Politecnico di Milano

Master of Science in
Space Engineering

Orbital Mechanics Project

Group 33

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Nomenclature

$\mu_{\odot}$	Gravitational Parameter of the Sun
$\mu_{\oplus}$	Gravitational Parameter of Earth
Ω	Right ascension of the ascending node
ω	periapsis anomaly
a	semi-major axis
AU	Astronomical Unit ($1.495978707 \times 10^8 km$)
e	eccentricity
f	true anomaly
i	inclination
r	radius
h	specific angular momentum per unit mass
mjd2000	modified julian date 2000

Contents

Assignment 1: Interplanetary Explorer Mission

Introduction

In this chapter, the aim is to design an interplanetary trajectory from Neptune to Mercury with a flyby of Venus. The mission is to be completed between the dates February 1st 2031 and February 1st 2071.
Figure 1 a and b – given orbits
The design process is composed of different consecutive steps:

1. Define the time windows for possible transfer phases (departure, gravity assist and arrival);
2. Set known parameters and compute the orbits of the three planets of interest;
3. Set and solve the Lambert's problem between first Neptune and Venus and then Venus and Mercury at different times inside the possible time windows;
4. Evaluate the powered gravity assist for every trajectory previously computed;
5. Selection of the most efficient transfer strategy.

1.1 Method of patched conics

The movement of a body inside the interplanetary space cannot be described by the two-body problem since the body itself is subject to the influence of different celestial bodies at once. In order to study such movement, an approximation is introduced with the method of the patched conics. With this approximation, the influence of a planet or a gas giant is limited to a sphere named Sphere of Influence (SOI from now on) which radius is calculate using the following equation:

$$R_{SOI} = R_{CB} \left(\frac{m_{CB}}{m_{\odot}} \right)^{\frac{2}{5}}$$

Where R_{CB} is the distance between the celestial body and the sun, m_{CB} is the mass of the central body and $m_{\odot}$ is the mass of the sun.
If the body is outside any planet's or gas giant's SOI, then it is considered to be subject only to the Sun's gravity attraction.
Figure 2 – patched conics
In this report, the spacecraft is considered to be already outside of Neptune's SOI at departure and it does not enter Mercury's SOI at arrival. The spacecraft only enters Venus' SOI during the flyby.

Mission analysis

2.1 Design process

2.1.1 Time windows

The Lambert's problem solution is a section of an elliptical orbit. Furthermore, the minimum time needed to complete a transfer strategy is the time required to run across an elliptical orbit with eccentricity $e \to 1$.

Therefore, given the parabolic times of flight from Neptune to Venus and from Venus to Mercury, it is possible to define the earliest possible date of flyby and arrival starting from the departure date. In particular the earliest possible flyby and arrival dates can be defined as follows:

$$Fly_{min} = Dep_{min} + ToF_{NV}$$

$$Arr_{min} = Fly_{min} + ToF_{VM}$$

$$Fly_{max} = Arr_{max} + ToF_{VM}$$

Where ToF_{ij} is the parabolic time of flight between the planets i and j. Similarly, given the latest arrival date it is possible to define the latest flyby and departure dates:

Figure 3 – time windows

Said dates are computed using the timeWindows script and they are:

$$Dep_{max} = Fly_{max} + ToF_{NV}$$

Fly_{min}	20-04-2043
Arr_{min}	28-04-2043
Dep_{max}	06-11-2058
Fly_{max}	23-01-2071

2.1.2 Additional constraints considered

One additional constraint that was to be considered is the minimum altitude of the perigee point for the planetocentric hyperbola at Venus flyby. We chose this as 300 km, giving us a minimum radius of pericenter of 6351.8 km. For the MATLAB fmincon optimization function this was set using an additional constraint function and also time constraints based on the parabolic time of flights were set using the inequality matrices and simple checks in the coarse grid search.

