Electric Propulsion for Small Satellites: A Case Study

IEPC-2019-418

Presented at the 36th International Electric Propulsion Conference
University of Vienna • Vienna, Austria
September 15-20, 2019

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This publication presents the operations of a small satellite using electric propulsion with a focus on the propulsion requirements and the mission planning. The study is done using Exotrail's mission design software ExoOPS™ – *Mission Design*. Two constellation deployment scenarios will be discussed: an orbit transfer following a rideshare launch and the adjustment of the local time of the ascending node, to spread the satellites onto multiple planes. The results will be presented to match an operator's point of view. Relevant performance indices that will be compared include time-to-orbit, average power consumption, energy consumption, lifespan remaining after the deployment, propellant consumption, number of thruster firings. A parametric analysis is performed, in order to consider several values of satellite mass, power and volume, as well as different propulsion technologies. Finally, the results are discussed in terms of their impact on the economics of the mission. The duration of the constellation deployment which impacts both the time before revenues and the lifetime remaining after the deployment, the number and duration of the firing sequences, which impacts the availability of the spacecraft during the station-keeping, and the mass of the different solutions are reported.

Nomenclature

FEEP	Field Emission Electric Propulsion
HT	Hall Thruster
RAAN	Right Ascension of the Ascending Node
SSO	Sun-Synchronous Orbit

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I. Introduction

THE “New Space” era, with new actors in the space industry and a rapid development of new, riskier technologies is now a reality. The multiplicity of new small satellites and constellations projects paves the way for new designs and qualification methods. The next frontier in the small satellites market is the development of adapted propulsion systems. Indeed, propulsion allows for both a more flexible launch strategy and an augmented lifespan of the satellite. The former can be obtained using rideshare launches followed by a maneuver to the operational orbit, while the latter results from an efficient station keeping strategy. Currently, electric propulsion technologies, other than Hall thrusters, still suffer from a low thrust to power ratio, making them unsuitable for maneuvers such as orbit raising and local time phasing. On top of this, the operation of a small satellite equipped with electric propulsion is still a not widely covered topic. Newcomers to the market might see the possibility of efficiently deploying their small-sat constellation hampered by a lack of knowledge and efficient tools.

To address these issues Exotrail develops both an efficient propulsion system for small satellites and a software to design, optimize and operate the maneuvers to perform the mission – ExoOPS^{TM} – *Mission Design* and ExoOPS^{TM} – *Mission Operation*. This approach enables the company to understand the operational needs of customers in order to tailor specific solutions adapted to their requirements.

This publication presents two mission scenarios related to the deployment of a constellation of small satellites using electric propulsion: both demonstrate the advantages of having an efficient propulsion system. The first scenario shows that rideshare launches can be exploited to reduce the costs by launching to an orbit lower than the operational one, which can then be reached with an orbital transfer maneuver. The second one demonstrates that efficient propulsion technologies are suited for optimized plane change maneuvers.

Several propulsion technologies are compared for different classes of small satellites. This is done both by a parametric analysis using analytical methods and by a numerical simulation. The study is done using Exotrail’s mission design software ExoOPS™ – *Mission Design*. The results will be oriented towards an operator point of view: time-to-orbit, average power consumption, energy consumption, lifespan remaining after the deployment, propellant consumption, number of thruster firings.

II. Methods and tools

Exotrail is developing ExoOPS^{TM} – *Mission Design* in order to help mission and systems engineers to design missions using low thrust propulsion systems. This software relies on an analytical module to rapidly perform parametric analysis that allow to identify the most interesting design choices for a given mission. A numerical module can then be used to refine and accurately simulate the mission and maneuvers identified thanks to the analytical module. It takes orbital perturbations and system constraints such as battery consumption and attitude into account. The software suite provides this capability via a secured cloud-based interface.

III. Missions description

A. Use-case introduction

In the present section we introduce the two mission scenarios that will be analyzed, together with our hypothesis and assumptions.

Two scenarios of constellation deployment are considered. In the first one, an orbit raising maneuver (change in orbit’s altitude and inclination) is followed by the repositioning of the two satellites on the orbit, later followed by the deorbiting. The second scenario consists of a relative Right Ascension of the Ascending Node (RAAN) phasing maneuver in order to spread the satellites on several operational planes. For each scenario, three different propulsion technologies are considered and three different classes of satellites. These will be compared using conventional merit indices such as duration of deployment, number and duration of the firing sequences and the consumed propellant during the maneuver. These directly impacts the economic aspects of the mission, via the lifetime after deployment and the availability of the spacecraft during station-keeping maneuvers.

B. Hypothesis

1. *Space mechanics hypothesis*

Analytical model

For the orbital transfer, all the results given using analytical formula assume that no other perturbation beside the thrust generated by the propulsion system acts on the spacecraft.

For the change in relative RAAN, the perturbation due to Earth's oblateness has been considered in the computation of an optimal coasting orbit.

Numerical model

When performing numerical simulations, the influence of the Earth's oblateness has been considered, including the effects of the J_2term in our analysis.

2. Satellite models

The term "small satellites" encompasses several satellite architectures. From the standard Cubesat, to more capable 50-kg satellites, many subsystems developed by the *New Space* actors have to be adaptable to numerous classes of satellites.

For this reason, three satellite models will be compared in this publication, ranging from a typical 12U Cubesat (a 20 x 20 x 30 cm satellite) to a more massive 50 kg satellite.

