E	13.95° E	14.05° E	WE – 75 km
Inclination	0°	-0.05°	0.05°	NS – 75 km
Eccentricity	0	-0.0004	0.0004	Radial – 35 km

Table 7 – Reference values for satellite positioning

The control of the satellite position is performed by a controller developed within STK. Depending on the satellite model, the controller activates thrusters when the satellite approaches the boundaries of the station keeping box.

Finally, the simulation is executed for a 1 year period, starting on 1st of January 2024 at 12.00.00.

3.2 Analysis results for station keeping

Results of simulated scenarios are reported in this section. No remarkable difference can be appreciated on XY plots between HET satellite and chemical bi-propellant satellite (CBS) for the one-year-long simulation, consequently results are organized in XY plots with one-year time scale solely for the HET satellite and XY plots with two-weeks time scale for the HET and CBS satellites. For this comparison, latitude, longitude and eccentricity have been considered since requirements for the definition of the station keeping box are imposed on these parameters. The selected two-weeks period is in the middle of one-year simulation time and ranges from 1st July 2024 to 15th July 2024.

Figure 22 reports UTCG time on horizontal axis and satellite latitude on vertical axis for the HET satellite for one-year simulation time. It can be seen that, after a transient period, latitude is kept within the limit of the station keeping box during the entire duration of the simulation.

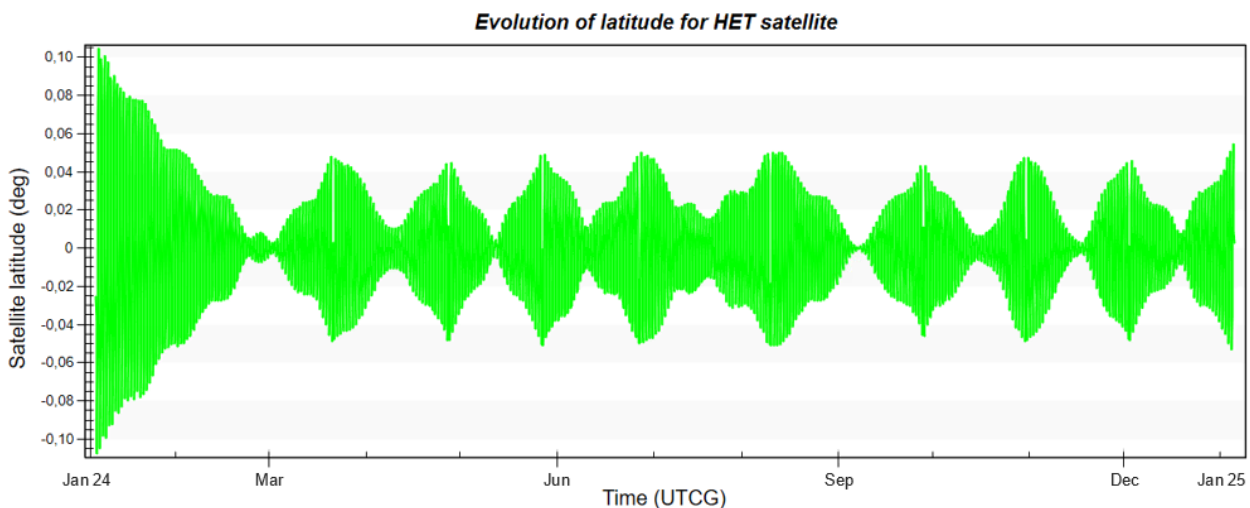


Figure 22- Evolution of latitude for electrical satellite

Figure 23 shows the same parameters with a limited time interval of two-weeks for the HET satellite and the CBS satellite: no large discrepancies can be observed in terms of station keeping performance, except for a slight difference in latitude when the N/S oscillation reaches its maximum value. Specifically, the HET satellite reverts N/S drift direction slightly before the chemical thruster satellite.