2.1.3 Transfer options exploration, analysis and comparison

For the selection of an optimum transfer we employed different strategies together. To analyze the solutions we built an objective function giving us the ΔV needed for the transfer given the *arrival*, *gravity assist* and *departure* dates inserted to the function. The function computes the two arcs connecting planets at departure and flyby and the flyby planet to the arrival planet. The outputs consist of the total ΔV required, then the ΔV's for the separate legs and the informations for each transfer arc as an object with various fields useful for the rest of the analysis and plotting.

```
compute and store first arc
compute and store second arc
join them with flyby
output total DV
```

Listing 2.1: GAtransfer objective function structure

Coarse grid search

The first step is a coarse grid search using three nested for loops:

```
parfor i = 1 :dep_size
    for j = 1:ga_size
        tof_1 = gravityassist(j) - departure(i);
        for k = 1:arr_size
            tof_2 = arrival(k) - gravityassist(j);
            %% Set NaN for TOFs<parabolic TOF and rp < rpmin
            if  tof_2 < t_p_VM || tof_1 < t_p_NV
                DV(i,j,k) = NaN;

            else
                [DV(i,j,k), ~, ~, ~, FLYBY,~,~]= GAtransfer(
    Neptune,Venus,Mercury,departure(i),gravityassist(j),arrival
    (k));
                if FLYBY.rp < rpmin
                    DV(i,j,k) = 1000000;        % Set an
    impossibly high value
                end
```

```
            end
        end
    end
end

[dv_min_grid,loc] = min(DV(:));
[ii,jj,kk] = ind2sub(size(DV),loc);

x_GRID = [departure(ii); gravityassist(jj); arrival(kk)];
```

Listing 2.2: coarse grid search

For these loops we employed a parfor outer loop to take advantage of the parallel processing capabilities of MATLAB, using all 4 cpu cores. This allowed us to speed up the grid search substantially, now taking only 500s with a size of $2.7e6$ elements.

fmincon refinement

The objective function GAtransfer can also be used for the fmincon MATLAB optimization function, which finds the minimum for a constrained nonlinear function using a gradient-based optimizer. This function takes as initial guess the minimum point found from the coarse grid search as a vector of 3 mjd2000 dates, the linear constraint on the windows with two **ub** and **lb** vectors, the time of flight constraints with the **A** and **b** inequality matrices and the *nonlinear constraint* on the minimum pericenter radius of the hyperbola as a function flyby_CONSTR

Other strategies explored

We also explored using the *genetic algorithm* ("ga" MATLAB function) to find an initial guess to then refine further using fmincon, making use of the parallel computing toolbox to us all the 4 CPU corse of our CPU simultaneously; however due to heuristic algorithm this gave us worse and often inconsistent results between successisve runs of the program.

2.1.4 Selection of the final solution

The final solution was selected as the one given from the coarse grid search plus the fmincon refinement as this gave us a lower total ΔV. By narrowing the windows we also found other shorter transfers but with an higher total ΔV so we decided to discard them.

2.2 Final solution

2.2.1 Heliocentric trajectory

The final transfer strategy is composed of four arcs: the first and the last one lie inside the Sun's SOI; the second and third one are inside Venus' SOI (and are analyzed in a later paragraph). The Transfer strategy is completed between the times 02:54:32 on June 27th 2032 and 03:09:55 on July 11th 2070. The powered flyby is performed at 13:49:55 on July 11th 2061.
Figure - complete strategy
The first arc connects Neptune and Venus orbits. It is characterised by the following data:

a [AU]	**e [-]**	**i [°]**	**RAAN [°]**	ω **[°]**
15,1835	0,9487	1,5798	109,6640	85,2838

The initial position is characterised by a true anomaly (f) of 180,2552°. In Cartesian coordinates the initial point is identified by the following data:

