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Investigation of different strategies for access to space of small satellites on a defined LEO orbit

Francesco Barato [a,*], Elena Toson [b], Fabiana Milza [b], Daniele Pavarin [b]

[a] Department of Industrial Engineering, Università degli Studi di Padova, Via Venezia 1, 35131, Padua, Italy
[b] Technology for Propulsion and Innovation S.p.A., Via Emilia 15, 35043, Monselice, Padua, Italy

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ABSTRACT

In recent years small satellites have shifted from being secondary items to dominate the space market thanks to, but not only, the development of the large LEO constellations. For small satellites to be highly effective, particularly when arranged in such constellations, each satellite has to be placed in a specific orbital plane and orbital position.

Currently, three main options are available to reach a specific orbit. The first, more straightforward solution is to employ a dedicated launch with a small launch vehicle. Plenty of small launchers are in development around the world (with few already operational); however, their cost per kilogram is predicted to be much higher than for large launchers. The second possibility is the exploitation of a rideshare option, where the satellite is transported by a large launcher together with other payloads on a general predetermined orbit. Afterwards the satellite needs to be transferred to its designed orbit. This can be done in two ways: through the use of a satellite carrier (also called self-propelled dispenser) or with the satellite own propulsion system. In both last cases, the mass transported by the launcher into the release orbit is higher than the final one necessary for the nominal mission, impacting total costs.

In this paper these three possibilities are compared considering the need to reach various specific orbits, starting from a different release one in the case of the rideshare options. First of all, the change in velocity for different orbital parameters (altitude, eccentricity, phase, argument of perigee, inclination, RAAN) is computed. Afterwards the propulsion mass budget is calculated.

Everything else being equal, it is demonstrated mathematically that a dispenser is inherently less efficient than a group of autonomous satellites, particularly for the RAAN change and a large number of carried satellites. However, it is not always possible or convenient to provide small satellites, particularly the smallest ones (nanosats/cubesats/microsats) with comparable propulsion capabilities of a larger dispenser, making the latter still an attractive option in several situations. A cost analysis also shows that, particularly for sophisticated small satellites, when the final orbit is far from the release one, a dedicated small launch vehicle can be cost competitive with the nominally much cheaper large launcher.

1. Introduction

In recent years, there has been a tremendous growth in the usage of small satellites operating in LEO and the trend is for a further increase in future years [1–5]. Small satellites can reach their orbit with a single dedicated launch to their final destination on a small launch vehicle or take a rideshare on a larger one and move on the correct position with their own propulsion system or be transported by a dedicated orbital platform often called space tug or self-propelled dispenser or satellite carrier. All options are currently subject to significant development efforts [6–10]. Several small launchers are already operational and many more are in development [11–24]. The same applies for satellite carriers, like D-Orbit ION [25], Firefly Aerospace SUV [26], Momentus Vigoride [27], Spaceflight SHERPA [28–30], MOOG OMVs [31–33] and many others [34,35]. Moreover, propulsion systems for small satellites are continuously growing in variety and performance [36]. Differential drag is also used for relative orbital maneuvering but with constraints on the parameters affected [37–40]. Lift has been also considered but classical satellites tend to have low aerodynamic efficiencies. Both

* Corresponding author.
E-mail addresses: francesco.barato@unipd.it (F. Barato), e.toson@t4innovation.com (E. Toson), f.milza@t4innovation.com (F. Milza), d.pavarin@t4innovation.com (D. Pavarin).

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Nomenclature

a	Major semiaxis
c	Costs
e	Eccentricity
g_0	Reference gravitational acceleration
h	Altitude
i	Inclination
I	Impulse
Isp	Specific impulse
J	Zonal coefficient of the gravitational potential
k	Structural index, constant
m	Mass (generic)
M	Satellite mass
n	Average orbital angular speed
N	Number of satellites, orbits
R, r	Radius
t	Time
T	Orbital period
v	Velocity
ε	Inert mass fraction
γ	Flight path angle
ϑ	Angle of rotation of the orbit plane
θ	True anomaly
ω	Argument of perigee
Ω	RAAN
μ	Earth gravitational constant

Subscripts/Superscripts

180°	180° angular variation
ap	Apogee
f	Final
fix	Fixed, constant
fs	Free (autonomous) satellites
i	Initial
inert	Inert
lv	Launch vehicle
m	Mission
max	Maximum
p	Perigee
pay	Payload
pr	Propulsion
prop	Propellant
ref	Reference
sc	Sat carrier
t	Terrestrial, fixed time
tot	Total
Δv	Fixed ΔV

Acronyms/Abbreviations

AVUM	Attitude and Vernier Upper Module
GEO	GEostationary Orbit
GTO	Geostationary Transfer Orbit
HAPS	Hydrazine Aux. Propulsion System
ION	In Orbit Now
LEO	Low Earth Orbit
OMV	Orbital Maneuvering Vehicle
RAAN	Right Ascension Ascending Node
SHERPA	SHuttle Expendable Rocket for Payload Augmentation
SSCM	Small Satellite Cost Model
SSO	Sun Synchronous Orbit
SUV	Space Utility Vehicle
VLEO	Very Low Earth Orbit

aerodynamic maneuvers work well only at altitudes where the satellite lifetime is affected. Finally, ejection systems like springs can be used together with proper orientation and timing of the releasing vehicle to deploy multiple satellites [41]. However, the resulting orbits are very similar between each other.

In the past Russia had the lead of rideshare missions [42] but now is almost out of the market, the Indians held the previous record of number of satellites launched in a single mission [43] until SpaceX started to dominate the market with its Transporter missions in SSO and the recently announced future Bandwagon missions at 45° of inclination [44].

Also in the past, when the GEO market was dominant, the more complex possibility to reach LEO from GTO exploiting drag was investigated several times [45,46], but today the opportunities to go directly to LEO are more frequent and thus this option is not considered in this paper.

Several papers have already investigated the optimal strategies for constellation deployment, but the focus is mainly on classical rideshare missions with relatively little propulsive capabilities [47–55].

The aim of this paper is to identify advantages, drawbacks, limits and constraints of the aforementioned three different propelled options and determine the best solutions and their preferred ranges of applicability.

2. Orbital manoeuvres

The position of a satellite around the Earth is determined by six parameters: the Right Ascension of the Ascending Node (RAAN) Ω, the inclination of the orbit i, the argument of perigee ω, the major semi-axis a, the eccentricity e, and the true anomaly θ. Each one requires a specific manoeuvre in order to modify its value [56,57]. The manoeuvre is generally performed with a propulsion system that imparts the required ΔV (both in magnitude and direction). In the following, the manoeuvres necessary to change each parameter are briefly presented [58–62].

2.1. Altitude change

The altitude of a satellite can be changed trough an in-plane manoeuvre, imparting a ΔV parallel to the velocity or orthogonal to the orbit radius vector. For a quasi-circular orbit or at the apsides the two strategies are the same. In the limiting case of an impulsive burn, a Hohmann transfer is performed:

$$\Delta v = v_p - v_1 + v_2 - v_{ap} \quad (1)$$

For continuous thrusting the satellite spirals from the initial to final orbit:

$$\Delta v = v_1 - v_2 \quad (2)$$

In LEO the difference between the two methods is negligible. Moreover, in LEO the variations are nearly linear, symmetric and the Hohmann transfer is the sum of two roughly equal burns. The ΔV can be linearized in the following way:

$$\Delta v / v = 0.5\Delta a / a$$

Or more empirically:

$$\Delta v(m/s) = 0.55\left(\frac{R_t + 500}{R_t + h(km)}\right)\Delta h(km) \quad (4)$$

Or the even simpler but less accurate:

$$\Delta v(m/s) = 0.55\Delta h(km)$$

Which shows that 55 m/s are roughly needed every 100 km of altitude change (see Fig. 1).