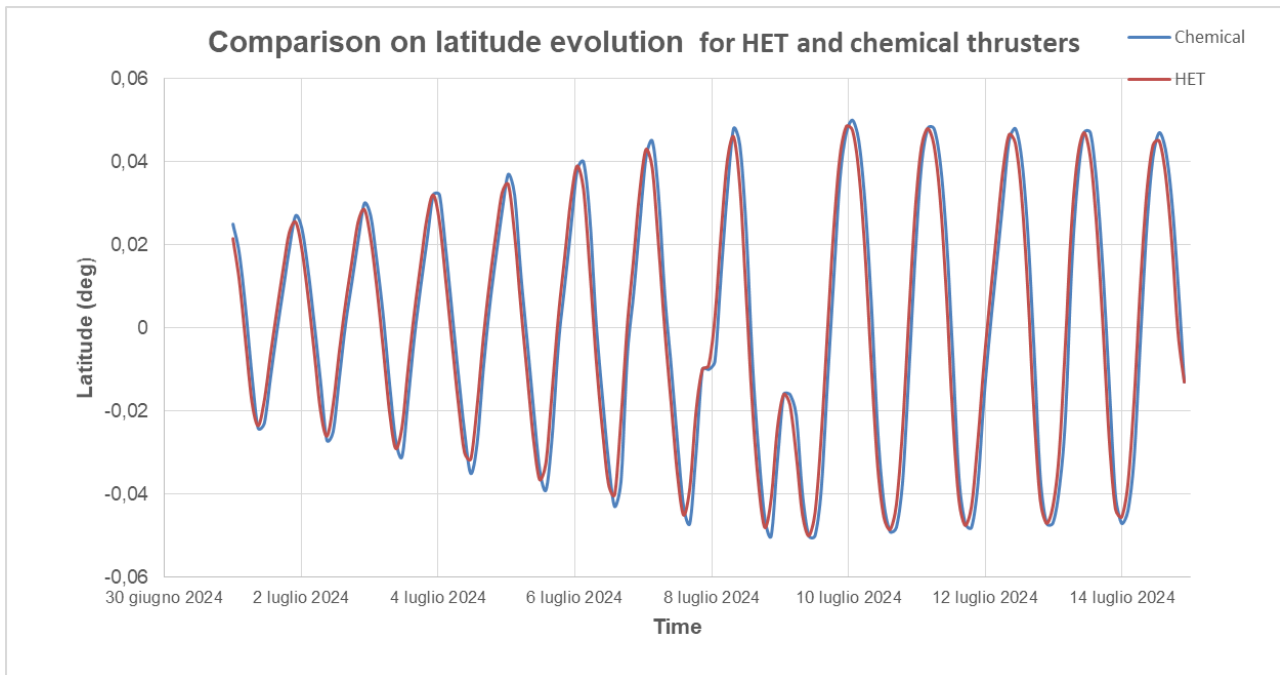


Figure 23 - Comparison on latitude evolution for HET and chemical thrusters satellites

Figure 24 reports UTCG time on horizontal axis and satellite longitude on vertical axis for the HET satellite for one-year simulation time. The graph shows that longitude is kept within the limit of the station keeping box during the one-year simulation period. The particular shape of the graph reflects the logic of the software controller, where there is a propulsion phase moving from 14.05°E to 13.95°E (so that the satellite will reach 13.95°E with no relative speed respect to Earth) and a subsequent drift phase when moving from 13.95°E to 14.05°E.