The main characteristics of the satellites are presented in Table 1, where:

- The drag area is the worst-case scenario with typical values representative of highly capable satellites of their respective class;
- The mean power is the average power dedicated to the propulsion system during maneuvers;
- The usable battery energy is the energy that can be drawn before the propulsion system must be disconnected. This value is typically the battery energy times the maximal depth of discharge allowed during the maneuvers.

Table 1: Main characteristics of the representative satellites

Satellite	Dry mass – without thruster mass (kg)	Drag area (m^2)	Mean power (W)	Usable battery energy (Wh)
12 U	12	0.18	30	30
27 U	27	0.4	50	50
50 kg-class	45	0.6	100	50

The satellites are represented as having deployable, fixed solar panels and a cubic shape.

Figure 1: Illustration of a small satellite with deployable solar panels (NASA)

During the simulations, the attitude of the spacecraft is computed and logged. In order to remove some complexity, it is assumed that the attitude can be changed arbitrarily. This assumes that the available torque to change the attitude is infinite and cannot saturate.

The batteries of the spacecraft are charged during sunlight. The energy gathered by the solar panels is maximized while thrusting optimally to perform the mission. During eclipse, the energy is drawn from the batteries to power the thruster. If more energy has been used by the thruster than what is available in the batteries, the maneuver is stopped and can only restart once the battery is full again.

3. Propulsion systems

Three propulsion systems will be compared in the following analysis. They are typical examples of electric propulsion technologies. Chemical thrusters are not considered here as they would be too bulky and heavy to fit the satellites considered here. These thrusters are characterized in Table 2.

Table 2. Characteristics of the three propulsion systems

Propulsion system	Thrust (mN)	Power (W)	Specific impulse (s)	Dry mass (kg)
Hall thruster [1]	5	100	1000	3
FEEP [2]	1	100	3500	2
Helicon thruster [3]	0.7	60	700	2

The pre-heating period for each technology before firing is not considered in the present study, even though ExoOPS^{TM} – *Mission Design* can account for this system-related feature.

It is worth noting that the power used by the thrusters exceeds the mean power delivered by the platform to the propulsion system. This translates into sparse thrust arcs instead of a continuous burn for the full duration of the maneuver.

In all the simulations, the mass of the satellite is equal to the sum of: the dry mass, without the propulsion system mass – Table 1 –, plus the propulsion system mass and the propellant mass – Table 2.

IV. Mission analysis and results

A. Orbital transfer analysis

1. Scenario / hypothesis

We consider two satellites with an operational 1200 km circular, Sun-Synchronous Orbit (SSO). Exploiting a rideshare solution, the launch orbit is a lower circular 500 km SSO orbit, making an orbit raising maneuver necessary. This maneuver will simultaneously change the semi-major axis and the inclination of the orbit, while taking care to keep the eccentricity close to zero. An orbit phasing maneuver that moves the satellites 180 degrees apart is performed, to better cover the orbital plane.

The first mission scenario is summarized as follows:

- Injection by a launcher on a 500 km SSO;
- Orbit raising from 500 km to a 1200 km SSO;
- Spreading of the satellites over the target orbit via a change in the anomaly;
- Station-keeping at this orbit to guarantee a five-year lifetime of the satellite from launch to decommissioning;
- Maneuvering of the satellite from 1200 km to a 600 km circular orbit where it will deorbit in less than 25 years as required by French law [4] and suggested by various agencies.

For this mission, the propellant used with each propulsion system is listed in Table 3.

Table 3: Propellant mass for the first mission

Propulsion system	12 U		27 U		50 kg	
	Propellant	Spacecraft	Propellant	Spacecraft	Propellant	Spacecraft
Hall thruster	3	18	3	33	5.5	53.5
FEEP	1.2	15.2	1.2	29.2	2	49
Helicon thruster	4	18	5	34	8	55

2. Maneuver computation.

Orbital Transfer

Exotrail's proprietary software includes several strategies to perform orbital transfer maneuvers. These take into account the fact that the satellite cannot thrust continuously over one orbit when the mean power required by the thruster exceeds the mean power available on the platform. On the one hand, the maneuver can be optimized to perform fuel and time efficient maneuvers. On the other hand, a strategy that reduces the number of thruster firings can be used to limit the stress system-wise. This is done at the cost of using larger maneuver arcs, resulting in a less efficient inclination change, since part of it will be performed further away from the nodes.

This analytical model relies on an integration of the instantaneous efficiency of the in-plane and out-of-plane thrust for the whole maneuver and only requires a function evaluation.

In the numerical module, the thrust vector for a combined semi-major axis and inclination change is computed in order to achieve the simultaneous convergence of the two elements to their target values as explained in [5]. The maneuver is performed around the orbital nodes and the duty cycle of the satellite is taken into account.

Orbital Phasing

For the orbital phasing, the optimization performs the integration of the mean motion difference during the three phases of the maneuver: positive or negative semi-major axis change, coasting, and getting back to the initial orbit.