2.2. Eccentricity change

The eccentricity of an orbit in LEO is always below 10 % up to 1500 km and below 5 % up to 800 km. Thus, again the problem becomes nearly linear. Changing the eccentricity means increasing/decreasing the altitude of the perigee/apogee with two opposite half-Hohmann transfers. Applying the linearization:

$$\frac{\Delta v}{v} = 0.5\Delta e(6)$$

This means that a variation from a circular orbit to $e = 0.01$ (e.g. 300–436 km) requires around 40 m/s. In the case of continuous thrusting, there is at least a 30 % penalty for the optimum steering program:

$$\Delta v / v = 0.649\Delta e$$

And slightly more (<3 %) for inertially fixed steering:

$$\Delta v / v = 0.667\Delta e$$

2.3. Phase change

To change the true anomaly of a satellite a phase change manoeuvre is necessary. The satellite is placed in a lower or higher energy orbit that has respectively a lower or higher orbital period:

$$T = 2\pi / n \quad n = \sqrt{\mu / a^3}$$

After a certain amount of time, the satellite comes back in the original orbit on a different angular position. This is a typical manoeuvre used to separate several satellites launched with the same vehicle and spread them uniformly along the whole orbital plane. Again, in LEO the manoeuvre is linear and symmetric, so it can be performed equally increasing the apogee, decreasing the perigee or transferring to a higher/lower circular orbit, the only difference is the direction of the burn and the relative motion. The last option requires four burns instead of two. The ΔV for an impulsive burn is:

$$\Delta v / v = 0.106\Delta\theta / N = 0.106\frac{d\theta}{dt}T$$

Where N is the number of orbits. If the trip time is above 7 days (87–114 orbits) the ΔV is small (<28 m/s for a 180° shift) and the transfer time is below 1.2 % for a full Hohmann transfer. In case of a two burns manoeuvre, the difference between considering a continuous function or the real discrete one (multiple of N) is again below 1.2 % above 7 days. For continuous thrusting the effort is doubled:

$$\Delta v / v = 0.212\Delta\theta / N = 0.212\frac{d\theta}{dt}T \quad (11)$$

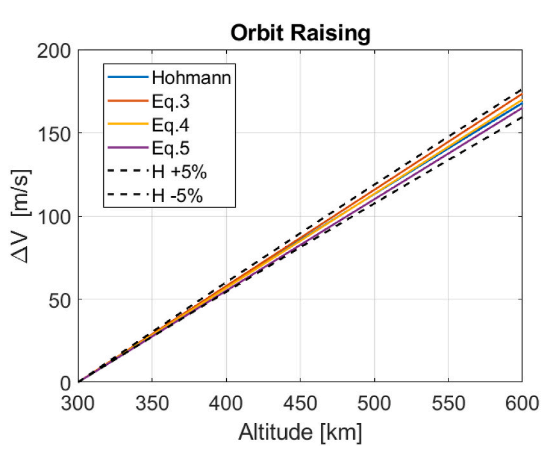

Fig. 1. ΔV as a function of the final altitude, starting from 300 km.

2.4. Argument of perigee change

Small changes of the argument of perigee can be obtained with an in-plane radial thrust at the apsides. The general formula for a radial impulse at the apsides is:

$$\Delta v = \mu\, e \tan(\Delta\omega) / (r_p v_p) \quad (12)$$

In the case of continuous thrusting the direction of thrust is parallel to the major axis of the ellipse:

$$\Delta v = \frac{2}{3}\sqrt{\frac{\mu}{a(1-e^2)}}e\Delta\omega$$

In case of large angle variations of ω, the impulsive thrust shall be performed at the intersection point between the old and new orbit, which in general is not at the apsides:

$$\Delta v = 2v \sin\gamma = 2\sqrt{\frac{\mu}{a(1-e^2)}}e \sin(\Delta\omega / 2)$$

For small angles Eq. 14 coincides with Eq. (12) as the intersection point move toward the apsides. It is also worth noting that this is the only case where continuous thrusting gives a saving, this time by 1/3.

Even if the ΔV is proportional to the eccentricity, a value of only 0.01 (e.g. 300–436 km) of e can lead to a substantial value of 155 m/s for a 180° rotation.

The argument of perigee can be also changed by a phasing manoeuvre, exploiting the differential drift caused by the J2 perturbation:

$$\frac{d\omega}{dt} = \frac{3}{4}nJ_2\left(\frac{R_t}{a(1-e^2)}\right)^2(4 - 5\sin^2 i)$$

The drift is null at the critical inclination of 63.4°.

2.5. Inclination change

A change of inclination requires an out-of-plane manoeuvre. With impulsive trusting at the orbital nodes:

$$\Delta v / v = 2\sin(\Delta i / 2)$$

That for small angles simplifies to:

$$\Delta v / v = \Delta i$$

In case of continuous thrusting the thrust is perpendicular to the orbital plane and the expression is:

$$\Delta v / v = 2\sin(\pi\Delta i / 4)$$

That for small angles simplifies to:

$$\Delta v / v = \frac{\pi}{2}\Delta i$$

Which is 57 % higher than the impulsive case.

For large angles, the inclination change requires a ΔV on the same order of magnitude of orbital velocity, with only 1° variation requiring already around 135 m/s. Therefore, it is generally applicable only for very small corrections, for example adjusting the inclination of a SSO after an orbit raising. In fact, the Sun synchronous inclination changes from 96.5° at 250 km to 100.5° at 1200 km. The inclination is thus defined for its major part by the launch vehicle.

2.6. RAAN change

The RAAN can be changed directly with a corresponding manoeuvre or indirectly exploiting the drift caused by the J_2 perturbation.

2.6.1. Direct RAAN change

The RAAN can be changed directly by an impulsive manoeuvre at the intersection between the new and old orbit. This out-of-plane manoeuvre is analogous to an inclination change (equal for $i = 90^\circ$) and requires a similar massive ΔV except at low inclinations:

$$\cos\vartheta = \cos^2 i + \sin^2 i \cos\Delta\Omega \tag{20}$$

$$\Delta v = 2v \sin\frac{\vartheta}{2}$$

In case of continuous thrusting the thrust is perpendicular to the orbital plane and the direction reverses at the nodes:

$$\Delta v \Big/ v = \frac{\pi}{2}\sin(i)\Delta\Omega \tag{22}$$

As for the inclination, direct RAAN modifications are performed only to make fine adjustments, for example to compensate the slight drift incurred during a phasing manoeuvre.

2.6.2. Indirect RAAN change

A change of RAAN can be obtained with a much lower ΔV exploiting the natural drift due to the J_2 perturbation (Fig. 2):

$$\frac{d\Omega}{dt} = -\frac{3}{2} n J_2 \left(\frac{R_t}{a(1-e^2)}\right)^2 \cos i \tag{23}$$

The satellite performs a manoeuvre analogous to a phasing one in order to increase or decrease its value of a. In this way the differential drift produces a finite change of $\Delta\Omega$. Finally, the satellite is brought back to its nominal orbit.

As the exponent of a is $-7/2 = -3.5$ linearity is kept in a much more restricted range compared to the previous cases. Reducing the altitude has thus a stronger effect than increasing it (see Fig. 3). The difference between a two burns manoeuvre and a full Hohmann transfer remains negligible, but the full Hohmann transfer allows halving the maximum change of altitude (see Fig. 4).

This is important in LEO as the altitude cannot be reduced too much in order to avoid excessive drag during the RAAN change (see Fig. 5). Not only the alternative orbit raising is less efficient but also in this case the altitude variation could be limited by communication or radiation issues. It also important to remember that the orbit raising will shift Ω in the opposite direction than the orbit drop.

For a similar ΔV (i.e. altitude variation) the timescale of the RAAN change manoeuvre is much longer than for the phase change. Therefore, a much higher propellant mass is required for the same trip time (see Fig. 6).

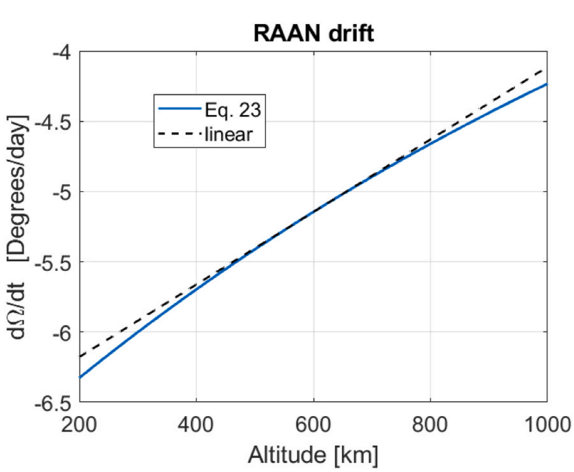

Fig. 2. Absolute RAAN drift as a function of altitude. $i = 45^\circ$.