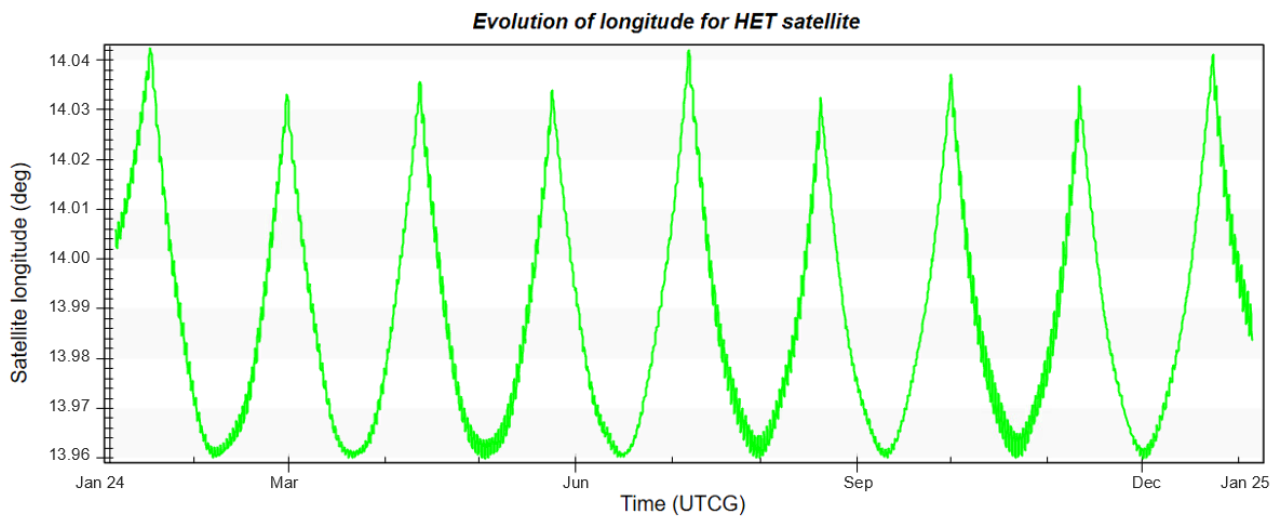


Figure 24 - Evolution of longitude for electrical satellite

Figure 25 shows the same parameters of Figure 24 with a limited time interval of two-weeks for the HET satellite and the CBS satellite. In particular, the graph focuses on the free-drift phase of satellites motion, moving from 13.95°E to 14.05°E; also in this case no large differences are remarkable, being the largest difference in latitude between the two type of satellites below 0,001°.

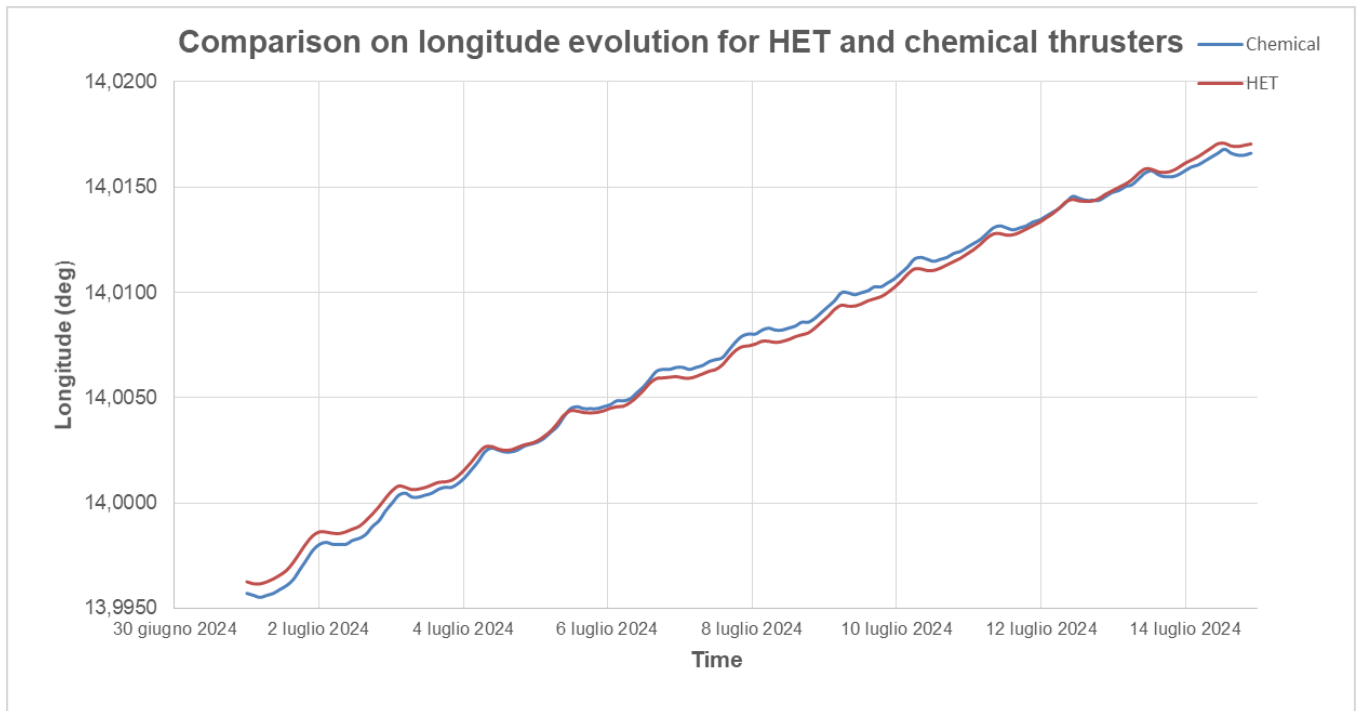


Figure 25 - Comparison on longitude evolution for HET and chemical thrusters satellites

Figure 26 reports UTCG time on horizontal axis and orbit eccentricity on vertical axis for the HET satellite for one-year simulation time. The graph shows that eccentricity is kept within the limit of the station keeping box during the one-year simulation period.