$\mathbf{R}_x$ **[AU]**	$\mathbf{R}_y$ **[AU]**	$\mathbf{R}_z$ **[AU]**
28.8260	7.8326	-0.8213
$\mathbf{v}_x$ **[km/s]**	$\mathbf{v}_y$ **[km/s]**	$\mathbf{v}_z$ **[km/s]**
-0.3810	0.8955	0.0016

The initial position's distance from the Sun is 29,8825AU and the velocity at the initial point is 0,9732 km/s. The final position is characterized by a true anomaly (f) of 94,7162°. In Cartesian coordinates the initial point is identified by the following data:

$\mathbf{R}_x$ **[AU]**	$\mathbf{R}_y$ **[AU]**	$\mathbf{R}_z$ **[AU]**
0.3421	-0.9573	0
$\mathbf{v}_x$ **[km/s]**	$\mathbf{v}_y$ **[km/s]**	$\mathbf{v}_z$ **[km/s]**
36.6802	-18.4610	-0.7813

The initial position's distance from the Sun is 1,0166 AU and the velocity at the initial point is 41,0714 km/s. The fourth arc connects the point where the spacecraft exits the Venus' sphere of influence with Mercury.

2.2.2 Powered gravity assist

2.2.3 Cost of the mission

Assignment 2: Planetary Explorer Mission

Mission requirements

In this chapter we are going to explain how we have designed a planetary explorer mission to perform Earth observation: we were in fact required to analyse the Earth-centred orbit characterised by the values in the table below and estimate its ground track, studying the effects of the assigned orbit perturbations by integrating both Gauß's planetary equations and Cartesian equations and subsequently comparing the results of the two methods. After the characterisation of the ground track, we propose a modification of the orbit aimed to get a ground track which repeats itself once a sidereal day. For the sake of simplicity, we have chosen a RAAN of 0 degrees.

a [$10^4 km$]	**e**	**i** [deg]	**hp** [km]
4.0718	0.6177	78.2195	15566.491
Repeating GT ratio k:m^{note}	**Perturbations**	**Parameters**	
1:1	J2 and SRP	cR = 1.2	A/m = 4.000 m^2/kg

Note: the ground track repeats itself **k** times every **m** revolutions of the planet.

Table 3.1: Mission requirements

The orbit we've been assigned, represented in the following images, is almost geosynchronous: in fact it has a period of $22h42min49s$, which is very close to a sidereal day. With a perigee height of 9195.5 km and an apogee height of 59498.5 km, the orbit is very eccentric and, since its inclination is very high, it would be suitable for a communication satellite designed to service high-latitude regions, as the satellite spends most of the time in the northern hemisphere (apogee dwell). Due to the fact that even at the perigee the atmospheric drag is negligible, the assigned perturbations should be sufficient to model the orbit with a good level of accuracy.