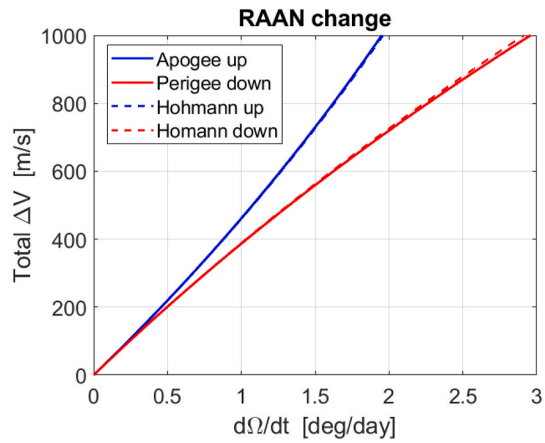

Fig. 3. ΔV as a function of RAAN change rate. $i = 45^\circ$.

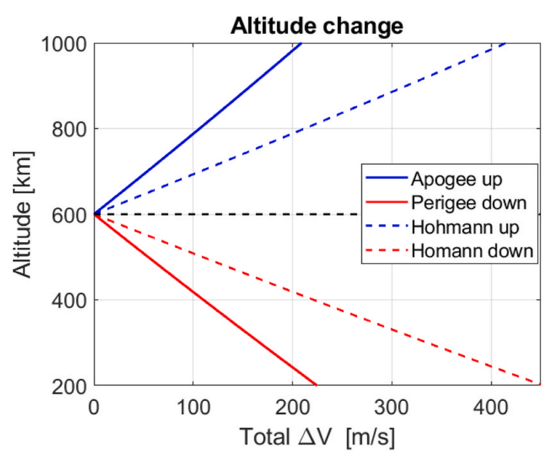

Fig. 4. Altitude changes as a function of ΔV. $i = 45^\circ$.

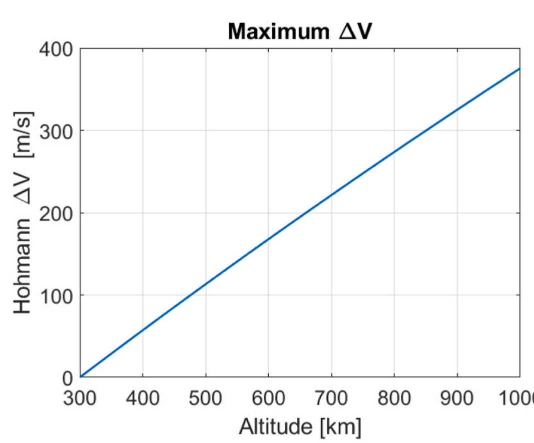

Fig. 5. Maximum ΔV applicable without dropping below 300 km for a full Hohmann transfer (one-way trip) as a function of initial altitude.

Concluding, the RAAN change can be performed effectively only for moderate variations, thus the initial orbital plane must be not too far away from the target one. Moreover, if the satellite is released in a lower altitude parking orbit, the RAAN drift can be obtained for free (in terms of propulsion but not time) simply by proper timing of the orbit raising manoeuvre.

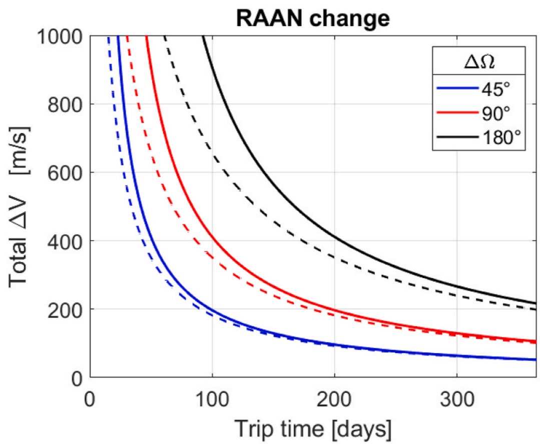

Fig. 6. ΔV as a function of trip time. $i = 45^{\circ}$. Solid lines refer to an orbit raising. Dotted lines refer to orbit drop.

3. Satellite carrier vs autonomous satellites

In this chapter the possibility to reach the final orbit with a satellite carrier is compared with the capability to do the same autonomously with the satellite own propulsion system.

A first qualitative comparison can be done to highlight the respective advantages/drawbacks. The satellite carrier must be designed to cope with a variety of situations while the satellite propulsion system can be tailored for its specific task and consequently be more efficient. The sat carrier propulsion system is bigger, thus it could exploit scale effects and be more performing, particularly respect to the smallest payloads. Regarding scale effects on the production number, it depends on the situation. The sat carrier is favoured if the same design is used multiple times to transport different satellites. On the contrary, large constellations of equal satellites (like OneWeb, Starlink etc.) favour the mass production of autonomous satellite propulsion systems. The use of COTS engines by satellites could also shift more or less the balance toward the autonomous depending on the production number of those engines. Moreover, the satellite will have its own in-orbit propulsion needs like orbit maintenance ad de-orbit, so, the more synergy can be done with the access to space needs, the more is preferable to go the autonomous way.

A simple example is a satellite originally designed with a cold gas system able to perform 50 m/s vs another one with a chemical thruster able to perform 300 m/s. If 100 m/s of ΔV need to be added for orbit raising, the first system should be probably completely redesigned and relying on a sat carrier is an easier alternative, while the second perhaps can only have the tank size extended.

Finally, the self-propelled dispenser will always carry some added inert mass (excluding propulsion), for example its own avionic and other subsystems, which can be limited thanks to miniaturization, but above all the inert structural mass required to keep all its payloads. This increases the initial mass to be transported in orbit by the launch vehicle. For example, autonomous CubeSats can release their boxes on the launch vehicle while the sat carrier has to transport them up to the final release, affecting its total impulse needs. Anyway, for the same capability, the "inert" CubeSats on the sat carrier will require smaller deployers.

The satellite carrier components could have a lower lifetime than the one of the individual satellites.

Autonomous satellites can be at least partially operational (e.g. taking images) even before reaching the final destination, while the same could not generally happen during the transport with a sat carrier.

Another aspect to be considered is that autonomous satellites will move in parallel to their final destinations while the sat carrier has to move all of them together and release them one by one. This affects both the total impulse and the total time for constellation deployment, and it is the subject of the next two paragraphs.

3.1. Altitude change

Let's consider at first the case of any manoeuvre except one of phasing (or RAAN drift). For simplicity, an altitude change will be taken as an example as it is the most interesting case. The case of only one satellite is not considered as it is trivial.

If all the satellites have to be released in a position between the starting point and the farthest one, the sat carrier ΔV will be the same of that of the farthest satellite (for the altitude this applies only in LEO where the sum of several smaller Hohmann transfers is equal to a single big one). In case of continuous thrusting, the trip time will be also the same (for equal characteristics). In case of an impulsive manoeuvre, the time for each satellite will be equal to the one of an Hohmann transfer, i.e. nearly half an orbit. On the contrary, for the sat carrier potentially one Hohmann transfer is needed for each satellite so the total time can be up to N times the one of the autonomous satellites, whit N the number of satellites. However, as half the orbital period is around 50 min, even 12 satellites will theoretically need a total of only 10 h with the sat carrier. The total impulse of the sat carrier will be the sum of the total impulses of the single satellites.

The situation becomes worse if the satellites need to be deployed in opposite direction respect to the starting point. In this case the sat carrier has to move first in one direction and the come back and go on the other direction. It is easy to determine that the worst situation in terms of ΔV is when the starting point is exactly in the middle of the two farthest satellites. In this case, the total ΔV for the sat carrier is 3 times the one of the single satellites. The trip times is also 3 times in case of continuous thrusting, while it remains the same as before for the multiple Hohmann transfers.

Regarding the total impulse the calculation becomes a bit more complicated. The sat carrier has to move multiple satellites and release them after different ΔVs. Let's consider at first the case of only two satellites of equal mass. The worst case happens when the two satellites have to be placed symmetrically at the opposite sides respect to the starting orbit. In this case the sat carrier will move 2 satellites in one place, release one satellite and then move on the other place. The total impulse will be the ΔV of the first movement multiplied by 2 masses plus 2 times the same ΔV for one mass. The total is four times the single ΔV multiplied one mass. Thus, the total impulse of the sat carrier is 2 times the sum of those of the single satellites.