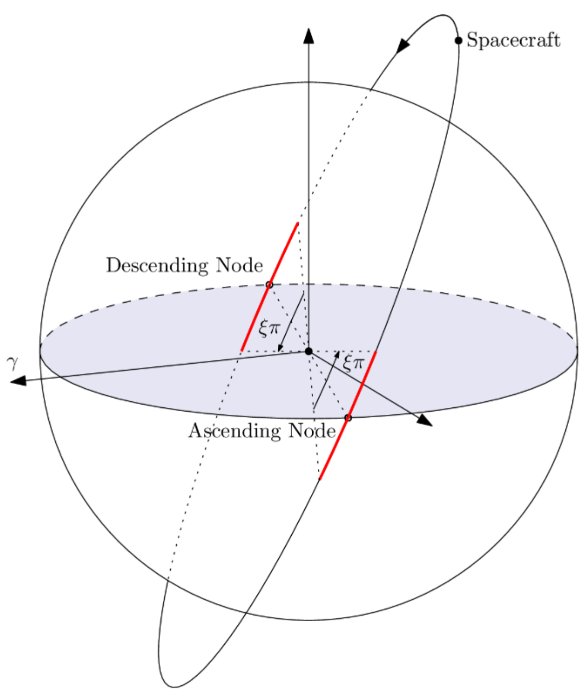


Figure 2: Finite thrust arcs represented on one orbit

The orbit phasing is achieved using the differential mean movement at different semi-major axis. An optimization is performed to look for the best drifting orbit while being constrained by two factors:

- Maximum ΔV usable for the maneuver;

- Maximum duration allowed for the maneuver.

Considering these two constraints, we can design the maneuver to perform in order to achieve the right anomaly phasing between the satellites.

A phasing of 180° is the most constrained case when performing constellation deployment and is used here as a worst-case scenario. The anomaly of the maneuvering satellite is changed relative to the anomaly of a satellite which would remain non-maneuvering.

Deorbiting

Deorbiting the spacecraft is achieved by lowering its altitude to 600 km, which guarantees a reentry in less than 25 years.

Number of burns estimation

The number of burns is computed with two thrust arcs per orbit centered on the nodes (Figure 2) when the available power for the thruster is lower than the actual instantaneous power required by the thruster. This strategy can be optimized if the energy stored inside the batteries is known and the charge of the battery occurs during several orbits. However, the ratio of the number of burns for each case will not be changed. In the case where the power available for the propulsion equals the power used by the thruster, only one burn is considered for each maneuver.

3. Results

Numerical results

The results of the simulations of two different cases are shown below:

- A 12 U satellite equipped with a Hall thruster (HT);
- A 12 U satellite equipped with a Field Emission Electric Propulsion (FEEP) thruster.

In average this mission uses 800 to 1020 m/s of ΔV depending on the thruster and the platform.

Table 4 gives the results for each phase of the mission. **OT** stands for orbit transfer, **OP** for orbital phasing, **DE** for deorbiting and **TOT** for the total of the mission. The fuel ratio α_f, which represents the ratio between the mass of fuel consumed for a maneuver and the overall mass:

$$\alpha_f = \frac{fuel\ mass\ consumed}{wet\ mass_0} \tag{1}$$

Table 4. Duration, number of burns and fuel ratio for the detailed simulations of the first case

Case	Duration [d]				Number of burns				Fuel ratio [%]			
	OT	**OP**	**DE**	**TOT**	**OT**	**OP**	**DE**	**TOT**	**OT**	**OP**	**DE**	**TOT**
12U Hall	73	7	39	119	2076	180	1101	3357	5.3	0.5	2.9	8.7
12U FEEP	316	15	175	506	8928	382	4909	14219	1.6	0.07	0.87	2.5

The following figures show the variation of the controlled orbital elements as a function of time for the orbital transfer, the orbital phasing and deorbiting.

It is to be noted that the x-axis does not have the same scale for the Hall thruster and the FEEP thruster.

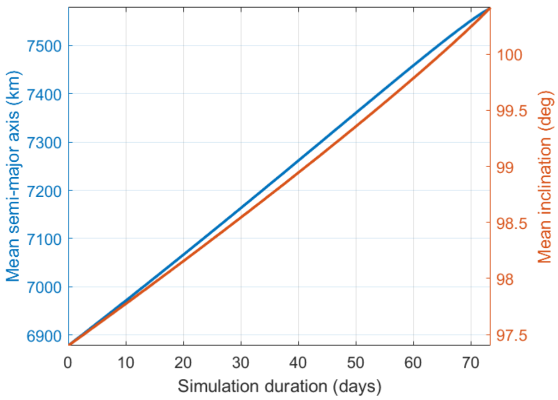


Figure 3: Orbital transfer Hall thruster

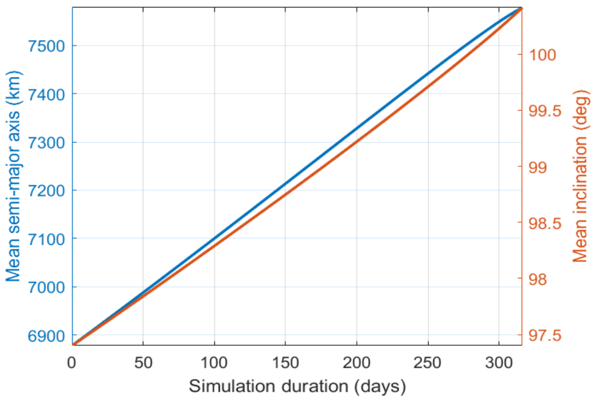


Figure 4: Orbital transfer FEEP

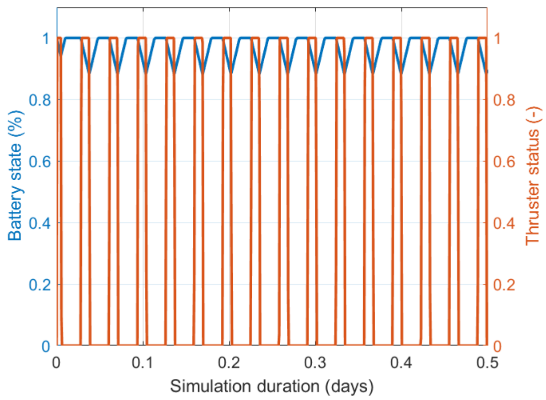


Figure 5: Battery level and thruster status over a fraction of the orbital transfer using a Hall thruster