Mission analysis outputs

4.1 Ground track

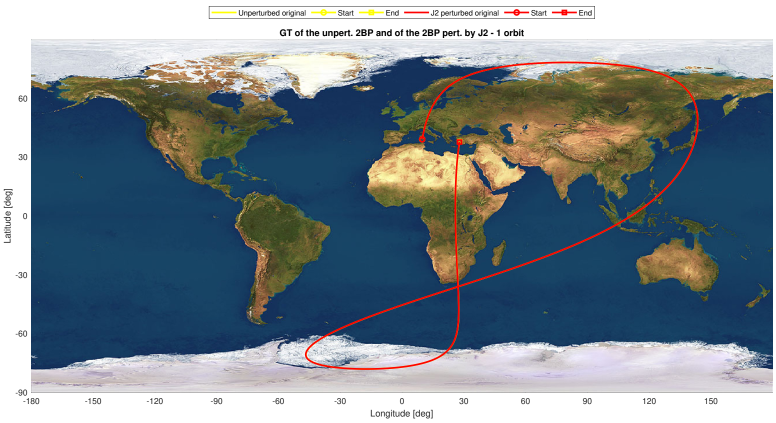


Figure 4.1: GT of the unpert. 2BP and of the 2BP pert. by J2 - 1 period

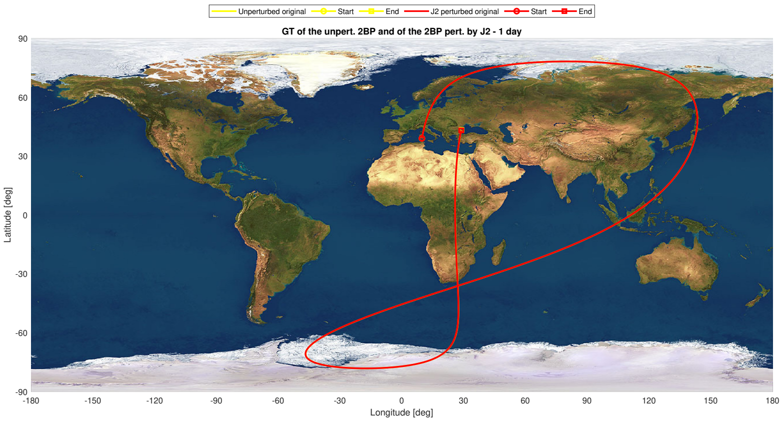


Figure 4.2: GT of the unpert. 2BP and of the 2BP pert. by J2 - 1 day

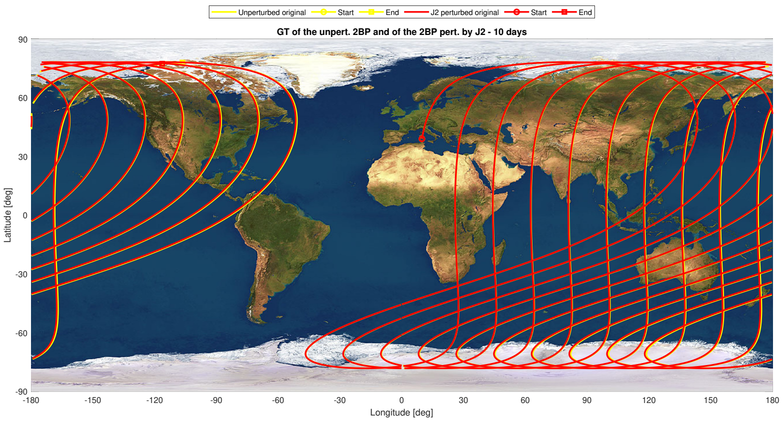


Figure 4.3: GT of the unpert. 2BP and of the 2BP pert. by J2 - 10 days

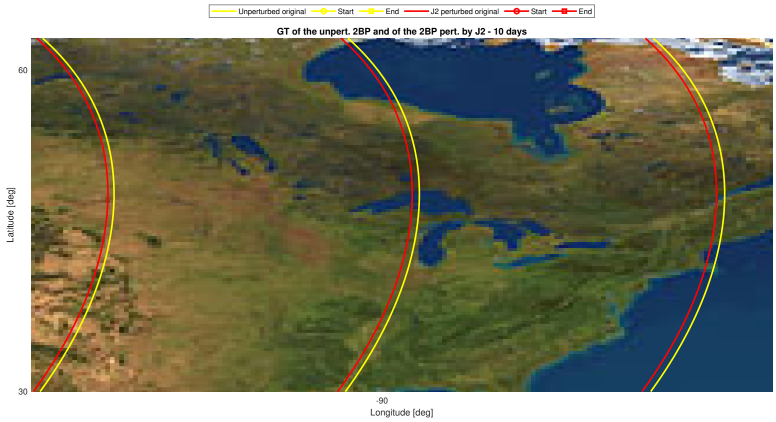


Figure 4.4: GT of the unpert. 2BP and of the 2BP pert. by J2 - 10 days, close-up

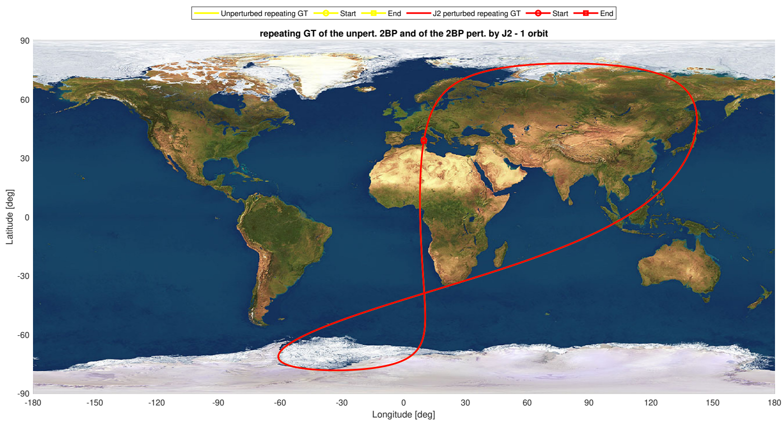


Figure 4.5: repeating GT of the unpert. 2BP and of the 2BP pert. by J2

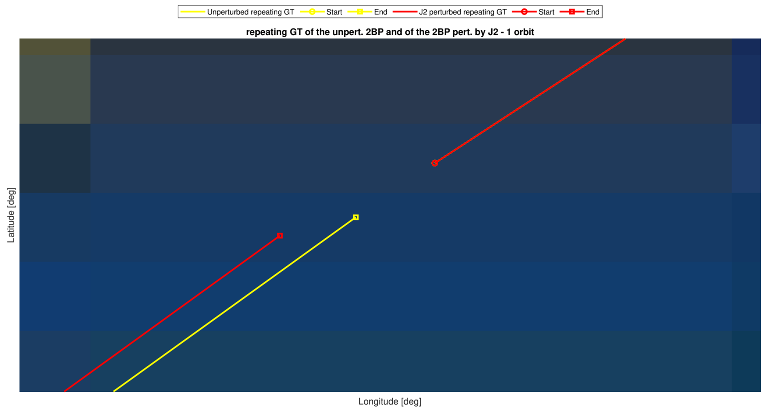


Figure 4.6: repeating GT of the unpert. 2BP and of the 2BP pert. by J2, close-up

To modify the semi-major axis in order to obtain a geosynchronous orbit, for the unperturbed case we simply calculated the semi-major axis which corresponds to a period of a sidereal day, whereas when taking into account the second zonal harmonic perturbation we included the modification of the nodal periods of satellite and Earth due to the Secular effects of J2: the regression of the nodes in the short term is therefore visible when comparing the ground tracks. In the first case we obtained a semi-major axis of 42166.2 km, while in the second one we got $a = 42163.6$ km

4.2 Orbit Perturbations

In our model we included perturbations due to two effects:

- **Second Zonal Harmonic *J2***

  Models the Earth Oblateness using the spherical geopotentials model, truncated to the second term.

- **Solar Radiation Pressure *SRP***

  A force given by the impact of momentum-carrying photons on the surfaces of the spacecraft. For this perturbation we have written a function that, based on the initial date gets the position vector of the Earth from the ephemerides function and uses it to calculate the direction of the disturbing acceleration (given the large difference in scale we approximated the position vector of the spacecraft with respect to the Sun with the heliocentric position vector of the Earth) . A simple algorithm to check for the eclipse condition using the position vector of the spacecraft found in [4] has been implemented.

$$\mathbf{a_{SRP}} = -P_{SR@1AU}\frac{AU^2}{||\mathbf{r}_{sc-Sun}||^3}c_R\frac{A_{Sun}}{m}\mathbf{r}_{sc-Sun} \tag{4.1}$$

These forces give an acceleration to the spacecraft so that the total acceleration it perceives in orbit is

$$\mathbf{a} = -\frac{\mu_\oplus}{r^3}\mathbf{r} + \mathbf{a}_{SRP} + \mathbf{a}_{J2} \tag{4.2}$$