The case of three satellite becomes more complex. If two satellites are on the opposite side of the farthest one, the best strategy is to release them first. For equal masses, the total impulse ratio is:

$$\frac{I_{tot\_sc}}{\sum I_{tot\_fs}} = \frac{\Delta v_1 + \Delta v_2 + \Delta v_3 + 2\Delta v_2}{\Delta v_1 + \Delta v_2 + \Delta v_3}$$

Where $\Delta v_1 \geq \Delta v_2 \geq \Delta v_3$.

If two out of three satellites are on the same side of the farthest one, three strategies are possible: release one satellite on one side and then two on the other, do the opposite, or zigzagging between the three. It is possible to demonstrate that each strategy is the best in a certain range of initial positions.

Considering $\Delta v_1 \geq \Delta v_2, \Delta v_3$ and calling Δv_2 the one on the same side of Δv_1, it is better to release first the satellite on the opposite side when $\Delta v_2 > 0.5\Delta v_3$ and $\Delta v_3 < 0.5\ \Delta v_1$:

$$\frac{I_{tot\_sc}}{\sum I_{tot\_fs}} = \frac{\Delta v_1 + \Delta v_2 + \Delta v_3 + 4\Delta v_3}{\Delta v_1 + \Delta v_2 + \Delta v_3}$$

It is better to zigzag when $\Delta v_2 < 0.5\Delta v_3$ and $\Delta v_3 < 1/(2a+1)\ \Delta v_1$ with $a = \Delta v_2/\Delta v_3$:

$$\frac{I_{tot\_sc}}{\sum I_{tot\_fs}} = \frac{\Delta v_1 + \Delta v_2 + \Delta v_3 + 4\Delta v_2 + 2\Delta v_3}{\Delta v_1 + \Delta v_2 + \Delta v_3} \quad (26)$$

In all the other situations it is better to release the lonely satellite for last:

$$\frac{I_{tot\_sc}}{\sum I_{tot\_fs}} = \frac{\Delta v_1 + \Delta v_2 + \Delta v_3 + 2\Delta v_1}{\Delta v_1 + \Delta v_2 + \Delta v_3}$$

The results are plotted in Figs. 7 and 8.

The worst case of them all (relative to the autonomous satellites) happens when all the three strategies become equivalent, that is when the relative ΔVs are −1, −0.25, 0.5 (dimensionless respect to the ΔV of the farthest satellite). In this caser the total impulse ratio is 15/7 = 2.14.

The cases for 4, 5 and 6 satellites have been solved numerically and are listed in Table 1. The worst cases for an even number of satellites are symmetric.

The total impulse is related to the propellant mass that the sat carrier has to carry onboard. That means that at first approximation, in the worst case, a satellite carrier propulsion system will weigh more than two times the weight of those of the single autonomous propelled satellites combined. This will in turn affect the launch costs.

It is worth noting that the positions that maximize the ratio between the sat carrier total impulse and the sum of the total impulses of the autonomous satellites are different respect to the condition that maximize the ratio between the sat carrier ΔV and the maximum one of the single satellites, which is always when there is at least one satellite at −1 and one at 1. Moreover, the dispenser maximum absolute (not relative) total impulse for a fixed maximum satellite ΔV happens when half of the satellites are at −1 and the others at 1. While the total impulse is directly related to the propellant mass, the ΔV value is related with the required performance level (*Isp*, *k*) of the propulsion system and the departure from linearity of the Tsiolkovsky equation. It is important to remember that the total impulse analysis of this chapter applies strictly only in the linear case, as shown in paragraph 3.3.

3.2. Phasing (or RAAN drift)

Let's now consider the case of a phasing manoeuvre or equivalently a RAAN drift. In particular, let's focus on the case where a certain number of satellites are released at the same spot and have to be uniformly distributed on the same orbit. Alternatively, in case of the RAAN drift, the satellites are released at the same spot and have to uniformly distribute their orbital planes on the equator.

The case of only one satellite is not considered as it is trivial. For multiple satellites, in case of autonomous propulsion, each satellite will modify its altitude in parallel, drift for a certain time, and then come back to the nominal orbit. The sat carrier, on the contrary, has to modify its altitude, drift for a certain time, come back to the nominal orbit, release one satellite, and then repeat the process for each satellite. As all the satellites are transported together, each time part of the sat carrier payload is moved back and forth from the nominal orbit without being released an inevitable loss occur.

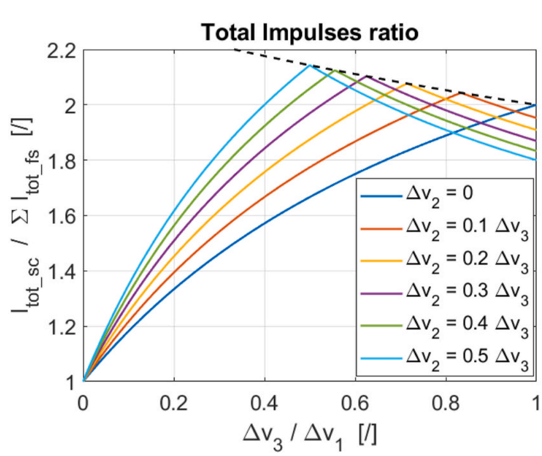

Fig. 7. Total impulse ratio between a sat carrier and 3 autonomous satellites. $\Delta v2 \leq 0.5\Delta v3$.

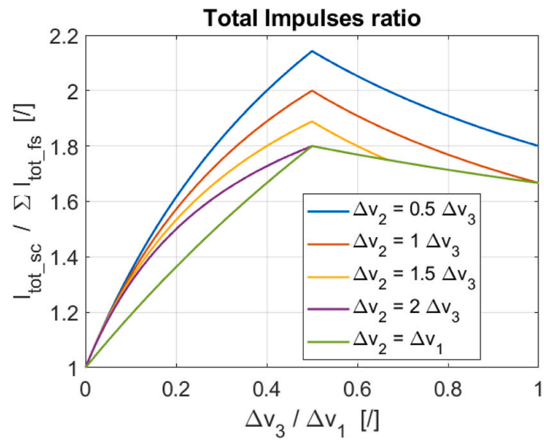

Fig. 8. Total impulse ratio between a sat carrier and 3 autonomous satellites. $\Delta v2 \geq 0.5\Delta v3$.

Table 1
Worst case total impulses for the sat carrier.

Number of satellites N	Worst case positions (dimensionless)	Total Impulse ratio
2	−1, 1	2
3	−1, −0.25, 0.5	2.14
4	−1, −0.33, 0.33, 1	2.5
5	−1, −0.44, −0.16, 0.25, 0.68	2.58
6	−1, −0.5, −0.2, 0.2, 0.5, 1	2.76

The sat carrier is forced to move along the orbit in one direction, while the autonomous satellites can move in opposite directions.

In the case of the autonomous satellites, if their number is odd, the best strategy is to keep one satellite at its original position and move the others symmetrically. If their number is even, it is possible to move all the satellites symmetrically or keep one satellite at its original position, move one at the opposite side (180°) and move the remaining ones symmetrically.

Let's consider the case where all the satellites have the same mass M and the same altitude variation is applied to all satellites, i.e. the same ΔV.

The sat carrier total ΔV will be:

$$\frac{\Delta v_{sc}}{\Delta v} = N - 1 \tag{28}$$

As the first satellite is released on the spot. The total impulse will be:

$$\frac{I_{tot\_sc}}{M\Delta v} = (N-1)N \Big/ 2$$

On the contrary the sum of the total impulses of the autonomous satellites will be:

$$\frac{\sum I_{tot\_fs}}{M\Delta v} = N - 1$$

In the strategy where one satellite is always released on the spot (also in the even case) while for the other solution:

$$\frac{\sum I_{tot\_fs}}{M\Delta v} = 2\,floor(N/2)$$

The results are plotted in Fig. 9.