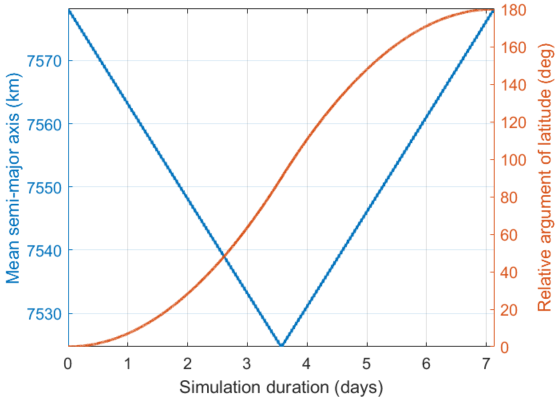


Figure 6: Orbital phasing Hall thruster

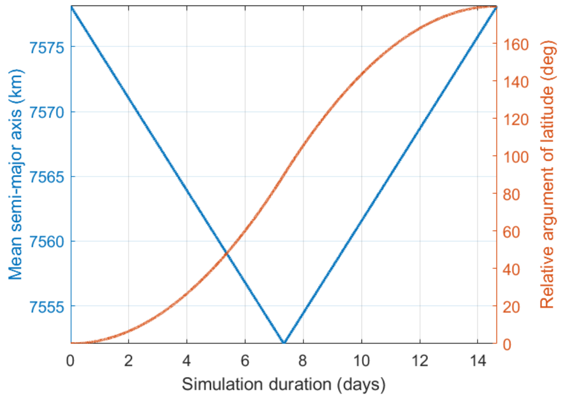


Figure 7: Orbital phasing FEEP

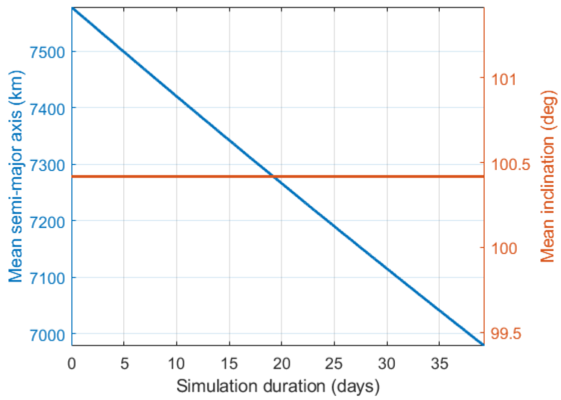


Figure 8: Deorbiting Hall thruster

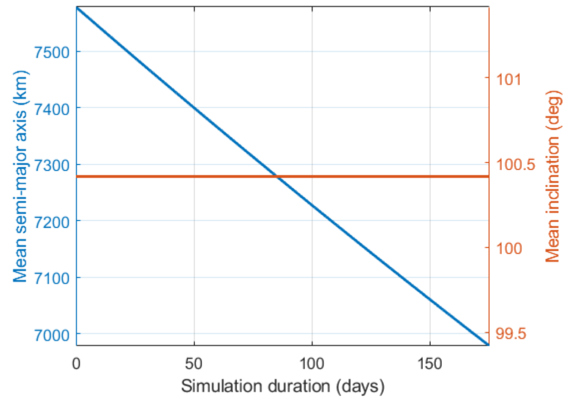


Figure 9: Deorbiting FEEP

Analytical results

All the other cases have been computed using the analytical model developed by Exotrail which is described in *Section 1*.

Table 5. Maneuver duration, number of burns and fuel ratio for the first mission

Platform	Thruster	Duration [d]	Number of burns	Fuel used [kg]	Fuel ratio [%]
12 U	Hall	123	3 240	1.5	8
	FEEP	506	14 110	0.38	2
	Helicon thruster	499	13 922	2.2	12
27 U	Hall	137	3 642	2.8	8.7
	FEEP	615	17 203	0.77	2.6
	Helicon thruster	587	16 893	4.4	13
50 kg-class	Hall	127	4	5.2	9.8
	FEEP	570	4	1.4	2.9
	Helicon thruster	858	4	7.5	13.7

The analytical model gives accurate results for any orbital transfer considered in this study. An internal study performed at Exotrail, has shown that this is true as long as the thrust-to-weight ratio remains in the realm of electrical propulsion and the maneuver requires more than a dozen orbit to be performed. The level of accuracy of the analytical model can be appreciated in Table 6 where we compare the thrust duration using the analytical model with the one found using numerical integration.

Table 6. Comparison between analytical results and detailed simulation for the first mission

Case	Simulation [d]	Analytical model [d]	Difference [%]
12U – Hall thruster	119	123	3
12U – FEEP	506	506	0

Discussion of the results

From the results that are presented here, we can see that the lighter mass of the Helicon thruster does not compensate for its lower specific impulse compared to the Hall thruster. In addition, its lower thrust leads to a longer mission duration and does not seem to outperform any of the two other thrusters.