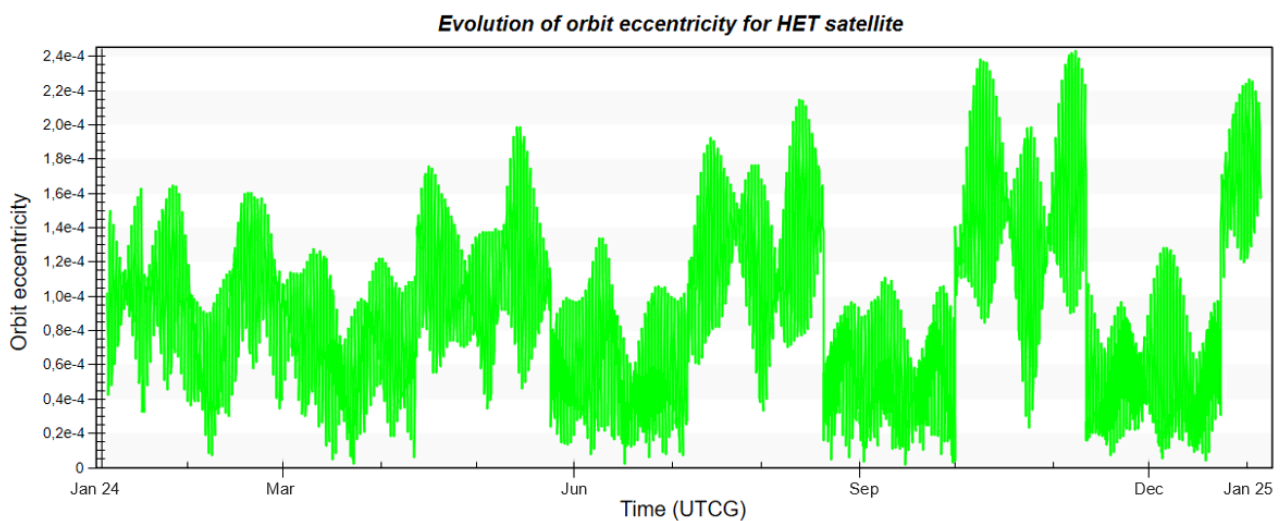


Figure 26 - Evolution of orbit eccentricity for electrical satellite

Figure 27 shows the same parameters with a limited time interval of two-weeks for the HET satellite and the CBS satellite. Basing on the graph, discrepancies are visually remarkable but the actual difference in orbit eccentricity appear to be limited since the actual difference in orbit eccentricity for the two types of satellites (occurring on 6th July 02024) is approximately about 0,000015. Additionally, the evolution of orbit eccentricity for the HET satellite appear smoother when compared to the CBS satellite, this characteristic may be caused by the prolonged thruster burns necessary for the N/S and E/W station keeping which can be contemporary used to adjust orbit eccentricity.