The total impulse ratio is:

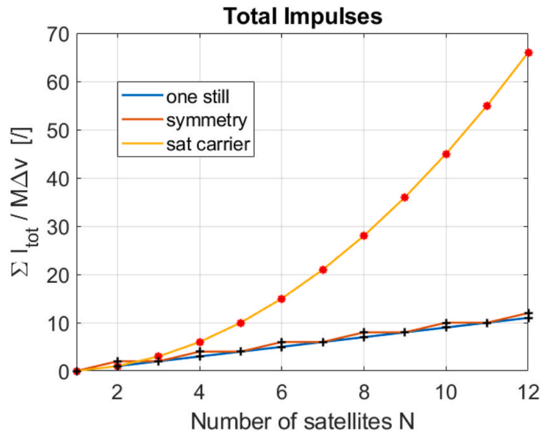

Fig. 9. Total impulses as a function of the number of satellites.

$$\frac{I_{tot\_sc}}{\sum I_{tot\_fs}} = N \Big/ 2$$

in the first case and:

$$\frac{I_{tot\_sc}}{\sum I_{tot\_fs}} = floor((N-1)/2) + 1 \Big/ 2$$

In the second case. Consequently, both the ΔV and the total impulse ratio growth linearly with the number of satellite N (Fig. 10).

This shows that the satellite carrier becomes much more inefficient compared to the autonomous satellites as the number of satellites increases. Thus, for a large number of satellites, it is probably preferable to use multiple satellite carriers instead of a bigger one. It is worth remembering that what counts is the number of satellites to be released in different places, not the total number of satellites carried. If multiple satellites have to be released in the same position, they simply behave as a single bigger one.

In case of a phasing manoeuvre, as mentioned in the previous chapter, the ΔV is small, so perhaps the sat carrier can cope anyway with its inherent inefficiency. The situation becomes much more critical in the case of a multiple RAAN drift.

As shown in Eq. 10, the trip time for a phasing manoeuvre is proportional to the angle covered and inversely proportional to the ΔV applied. The total time for the satellite carrier is:

$$\frac{\Delta t_{tot\_sc}}{\Delta t_{180^\circ}} = 2(N-1) \Big/ N \tag{34}$$

while the total time for the autonomous satellites will be the equal to the time needed by the farthest satellite. In the first strategy:

$$\frac{\Delta t_{tot\_fs}}{\Delta t_{180^\circ}} = 2\,floor(N/2) \Big/ N \tag{35}$$

while in the second:

$$\frac{\Delta t_{tot\_fs}}{\Delta t_{180^\circ}} = (N-1) \Big/ N \tag{36}$$

The ratio of times compared to a 180° drift is also the ratio of the maximum angles travelled (Fig. 11).

The ratio of times (and angles) between the sat carrier and the autonomous satellite is:

$$\frac{\Delta t_{tot\_sc}}{\Delta t_{tot\_fs}} = 4(floor((N-1)/2) + 1/2) \Big/ N \tag{37}$$

For the first strategy, which always tends asymptotically to 2 because the sat carrier has to cover an entire orbit while the farthest satellite position is 180°. In the second strategy the ratio is always 2 (Fig. 12).

Therefore, the sat carrier will take more time to deploy all its payloads. The nearly first half of the satellites (i.e. from 0 to 180°) will be deployed at the same time, while the second half (from 180° to the last point, which asymptotically tends to 360°) will take more and more time compared to the direct case as the sat carrier approach the original point.

Up to now the comparison consider the same altitude variation, i.e. ΔV. Remembering that the trip time for a phasing manoeuvre is inversely proportional to the ΔV, it is possible to vary its value to adjust the trip time. It is therefore useful to multiply the total impulse with the trip time. The result can be interpreted as the required total impulse for a fixed trip time equal for all options or equivalently, the trip time for a fixed total impulse equal for all options. For the autonomous satellites the product is always the same. In fact, for an even number of satellites, the faster strategy 2 can be applied with a proportionally smaller ΔV in order to obtain the same trip time as strategy 1. The final total impulse becomes now the same. It is easy to visualize this result in the case of only two satellites. For strategy 1, one satellite will remain still and the other will apply a certain ΔV to drift by 180°. For strategy 2, both satellites will have to apply a halved ΔV to drift by±90° in the same time as strategy 1. The sum of the MΔVs remains the same.

The total impulse ratio between the sat carrier and the autonomous satellites for a fixed total trip time becomes:

$$\frac{I_{tot\_sc}\Delta t_{tot\_sc}}{\sum I_{tot\_fs}\Delta t_{tot\_fs}} = 2\,floor((N-1)/2) + 1$$

Which is obviously 2 times Eq. 33, doubling the propellant mass penalty (Fig. 13).

For the same trip time, the sat carrier will deploy the first half of the

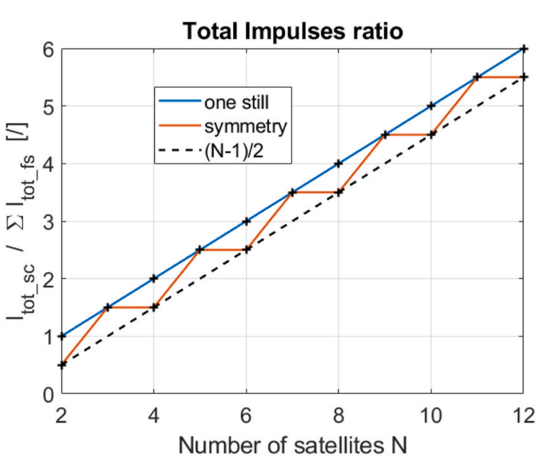

Fig. 10. Total impulses ratio as a function of the number of satellites.

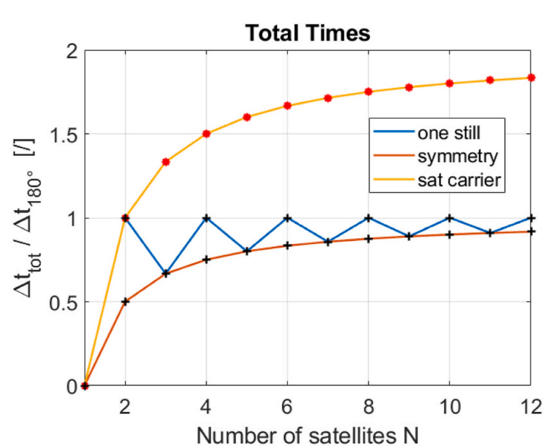

Fig. 11. Total times as a function of the number of satellites.

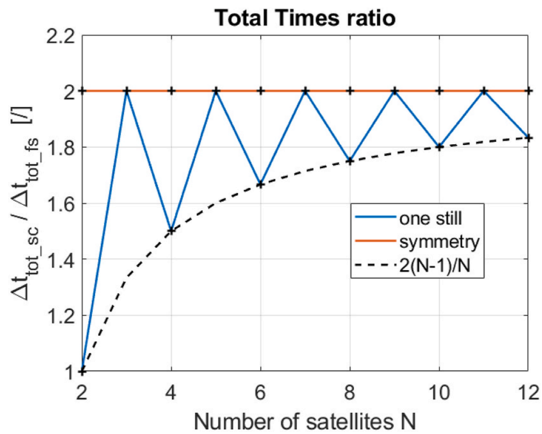

Fig. 12. Total times ratio a function of the number of satellites.

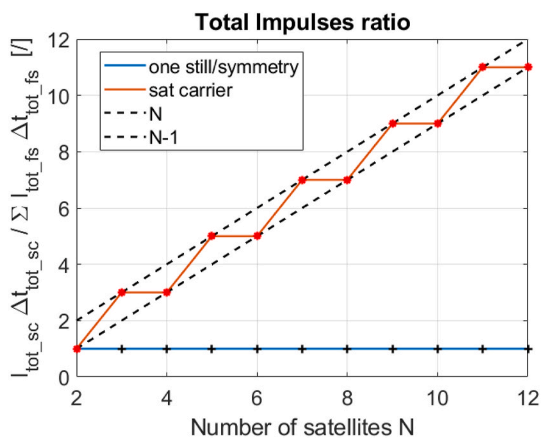

Fig. 13. Total impulses ratio a function of the number of satellites for a fixed trip time.

satellites (from 0 to 180°) twice as fast. The satellites from 180° to 240° will be deployed between two times and one time as fast as the autonomous satellites. The satellites from 240° to 270° will be deployed between one time and 1.5 times slower than in the case of the autonomous satellites. Finally, the last satellites (from 270° to 360°) will take from 1.5 to infinite times slower than the autonomous satellites.