4.3 Orbit Propagation

4.3.1 Methods

To propagate the orbit we used two different methods:

- **Gauss Planetary Equations**

$$
\begin{aligned}
\frac{da}{dt} &= \frac{2a^2}{h}\Big(e\sin f\ a_r + \frac{p}{r}a_s\Big)\\
\frac{de}{dt} &= \frac{1}{h}\Big(p\sin(f)\ a_r + \Big((p+r)\cos f + re\Big)a_s\Big)\\
\frac{di}{dt} &= \frac{r\cos(f+\omega)}{h}a_w\\
\frac{d\Omega}{dt} &= \frac{r\sin(f+\omega)}{h\sin i}a_w\\
\frac{d\omega}{dt} &= \frac{1}{he}\Big(\ \cos f\ a_r + (p+r)\sin f\ a_s\Big) - \frac{r\sin(f+\omega)\cos i}{h\sin i}a_w\\
\frac{df}{dt} &= \frac{h}{r^2} + \frac{1}{eh}\Big(p\cos f\ a_r - (p+r)\sin f\ a_s
\end{aligned}
$$

  We numerically integrate these equations with the ode113 solver, by using them to set the derivatives of the state. The reference frame for this equations is the RSW frame so a_r, a_s, a_w are respectively the radial, transversal and out-of-plane components. [1] [2] [3]

- **Numeric integration of cartesian equations** This method consists in directly integrating the Cartesian equations of motion

$$\ddot{\mathbf{r}} = -\frac{\mu_\oplus}{r^3}\mathbf{r} + \sum a_p \tag{4.3}$$

4.3.2 Comparison

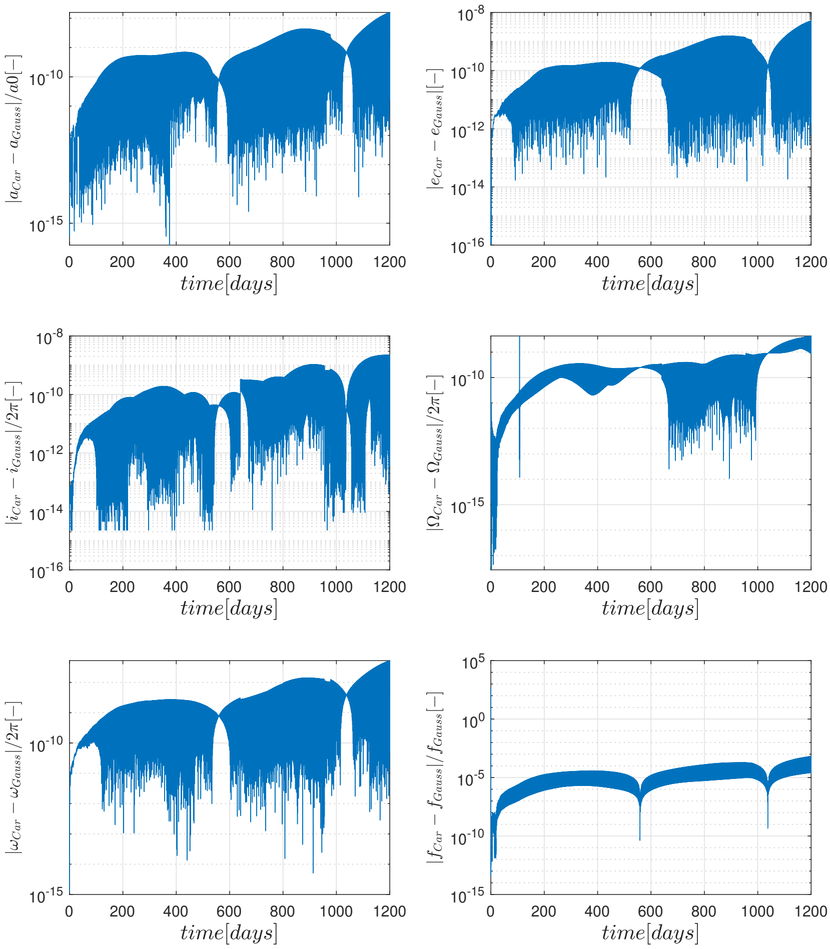
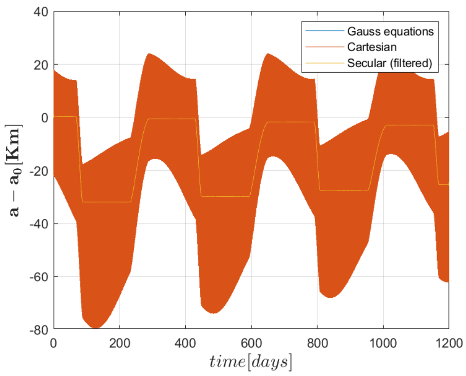