Therefore, for this mission, the trade-off is to be made between the Hall thruster, a high thrust-to-power ratio but medium specific impulse thruster, and a low thrust-to-power ratio but high specific impulse thruster, the FEEP thruster. In this case, we can see that a modest 6% increase of the fuel ratio, from 2 to 8 %, leads to a reduction of the mission duration by a factor of 5. If the mass is the most stringent requirement, a FEEP thruster is the good solution, but if the mission duration is important, the fuel mass penalty linked to using a less fuel-efficient thruster is easily offset by the shorter maneuvers.

This quick study shows two major results: an analytical model, as developed by Exotrail, is precise enough to finely approximate the fuel consumption and firing duration, even for combined, low-thrust maneuvers; and the trade-off between specific impulse over thrust-to-power ratio must be performed using additional metrics related to the cost of mass launched and cost of operating a spacecraft in orbit while not performing its nominal mission.

B. RAAN phasing mission analysis

1. Scenario and RAAN phasing mission principle

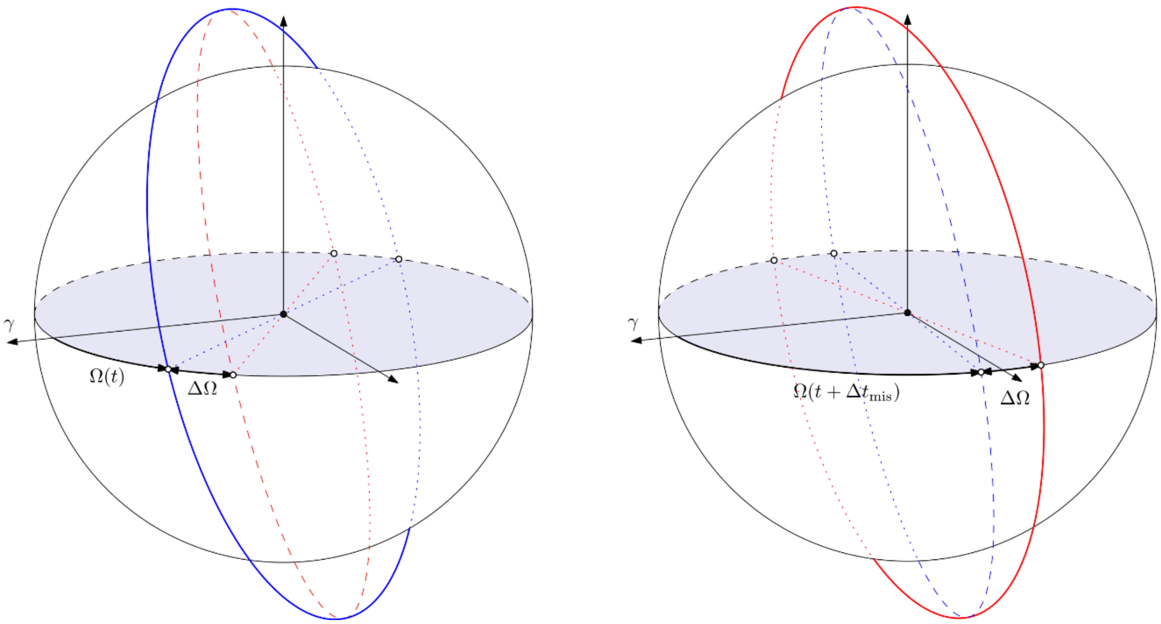
In this scenario, we want to deploy a constellation of satellites launched using a rideshare launch. A satellite is injected on a Sun-synchronous circular orbit at an altitude of 500 km. This injection is followed by a relative RAAN phasing to change its orbital plane. The satellite stays on this orbit throughout the rest of the mission. It is then deorbited by the natural orbital decay due to the atmospheric drag.

RAAN phasing missions are performed to shift the orbital node of the orbit of a satellite with respect to that of its initial orbit. The strategy used here exploits the natural drift in RAAN produced by the oblateness of the Earth – J_2 harmonic of the Earth potential. The RAAN drift rate $\dot{\Omega}$ due to the J_2 effect changes with the semi-major axis a and the inclination i of the orbit according as follows [6]:

$$\dot{\Omega} = -\frac{3}{2}\sqrt{\frac{\mu_\oplus}{a^3}} J_2 \left(\frac{R_\oplus}{a}\right)^2 \frac{\cos(i)}{(1-e^2)^2} \tag{2}$$

where $\mu_\oplus$ is the standard gravitational parameter of the Earth and $R_\oplus$ the equatorial radius of the Earth.

We proceed by maneuvering from the initial orbit to a coasting orbit, with a different semi-major axis and a different inclination. After an optimal coasting duration, we maneuver again to bring the satellite from the drifting orbit to its final orbit. The change in RAAN drift rate during these three phases make the RAAN of the orbit of the satellite slowly drift with respect to the RAAN of the initial orbit. Note that for Sun-synchronous orbits, this mission performs a change in the local time at the ascending node.



(a) Beginning of the maneuver **(b) End of the maneuver**

Figure 10: Plane change maneuver, including natural RAAN drift. The initial orbit is the blue one, the target orbit is the red one.

The parameters of the coasting orbit are found using an optimization algorithm in order to perform one of the following:

- Optimizing the ΔV while having as an input the target duration and target $\Delta RAAN$;
- Optimizing the maneuver duration with a fixed ΔV budget and a target $\Delta RAAN$;
- Optimizing the $\Delta RAAN$ with a fixed ΔV and a given duration.