The rate of satellites deployment obviously remains the same, as the double speed of the satellite carrier on one side is compensated by the doubled symmetric delivery of the autonomous satellites.

In all situations considered so far, all satellites are delivered at different times. For the sat carrier there is no alternative. For the autonomous satellites, this implies the same altitude variation and propulsion needs for all satellites; but, if these constraints can be dropped and the interest is only on the total deployment time, the nearest and thus faster satellites can be slowed down imparting a proportionally smaller ΔV in order to reach the final destination all at the same time.

For a constant ΔV strategy the product of the total impulse and the total time is:

$$\frac{\sum I_{tot\_\Delta v}\Delta t_{tot\_\Delta v}}{M\Delta v_{max}\Delta t_{180^\circ}} = 2\,floor(N/2)(N-1)\Big/N$$

While for a constant time strategy the product of the total impulse and the total time is:

$$\frac{\sum I_{tot_t}\Delta t_{tot_t}}{M\Delta v_{max}\Delta t_{180^\circ}} = 2\,floor\left(\frac{N+1}{2}\right)$$

$ceil((N-1)/2)/N$

The results are plotted in Fig. 14. The ratio between the two solutions is:

$$\frac{\sum I_{tot\_\Delta v}\Delta t_{tot\_\Delta v}}{\sum I_{tot\_t}\Delta t_{tot\_t}} = (N-1)\Big/ceil(N/2)$$

Which asymptotically tends to 2 (Fig. 15). This potentially translates in an equivalent mass saving at launch. However, this also implies that the satellites have different propellant tanks, and this could not be optimal from a production point of view. Alternatively, for the same tanks and launch weight, some satellites will have higher margins on the residual total impulse to be used. How much this is beneficial for the mission is difficult to establish in general but depends on the specific case.

Even the sat carrier has the possibility to optimize its total impulse. In fact, each time a satellite is released, the total mass decreases, so the following ΔV requires a lower total impulse. Consequently, it is possible to think that, for the same total time, it will be favourable to accelerate the sat carrier as it becomes lighter.

This strategy can be applied starting from 3 satellites. The first is left on spot, two are moved to the second position where the second satellite is released and finally the last satellite is moved in its final destination. Considering that the total time is the sum of the two transfer times and that each transfer time is inversely proportional to its ΔV, while the whole total impulse is the sum of the total impulses of the two transfers, it is possible to write the product of the total impulse with the total time as a function of only a single ΔV. Afterwards it is possible to find its minimum point taking the derivative equal to zero. In this way it is possible to calculate the optimal ΔVs that minimize the total impulse for a fixed total time:

$$\Delta v_1\Big/\Delta v_{ref} = 0.5\left(1+1\Big/\sqrt{2}\right) = 0.8536$$

$$\Delta v_2\Big/\Delta v_{ref} = \sqrt{2}\Delta v_1\Big/\Delta v_{ref} = 1.2071$$

Where *ref* refers to the case of equal repartition of ΔVs (see Fig. 16). Δt_{ref} is the transfer time of a single ΔV, while Δt_{tot} is the sum of the two (see Fig. 17).

As expected, in the optimal case the first ΔV is smaller as it is multiplied by two masses while the second is multiplied only by one mass. The total impulse obtained at the optimal point is:

$$\frac{I_{tot}}{I_{tot_{ref}}} = \frac{1}{3}\left(1+\frac{1}{\sqrt{2}}\right)^2 = 0.9714$$

Which is only 3 % less than for equal repartition.

The cases for 4, 5, 6, 7 and 8 satellites have been solved numerically. The results are presented in Tables 2 and 3. The first satellite ΔV is

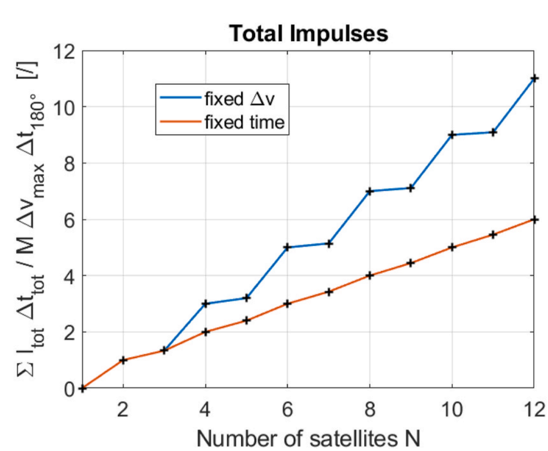

Fig. 14. Total impulses for a fixed ΔV strategy and a fixed time strategy as a function of the number of satellites.

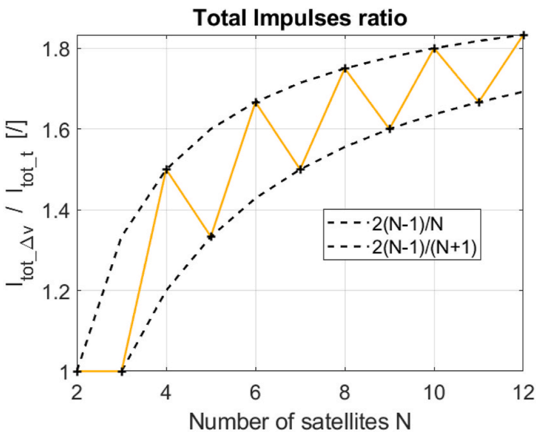

Fig. 15. Total impulses ratio between a fixed ΔV strategy and a fixed time strategy as a function of the number of satellites.

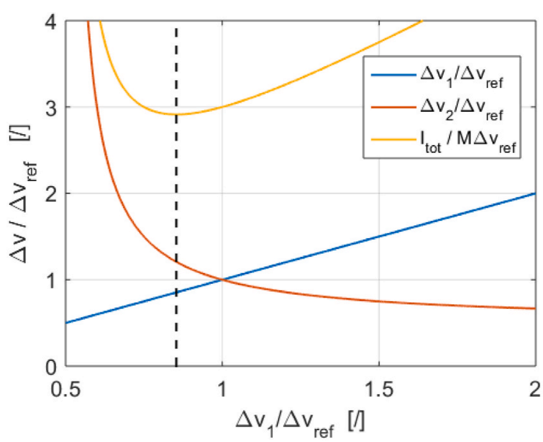

Fig. 16. ΔV repartition as a function of the first ΔV. 3 satellites phasing with sat carrier.

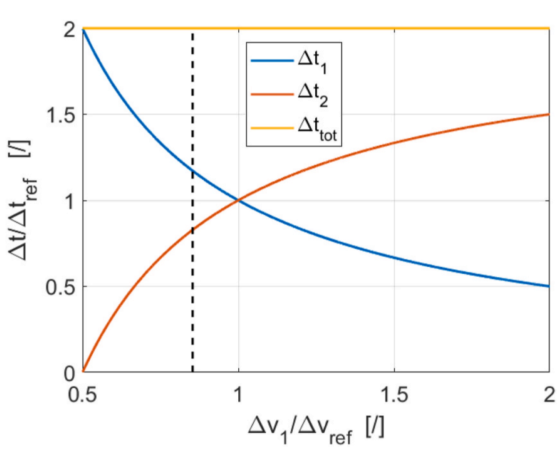

Fig. 17. Time repartition as a function of the first ΔV. 3 satellites phasing with sat carrier.

always null and only the ΔVs of the others that are moved are reported in Table 2. As expected, in the optimal situation the sat carrier needs to continuously accelerate as more mass is discharged.

The advantage of optimizing the sat carrier ΔVs growth very slowly with the number of satellites compared to the general penalty obtained respect to the autonomous satellites.

Table 2
Optimal ΔVs for a sat carrier performing a phasing manoeuvre.