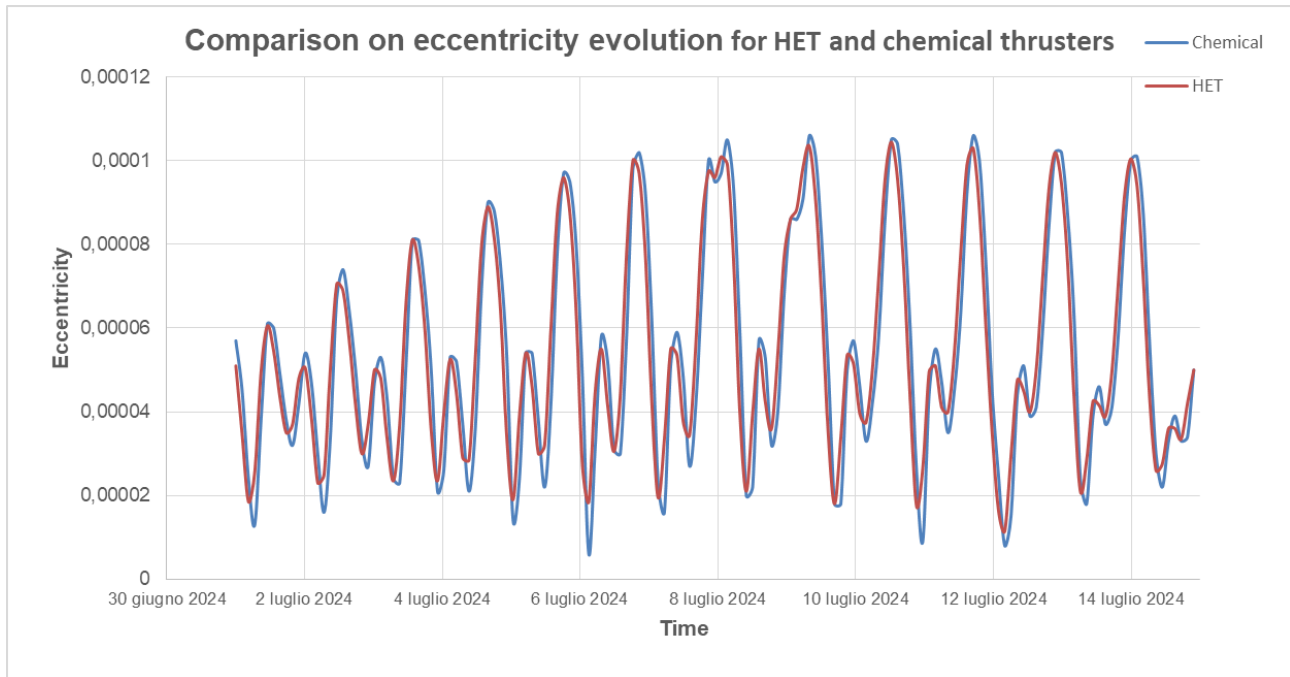


Figure 27 - Comparison on eccentricity evolution for HET and chemical thrusters satellites

Figure 28 reports UTCG time on horizontal axis and orbit inclination on vertical axis for the HET satellite for one-year simulation time. The graph shows that, after a transient period, eccentricity is kept within typical values for geostationary satellite during the one-year simulation period.

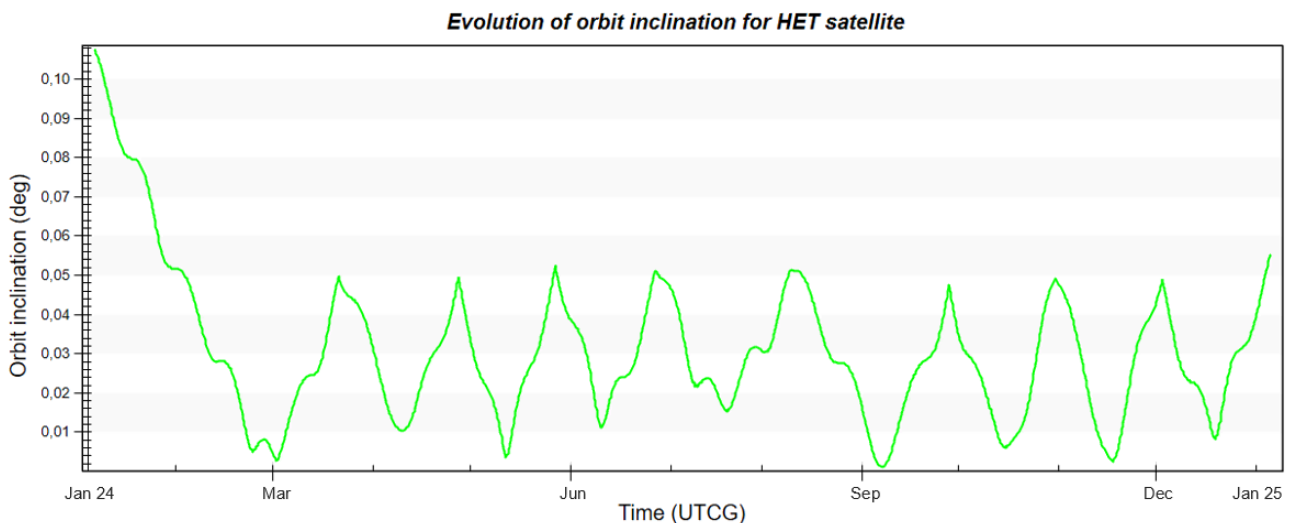


Figure 28 - Evolution of orbit inclination for electrical satellite

Subsequently, the requested ΔV , necessary to guarantee the control of the satellites over one year period, has been considered. It has to be noted that this value slightly differs among the satellite equipped with Hall effect thrusters and the satellite equipped with chemical bi-propellant thrusters because of different formulation of *Equation (8)* and *(9)*. Results are presented in the following tables; ΔV is referred to the RTN reference frame.