This mission is time and fuel consuming. However, it is very efficient at quickly deploying a constellation launched via cheap rideshare. Indeed, it can spread a batch of satellites launched on the same initial orbital plane on distinct orbital planes. In many cases, this strategy is more cost-effective than relying on expensive dedicated launches to populate each plane. To perform such a maneuver, a good trade-off is to have a high thrust-to-power ratio – see [7].

1. Use-case data

In our example, we want to deploy a constellation consisting of twenty satellites spread out on four different planes. The satellites are launched on a 500 km circular SSO and operate on this orbit. There are three possible launch scenarios. All satellites can be sent to space in a single launch and spread out by plane changing missions, they can be sent in two launches also followed by plane change missions, or they can be launched by four dedicated launches, one for each plane. In this example, the difference in RAAN between two consecutive planes is equal to 30°.

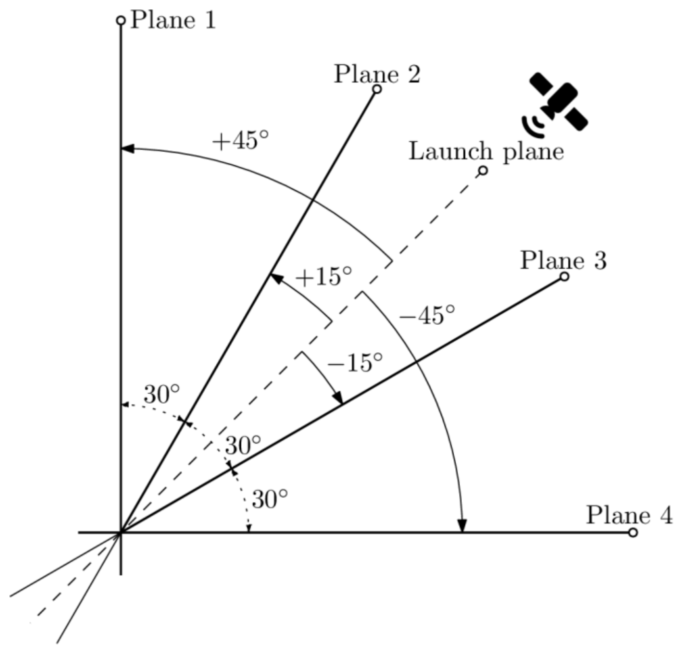


Figure 11: Deployment strategy from one plane to four planes

To assess the economics of the different launch options, we consider the cost of launching and deploying the constellation in all three cases. In order to be agnostic of the thruster's price, we will only consider the price-per-kilogram of mass launched and the cost of time, which is an aggregate of the cost of having the satellite in orbit while not performing its main revenue-generating mission. This number comes from three major sources of cost: the cost of operations to perform the maneuver, the cost of reducing the operational lifetime of the satellite once in orbit and the revenue loss during this period. The different hypotheses are the following:

- 1 launch: in this case, the satellites are launched using a single rideshare with an aggressive pricing at 5,000 $/kg such as announced by SpaceX [8];
- 2 launches: the price-per-kilogram is at 15,000 $/kg with average rideshare pricing;
- 4 launches: using dedicated launchers to send the spacecraft at the same orbit and the right RAAN for each is more costly and is set at 50,000 $/kg [9];
- The cost of time is of 3,000 $/day and represent the revenue loss and the operations cost while performing maneuvers. This cost is equal for all the different configurations. It can be computed as follow: if a satellite costs 5 M$ and has a lifetime of 5 years, each day of having the spacecraft in orbit costs 2,700 $. Then you add a 300 $/day for operations, which leads to the 3,000 $/day;
- The time to arrange the different flights is the duration between the launch of the first satellite and the launch of the last satellite. It ranges from 0 days for 1 launch to 3 months to arrange 4 different launches.

All the data are summarized in Table 7.

Table 7. Economics inputs for constellation deployment case study

Number of launches	$\Delta RAAN$ to perform [deg]	Cost of launched mass [$/kg]	Cost of time [$/d]	Time to arrange launches [d]
1	± 45	5,000	3,000	0
2	± 15	15,000	3,000	30
4	0	50,000	3,000	90

For the cases studied here, we will optimize on the duration of the maneuver with the following constraints:

- Maximum ΔV available is 400 m/s;
- $\Delta RAAN$ is given by the case: ± 15° and ± 45°.

For this mission, the propellant mass will be adjusted to deliver the expected 400 m/s of ΔV and the masses used are given in Table 8.

Table 8: Propellant mass for the plane change mission

Propulsion system	12 U		27 U		50 kg	
	Propellant	Spacecraft	Propellant	Spacecraft	Propellant	Spacecraft
Hall thruster	0.72	15.7	1.2	31.2	2.0	50.0
FEEP thruster	0.21	14.2	0.35	29.4	0.58	47.6
Helicon thruster	1.0	15.0	1.7	30.7	2.8	49.8

2. *Results*

As for the first mission scenario, we have performed numerical simulations on two cases of interest:

- A 12 U satellite equipped with a Hall thruster;
- A 12 U satellite equipped with a FEEP thruster.

The semi-major axis and inclination of the spacecrafts during a maneuver to achieve a $\Delta RAAN$ of -15° are shown in Figure 12 and Figure 13. Figure 14 and Figure 15 show the build-up of the relative RAAN variation during the maneuver until the final value is reached.