Number of satellites *N*	Optimal ΔVs
3	0.85, 1.21
4	0.80, 0.98, 1.38
5	0.77, 0.89, 1.08, 1.53
6	0.75, 0.84, 0.97, 1.18, 1.67
7	0.74, 0.80, 0.90, 1.05, 1.27, 1.83
8	0.74, 0.79, 0.87, 0.96, 1.11, 1.37, 1.78

Table 3
Total impulse ratio between optimal and equal repartition of ΔVs for a sat carrier performing a phasing manoeuvre.

Number of satellites *N*	Total Impulse ratio
3	0.971
4	0.955
5	0.944
6	0.937
7	0.931
8	0.927

3.3. Non-linearity

The analyses in the previous paragraphs have been performed considering a fixed mass M of the satellites in the calculation of the total impulses. This allowed to find simplified analytical or numerical solutions to the sat carrier comparison problem. In reality, the mass will vary as the propellant is consumed. This non-linearity is taken into account in the Tsiolkovsky equation:

$$m_{prop} = m_f\left(e^{\frac{\Delta v}{Isp\ g_0}} - 1\right) \tag{45}$$

Figs. 18 and 19 show the difference in the calculated propellant mass when a constant mass is used instead. Using the initial mass in a linear calculation will underestimate the results while using the final mass will overestimates them (see Figs. 20–21-22).

The nonlinearity is a function of the dimensionless parameter ΔV/(Ispg0), which in turn determines univocally the ratio between the propellant mas and the initial and final masses

In the previous analyses the mass M was the ‘operational’ satellite mass common to both transport solutions, i.e. without the farther mass needed to move into the final destination. Consequently, the previous results underestimate the total impulse, the propellant and total mass of both the autonomous satellites and the satellite carrier, with the latter being more affected as its total ΔV is higher.

Coming back to the altitude example, let’s consider the case of two

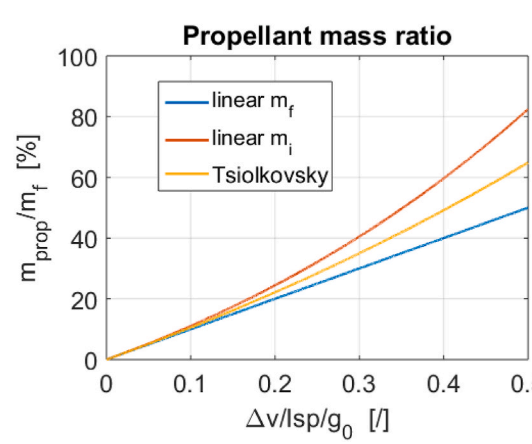

Fig. 18. Propellant mass divided by final mass as a function of the dimensionless ΔV.

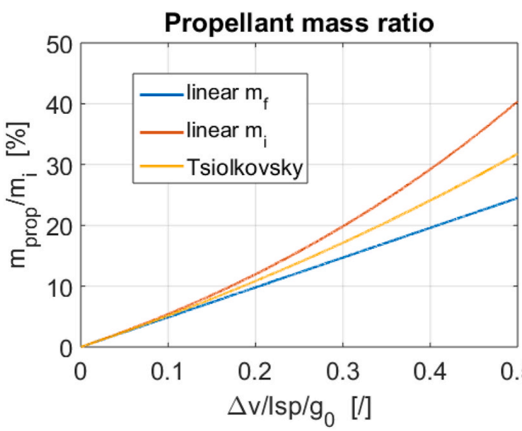

Fig. 19. Propellant mass divided by initial mass as a function of the dimensionless ΔV.

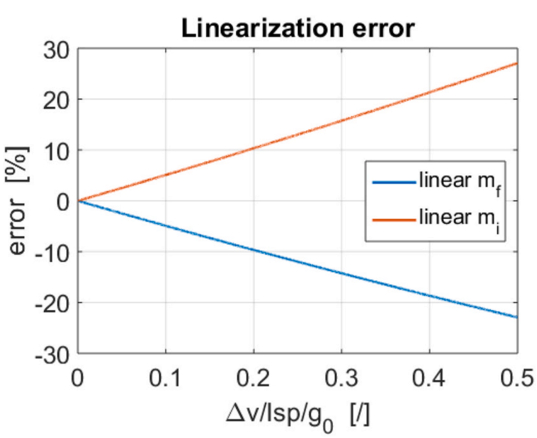

Fig. 20. Linearization error as a function of the dimensionless ΔV.

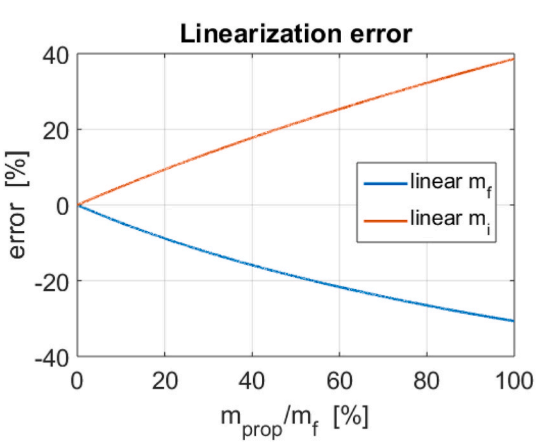

Fig. 21. Linearization error as a function of the propellant mass divided by final mass.

satellites placed on opposite sides. For $\Delta V/(Ispg_0) = 0.1$ (i.e. nearly 300 m/s for a 300 s *Isp*) the satellite propellant mass is 5 % higher than the linear prediction while for $\Delta V/(Ispg_0) = 0.5$ (i.e. nearly 1500 m/s for a 300 s *Isp*, a fairly remarkable value) the satellite propellant mass is 30 % higher than the linear prediction. In the case of the sat carrier, for $\Delta V/(Ispg_0) = 0.1$ the total propellant mass is 14 % higher than the linear prediction while for $\Delta V/(Ispg_0) = 0.5$ the satellite propellant mass is 2.1 times the linear prediction. The ratio of the propellant masses between

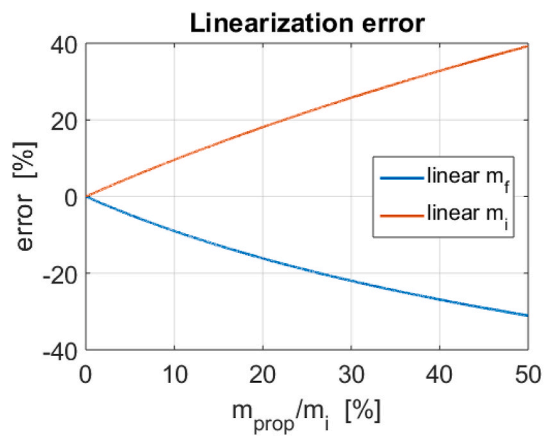

Fig. 22. Linearization error as a function of the propellant mass divided by initial mass.

the sat carrier and the autonomous satellites increase from 2 to 2.2 for $\Delta V/(Ispg_0) = 0.1$, while for $\Delta V/(Ispg_0) = 0.5$ the ratio increases from 2 to 3.2.

Regarding the case of 3 satellites, one placed on one side and the other two on the opposite side, all at the same distance (e.g. −1, 1, 1), for $\Delta V/(Ispg_0) = 0.1$ the total propellant mass of the sat carrier is 12 % higher than the linear prediction while for $\Delta V/(Ispg_0) = 0.5$ the satellite propellant mass is 91 % higher than the linear prediction. The ratio of the propellant masses between the sat carrier and the autonomous satellites increase from 5/3 = 1.67 to 1.8 for $\Delta V/(Ispg_0) = 0.1$, while for $\Delta V/(Ispg_0) = 0.5$ the ratio increases to 2.5.

Regarding the case of 3 satellites, placed on the previously calculated worst case (−1, −0.25, 0.5), for $\Delta V_{max}/(Ispg_0) = 0.1$ the total propellant mass of the satellites is 4 % higher than the linear prediction while for $\Delta V_{max}/(Ispg_0) = 0.5$ the satellites propellant mass is 22 % higher than the linear prediction. For the sat carrier, the total propellant mass is 11 % higher than the linear prediction for $\Delta V_{max}/(Ispg_0) = 0.1$ while for $\Delta V_{max}/(Ispg_0) = 0.5$ the satellite propellant mass is 75 % higher than the linear prediction. The ratio of the propellant masses between the sat carrier and the autonomous satellites increase from 15/7 = 2.14 to 2.3 for $\Delta V/(Ispg_0) = 0.1$, while for $\Delta V/(Ispg_0) = 0.5$ the ratio increases to 3.1.