Hall effect thrusters satellite	
Coordinate axis	Requested ΔV
Radial (R)	1.48 m/s/yr
Tangential (T)	2.88 m/s/yr
Normal (N)	51.12 m/s/yr

Table 8 – Requested ΔV for satellite control (Hall effect)

Chemical bi-propellant thrusters satellite	
Coordinate axis	Requested ΔV
Radial (R)	1.66 m/s/yr
Tangential (T)	2.95 m/s/yr
Normal (N)	51.38 m/s/yr

Table 9 – Requested ΔV for satellite control (chemical)

Recalling *Figure 19* and considering *Figure 29* it can be seen that a thruster cannot generate a generic ΔV along a single direction. For this reason, when a thruster fires, it generates ΔV in three different directions but not all of them may be necessary for the specific maneuver; generally some of them have to be compensated by firing one of the three remaining thrusters.

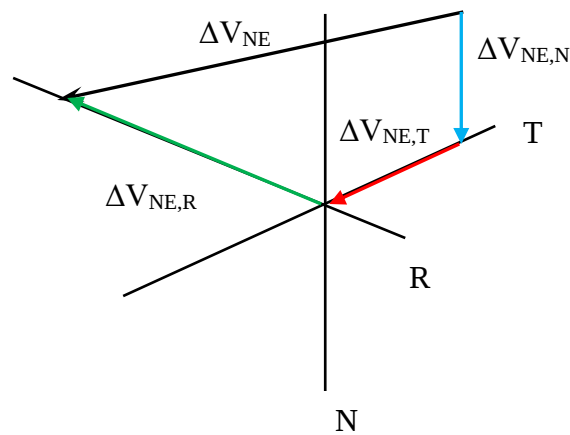


Figure 29 – ΔV for NE thruster on RTN frame

Consequently the requested ΔV have to be referred to every single thruster. These numbers, in addition to firing duration, are reported in the tables below.

Thruster	ΔV requirement		Firing duration (electrical)	
NW	20,066	m/s/yr	272468	s/year
NE	19,542	m/s/yr	265352	s/year
SW	20,787	m/s/yr	282273	s/year
SE	20,792	m/s/yr	282335	s/year

Table 10 – Thrusters ΔV requirement and firing duration (Hall effect)

Thruster	ΔV requirement		Firing duration (chemical)	
NW	20,164	m/s/yr	7529	s/year
NE	19,636	m/s/yr	7332	s/year
SW	20,896	m/s/yr	7802	s/year
SE	20,904	m/s/yr	7805	s/year

Table 11 – Thrusters ΔV requirement and firing duration (chemical)

It can be observed that the electrical propulsion requires approximately 36 times the firing duration of the chemical propulsion system to obtain the requested station keeping performances. This is determined by the much lower amount of thrust the electrical thrusters can provide.

Referring to a 1-day-time-interval, the averaged firing duration of each thruster is reported in the following tables.

Hall effect thrusters				
Thruster	Firing time		Average frequency	
NW	272468	s/yr	12,44	(min/day)/yr
NE	265352	s/yr	12,12	(min/day)/yr
SW	282273	s/yr	12,89	(min/day)/yr
SE	282335	s/yr	12,89	(min/day)/yr

Table 12 – Average firing duration for electrical thruster

Chemical bi-propellant thrusters				
Thruster	Firing time		Average frequency	
NW	7529	s/yr	0,34	(min/day)/yr
NE	7332	s/yr	0,33	(min/day)/yr
SW	7802	s/yr	0,35	(min/day)/yr
SE	7805	s/yr	0,35	(min/day)/yr

Table 13 – Average firing duration for chemical thruster

Considering fuel consumption, the electrical propulsion, thanks to a high specific impulse, requires much less fuel than the chemical type. Results for one-year simulated scenario are reported below.