Figure 4.7: error between Gaussian and Cartesian propagation methods (log scale)

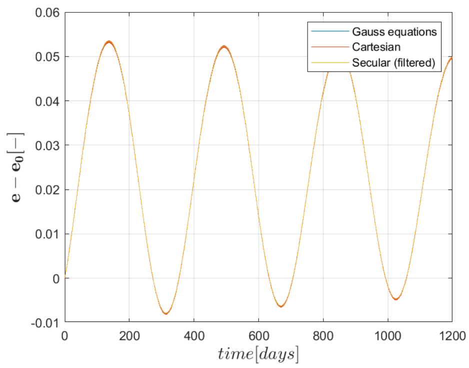
There are numerical error introducing differences between the two methods, but the error only grows up to an order between 10^{-10} and 10^{-8} for a propagation of 1200 days, except for the true anomaly which grows to a still small error of 10^{-3}.

4.4 History of the Keplerian elements

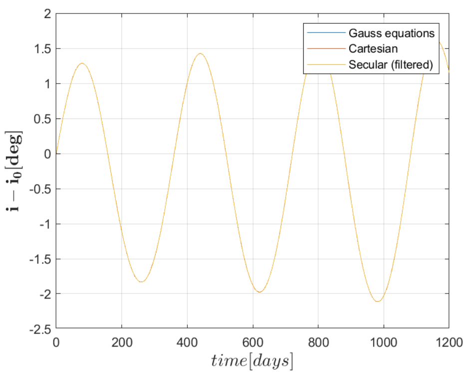
The Keplerian elements are plotted here for 1200 days from 2021-01-01