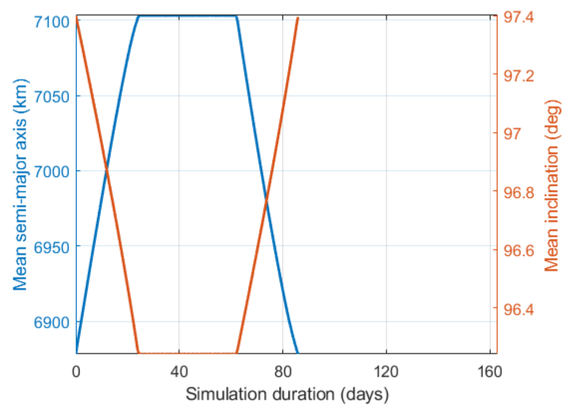


Figure 12: Plane change maneuver ΔRAAN = -15° Hall thruster

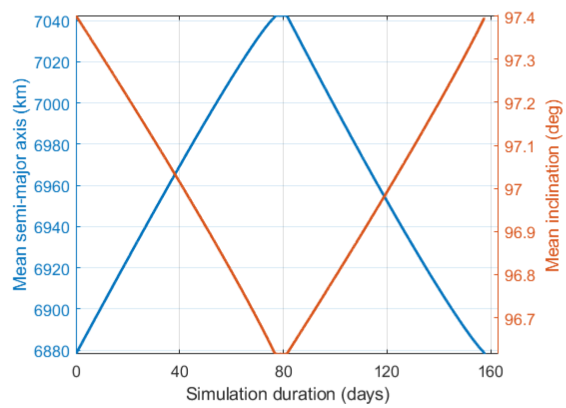


Figure 13: Plane change maneuver ΔRAAN = -15° FEEP

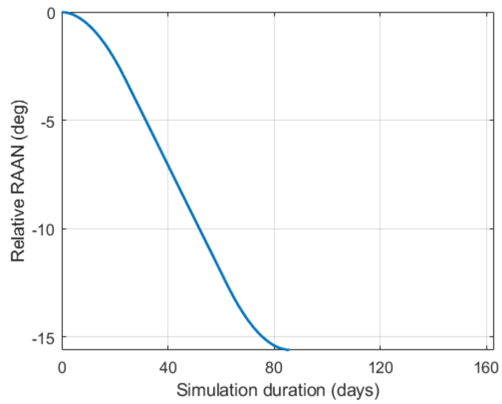


Figure 14: Plane change maneuver ΔRAAN = -15° Hall thruster

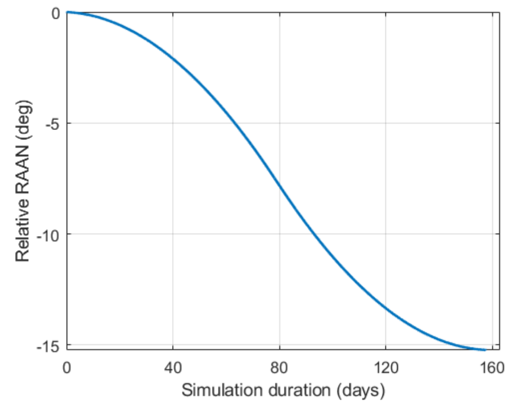


Figure 15: Plane change maneuver ΔRAAN = -15° FEEP

Analytical study of all cases

A plane phasing maneuver is composed of two orbital transfer and a coasting orbit. Using our analytical model, we can compute the thrust duration for a fixed ΔV and $\Delta RAAN$. The mission duration as a function of $\Delta RAAN$ has been computed and is presented on Figure 16. This figure is directly taken from ExoOPS™ – *Mission Design*.

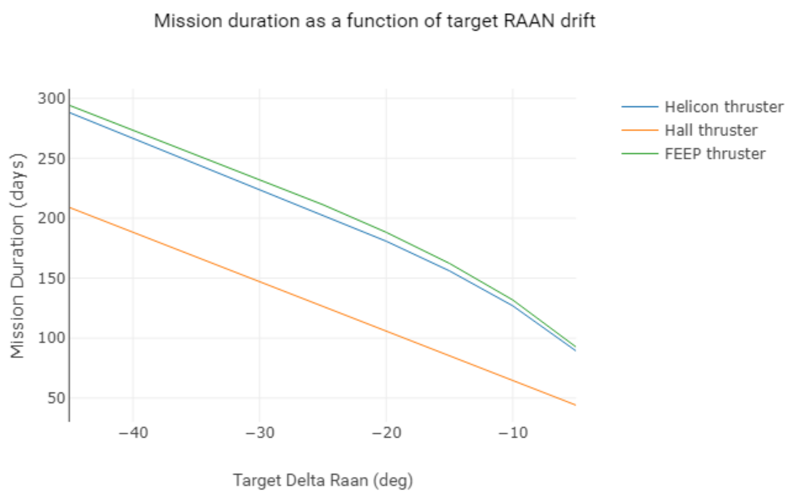


Figure 16: Mission duration as a function of targeted ΔRAAN for the three thrusters on a 12U platform

The results for a -15° RAAN change are presented in Table 9.