Fuel mass consumption		
Hall effect thrusters	11.1	kg/yr
Chemical bi-propellant	65.3	kg/yr

Table 14 – Fuel mass consumption

3.3 Considerations

Basing on the analysis performed, Hall effect thrusters appear to be much more efficient in terms of fuel consumption thanks to the high specific impulse: this may allow for a global optimization of the mission design process since the fuel mass saved is larger than 800 kg if considering a typical GEO mission of 15 years. From a system point of view, it should also be considered that electrical propulsion requires ancillary system for electrical power generation and stabilization such as dedicated solar array and propulsion power units. This reduces the overall mass saving that the low fuel consumption may allow.

From an operational point of view, Hall effect thrusters require a longer firing time respect to the chemical bi-propellant propulsion system. As can be seen from *Table 15* and *Table 16*, the average firing frequency (on year basis) for Hall effect thrusters is approximately 13 minutes every day for each thruster, whilst for the chemical bi-propellant engine the average frequency is about 20 seconds per day. Consequently, basing on operating experience, the operation frequency can be estimated in 2 days for the electrical thrusters and 14 days for the chemical ones. This lead to an overall firing times of 101 minutes and 19 minutes respectively.

Hall effect thrusters						
Thruster	Firing time		Average frequency		Summary	
NW	272468	s/yr	12,44	(min/day)/yr	Operation frequency	
NE	265352	s/yr	12,12	(min/day)/yr	2	days
SW	282273	s/yr	12,89	(min/day)/yr	Overall firing time	
SE	282335	s/yr	12,89	(min/day)/yr	101	minutes

Table 15 – Operations frequency for Hall effect thrusters

Chemical bi-propellant thrusters						
Thruster	Firing time		Average frequency		Summary	
NW	7492	s/yr	0,34	(min/day)/yr	Operation frequency	
NE	7296	s/yr	0,33	(min/day)/yr	14	days
SW	7761	s/yr	0,35	(min/day)/yr	Overall firing time	
SE	7763	s/yr	0,35	(min/day)/yr	19	minutes

Table 16 – Operations frequency for chemical thrusters

These values are in line with numbers proposed in literature and it is evident that the management of a spacecraft with Hall effect thrusters, and in general with electrical propulsion, requires a change of paradigm in the conduct of operations. In fact, performing long positioning manoeuvres arises numerous problems to address: first, the service provided by the satellite may need to be suspended, or at least limited, reducing the overall satellite availability; then, each manoeuvre requires a long planning phase because a great number of systems should be checked and some preliminary activities may need to be executed (i.e. health checks, switch between nominal and redundant apparatus...); considering that N/S station keeping manoeuvre should be performed at the ascending or descending node, a prolonged thruster fire need to be started before reaching the node and terminated after passing it and this may add additional difficulties. Additionally, operating constraints shall be considered, in fact, thrusters generally cannot be fired during eclipses due to the large amount of required electrical power (usually in the range of 5-7 kW for the class of satellite considered) and may have additional limitation such as minimum time interval among two consecutive fires or simultaneous firing of different thrusters. Furthermore, there may be cases where satellites (generally two) share the same orbital slot, this impose more stringent constraints on tolerances for station keeping and may require very frequent manoeuvres for the satellites. Finally, personnel availability and, possibly, limited personnel resources should be taken into account. These difficulties require some new techniques to be introduced for the successful management of space operations. First, satellite manufacturers may implement autonomous control system that may allow for a quasi-non-supervised satellite operation: in this way the effort on operators should be greatly reduced. On the other hand, satellite operators may adopt lean-management strategies, limiting intervention on satellite to most critical cases and to periodic overall check.

4 Outcomes

A generic telecommunication geostationary satellite in the class of 2500 kg has been considered and a comparison has been conducted for station keeping manoeuvres, considering the installation of Hall effect thrusters and chemical bi-propellant thrusters as propulsion system. Results of this comparison have been generated running a simple simulation on AGI Software Tool Kit (STK) scenario developed for this purpose. Obtained results are in line with values proposed by literature and have demonstrated that station keeping performance do not differ significantly for the two types of satellite but manoeuvres require an overall firing duration of few tens of minutes every fourteen days for the chemical satellite and of approximately two hours every two days for the HET satellite. This requires a change of strategy in the management of the space asset which can be obtained by increasing the level of control autonomy of the satellite and by adopting lean management techniques by satellite operators.

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