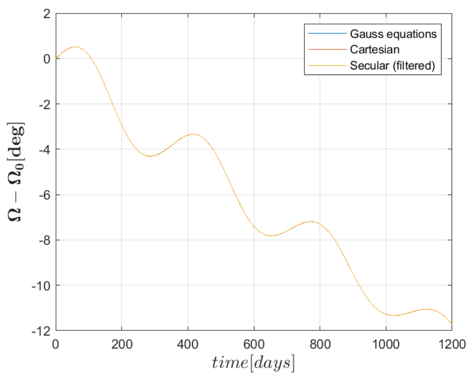
(a) semi-major axis



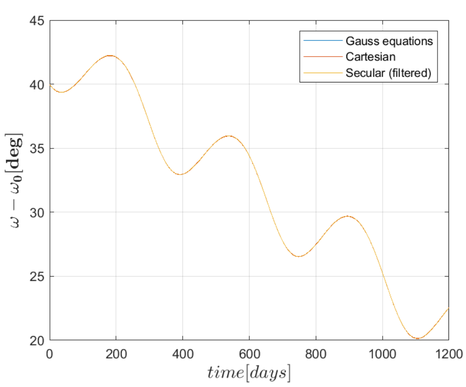
(b) eccentricity



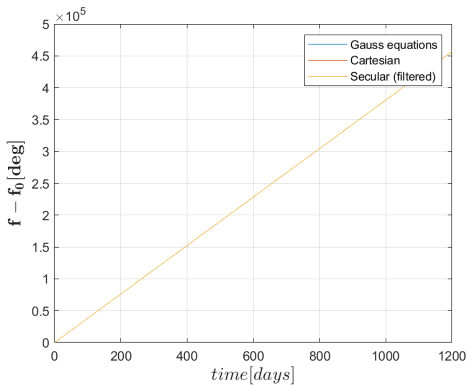
(c) inclination



(d) right ascension of ascending node



(e) perigee anomaly



(f) true anomaly

Figure 4.8: Evolution of keplerian elements

4.5 Orbit evolution representation

4.6 HF filtering

For the filtering we have used the simple *movmean* function integrated in MATLAB which computes the average at each time instant between the current point and the neighboring ones with a cutoff period of $3T_{orbit}$.

We can see in the *secular evolution* that there is a superposition of effects from the J2 perturbation and the SRP:

- a : there is a slight secular decrease in the semi-major axis due to the effect of the SRP
- Ω : there is a *nodal regression* due to the inclination i being less than $90°$ for the J2 effect
- ω : there is a negative *perigee precession* for the J2 effect due to the inclination $i > 63.4°$ giving a so-called *"Hula-hooping effect"*
- e : there are periodic oscillations with a secular increase caused by ovalization of the orbit by the SRP

4.7 Comparison with real data

4.7.1 Satellite selection

We searched for a satellite in the same orbital region on celestrack.org website and we found a satellite with the same inclination and eccentricity as ours. The satellite is NASA's Global Geospace Science (GSS) Polar satellite. This satellite was operative from 24th of February 1996 to 28th of April 2008 with the following parameters :

a [km]	**e [-]**	**i [deg]**	**RAAN [deg]**	ω **[deg]**
35490.94	0.701992	78.63	260.60	306.12

4.7.2 Comparison with our model

To get a significant time span, we resembled operative years from first of January 2000 to 28 of April 2008 in our model. By setting the initial conditions in the model as that of the real satellite, we got the following results:

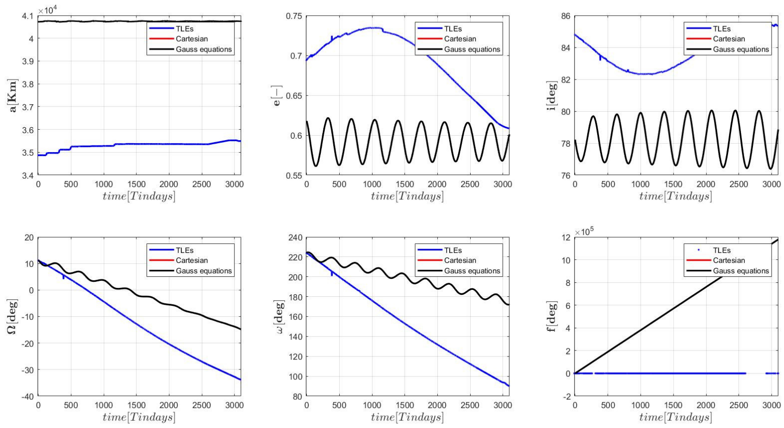


Figure 4.9: comparison with our model

Bibliography

[1] Curtis, H.D. *Orbital mechanics for engineering students.* Butterworth-Heinemann , 2014. Chapter 12

[2] Vallado, D.A. *Fundamental of Astrodynamics and Applications, 4th Ed*, Microcosm Press, 2013. Chapters 8 and 9

[3] Battin, R, *An Introduction to the Mathematics and Methods of Astrodynamics*, AIAA Education Series, 1999. Chapter 10

[4] Curtis,H.D. *Orbital mechanics for engineering students.* Butterworth-Heinemann , 2019. Chapter 10.9