Table 9: Results for the plane change maneuver

Platform	Thruster	Duration [d]	Number of burns	Fuel ratio [%]
12 U	Hall thruster	85	1,410	4.0
	FEEP	162	4,712	0.8
	Helicon thruster	156	4,556	4.5
27 U	Hall thruster	92	1,680	4.0
	FEEP	184	5,354	0.8
	Helicon thruster	182	5,322	4.3
50 kg	Hall thruster	99	2	4.0
	FEEP	181	2	0.9
	Helicon thruster	219	2	3.8

From the economics point of view, Table 10 gives the results. In this table, for each platform the most cost-effective solution is highlighted in green. The costs R are given relative to the lower launch cost for each platform.

$$C = c_{launch} \times \left(m_{sat} + m_{thruster} + m_{prop}\right) + c_{time} \times T_{man} + c_{to\ launch} \tag{3}$$

where c_{launch} is the cost of launch per kilogram, m_{sat}, $m_{thruster}$ and m_{prop} are the mass of the satellite, thruster and propellant respectively, c_{time} is the cost of time, T_{man} is the mission duration and $c_{to\ launch}$ is the cost of the time between the different launches. And:

$$R = C - C_{min,\ \ per\ platform} \tag{4}$$

Table 10: Economic data about constellation deployment cases

Platform	Thruster	Number of launches	R = Relative total cost (launch + transfer duration) [k$]
12 U	Hall thruster	1	303
		2	-
		4	563
	FEEP	1	551
		2	208
		4	488
	Helicon thruster	1	537
		2	201
		4	528
27 U	Hall thruster	1	162
		2	-
		4	1,085
	FEEP	1	473
		2	247
		4	995
	Helicon thruster	1	490
		2	261
		4	1060
50 kg	Hall thruster	1	-
		2	133
		4	1757
	FEEP	1	249
		2	345
		4	1647
	Helicon thruster	1	411
		2	491
		4	1757

The following equation gives the total impulse I_{tot} as a function of the specific impulse I_{sp} and propellant mass m_{prop}.

$$I_{tot} = g_0 \times I_{sp} \times m_{prop} \tag{5}$$

With Equation 4 and Equation 5, we can compute an analytical formula ta compare different thrusters for a given scenario:

$$\text{If } \Delta T_{man} > \frac{c_{launch}}{c_{time}} \times \left(\Delta m_{thruster} + \frac{I_{tot}}{g_0} \frac{\Delta I_{sp}}{I_{sp,1} \times I_{sp,2}}\right), \text{ then } \Delta C > 0 \tag{6}$$

where, ΔT_{man} is the difference between the two mission durations, $\Delta m_{thruster}$ is the difference between the mass of two thrusters, ΔI_{sp} is the difference between the two thrusters, $I_{sp,1}$ and $I_{sp,2}$ are the specific impulse of the two thrusters and ΔC is the difference between the cost of the two missions.

In our case, for the 50 kg satellite, between the Hall thruster and the FEEP thruster, we have:

- $\Delta T_{man} = T_{man,\ FEEP} - T_{man,\ Hall} = 82\ days$
- $\frac{c_{launch}}{c_{time}} \times \left(\Delta M_{thruster} + \frac{I_{tot}}{g_0} \frac{\Delta I_{sp}}{I_{sp,1} \times I_{sp,2}}\right) = 4.1$

Therefore, we can directly conclude that for this mission the Hall thruster is more economical.

As for the first mission, we will compare the difference between the analytical modelling and the numerical simulations. Table 11 summarizes the results and give the difference between them in terms of maneuver duration.

Table 11. Comparison between the analytical model and the detailed simulations for the second mission

Case	Mission duration (analytical) [d]	Mission duration (numerical) [d]	Difference [%]
12 U – Hall thruster	88	86	2
12 U – FEEP	162	158	2

C. Discussion of the results

This simulation is quite thorough and gives a good overview of the different challenges of launching a constellation. In all the cases, the cost of time is the same, therefore the relative weight of the time to reach the orbit to the mass launched changes along with the mass of the satellite. With the lighter spacecraft, the cost of time and the launch cost are balanced, and the best solution is to use only 2 launches along with a minor plane change maneuver. When the spacecraft mass increases, for the 27U or the 50kg-class satellites, the cost of sending the spacecraft in orbit is the driving factor and therefore the better solution is always to use one cheap rideshare launch and then to perform a more important plane change maneuver of 45° in RAAN.

The economic benefit of using a rideshare plus propulsion solution versus dedicated launches can be as high as 500 k$ per satellite for a 12U, 1 M$ for a 27U and 1.7 M$ for a 50kg spacecraft.

V.Conclusion

In this publication we detailed the analysis of two case studies. These two are typical mission scenarios for capable small satellites and include orbit raising, in-orbit repositioning and deorbiting, and a plane change maneuver used to deploy constellations. These missions can all be studied using our ExoOPS™ – *Mission Design* software.

For these two missions, three different platforms equipped with three different thrusters each have been considered to compute the mission duration, number of burns and fuel ratio. Two models have been used: an analytical one to rapidly compute the results for all cases, through a parametric analysis and a numerical one, considering perturbations and system-related events to perform an accurate simulation of the more interesting alternatives. A very good agreement between the two models has been found by comparing two cases, for both missions.

For the first mission, we studied the impact of using either a high thrust-to-power ratio propulsion system versus a high specific impulse thruster. We have shown that a much shorter mission duration can be obtained with a high thrust-to-power ratio at the price of a slight increase in mass penalty. Indeed, a 6% increase in the total mass allows for a five-fold reduction in mission duration.

In the second analysis, we analyzed the economic impact of using rideshare launches as an alternative to dedicated ones for constellation deployment. In this case, the parametric study performed shows good agreement with the conclusions of the previous analysis: the smaller mass penalty of using a thruster with a higher specific impulse is totally offset by the gain in mission duration with a higher thrust-to-power ratio.

This second study also shows that the trade-off between rideshare and dedicated launches must be performed early in the development, before choosing the propulsion solution, as it can bring a reduction of up to 1.7 M$ per satellite to the cost of deploying a constellation when using Hall thrusters.

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