In the case of phasing, both the single satellites and the sat carrier will require more total impulse and propellant that previously predicted, with the sat carrier penalty being higher as it is its total ΔV. Moreover, the non-linear penalty of the sat carrier will also increase with the number of satellites.

Regarding the optimum point, the non-linearity will force the repartition of the ΔVs to even out to limit the exponential effect of Tsiolkovsky equation. The results for 3 satellites and $\Delta V/(Ispg_0) = 0.5$ are shown in Fig. 23. In this case the satellite propellant mass is 1.3 times higher than the linear prediction while the sat carrier propellant is 1.6 times higher than the linear prediction.

The situation becomes even worse when it is considered that the propulsion system inert mass is dependent on the propellant mass trough the structural index k, so the propellant mass is given by:

$$m_{prop} = \frac{m_{pay}\left(e^{\frac{\Delta v}{Isp\ g_0}} - 1\right)}{1 - k\left(e^{\frac{\Delta v}{Isp\ g_0}} - 1\right)} \tag{46}$$

Where the structural index is related to the inert mass fraction:

$$k = \frac{\varepsilon}{1 - \varepsilon} \tag{47}$$

The highest the structural index the higher is the sensitivity to the ΔV, exacerbating the mass penalty for the sat carrier, which has to

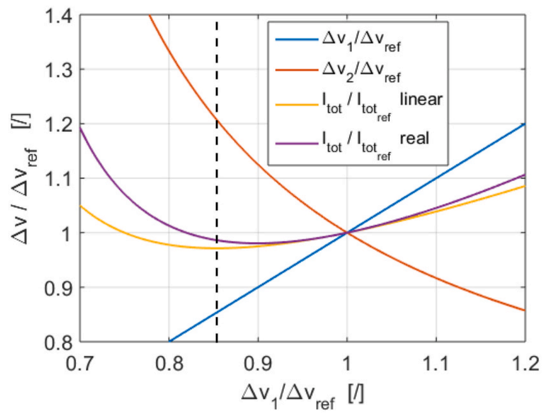

Fig. 23. ΔV repartition as a function of the first ΔV. 3 satellites phasing with sat carrier. $\Delta V/(Ispg_0) = 0.5$.

generates a higher ΔV than the single satellites.

Concluding, the sat carrier is theoretically more inefficient than autonomous propulsion, and the disadvantage increases rapidly with the number of satellites and the ΔV. Moreover, as seen by the previous worst-case examples, particularly for the altitude, the penalty also depends on the higher/lower capability/possibility of the sat carrier provider to arrange together multiple satellites with similar needs.

Real world examples confirm the trend obtained in this study. In fact, larger constellations of high performing satellites with relevant propulsion capabilities like Starlink, OneWeb, Iceye, SkySat etc. have chosen the path of autonomous propulsion as this choice presents less drawback at such scale (both in terms of size and numbers) and provides higher overall efficiency (both technical and economical).

On the opposite, looking to the launch manifest of today most successful self-propelled deployer, D-Orbit ION [25], the payloads list is always varied, full of CubeSats conceived for a single mission that could not afford to consider the last mile delivery into the design. At the same time, sometimes similar CubeSats belonging to constellations are also present, like Astrocast, Kleos, SatRevolution, Apogeo Space, Swarm Technologies, Kepler, Spire Global, Planet Labs and others. Even in these cases the simplification in the satellite architecture is worthy the reduction in the deployment capabilities and the occasional need for a specific external service.

The Dove satellites of the Flock constellations have the capability to perform phasing manoeuvres using differential drag, but at the cost of a reduced lifetime [37–40]. Consequently, if possible, they prefer to be placed as near as possible to their intended position. On the contrary, the chemical monopropellant propulsion system of the larger SkySats can easily afford to spend around 10 % of its ΔV budget for the phasing [63].

It is worth noting that the SkySat and Flock constellations, while belonging to the same company (Planet Labs PBC, even if SkySat was initially developed by Terra Bella) are using different strategies better tailored to their specific characteristics.

4. Orbit raising trade-off

Generally, the optimal solution is to place the satellite in the proper position. If the satellite is placed in a different position, the satellite has to move in the right place at the expense of a reduced payload mass for the same initial mass. However, in the case of an orbit raising, the energy provided by the satellite or sat carrier is saved by the launch vehicle. Thus, the initial mass into orbit can increase. Consequently, it is interesting to assess if it is more efficient to go directly at the required altitude with the launch vehicle or perform an orbit raising with the satellite. As an example, the ABL RS1 launcher has been considered [20]. An orbit raising has been performed starting from the initial mass at each altitude. The payload mass is the initial mass minus the propulsion mass. The propulsion mass is the sum of the propellant mass and the inert mass, which is k times the propellant mass. The calculation has been performed for several combinations of satellite propulsion system specific impulse and structural index k, see Figs. 24–31.

As expected, the autonomous orbit raising provides an increase of payload when the specific impulse is high and the inert mass fraction is low. However, as the orbit raising manoeuvre behaves as an added stage, the performance of the latter (*Isp* and k) does not need to be as good as the one of the launcher's upper stage to get a net advantage. This is important as it is almost always the case, except when electric propulsion is employed. In the example, an *Isp* of 300 s and a $k = 0.25$ are sufficient to get a marginal benefit at high altitudes. The advantage improves as the total ΔV is increased, as expected from the non-linearity of the Tsiolkovsky equation. Consequently, the results for an SSO orbit are superior compared to a low inclination orbit. In fact, in this case an *Isp* of 250 s or a k of 0.5 are sufficient for some marginal gains at high altitudes.

Electric thrusters provide huge benefits as they flatten the original payload vs altitude curve of the launch vehicle. The problems in this case are: required power, time to reach the final destination and minimum release altitude to avoid atmospheric drag during the initial part of the long duration orbit raising.

Obviously, everything else being equal, a higher number of stages of the original launch vehicle decreases the benefit of autonomous orbit raising. Storable propellants also favour a more flexible in-orbit mobility. In fact, sometimes the last stage of the launch vehicle is already a sort of space tug capable of multiple precise orbit injections, like for instance in the case of Vega AVUM [64], the Russian Fregat [65] and Briz [66], or Pegasus HAPS [67]. In this case, autonomous orbit raising is less attractive as it becomes a sort of duplication of the already available capabilities.

The same calculation has been repeated for the SpaceX Falcon 9 v1.0 [68], see Figs. 32–39. The results are similar but in this case the advantage is lower and the break-even point requires higher performance, at least better than $Isp = 300$ s and $k = 0.25$. This is expected as larger launchers tend to have higher performance (both *Isp* and inert mass fraction) so the same should apply to the autonomous orbit raising system to get an advantage. Again, the SSO case (and higher altitudes) gives the highest benefits.

It is worth noting that current satellite carriers do not achieve such value of inert mass fraction ($k = 0.25 \rightarrow \varepsilon = 0.2$). Moreover, it is also worth noting that a lot of small satellites requires some kind of deployers (e.g. Cubesats), which represent an inert mass that has to be lifted in orbit. Reaching the final destination with the satellite propulsion reduces the ΔV applied to this inert mass.

An advantage of the autonomous orbit raising is the possibility to obtain an almost free change of phase and/or RAAN with the proper

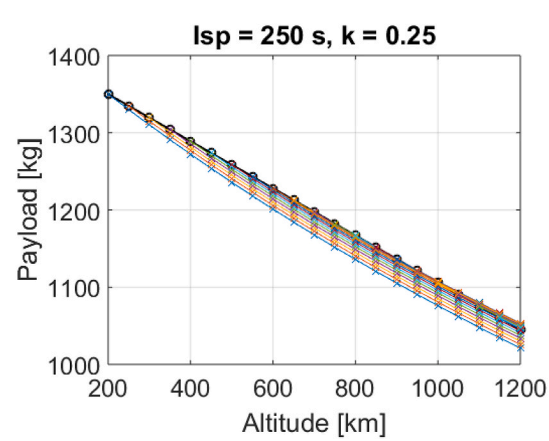

Fig. 24. Payload of the RS1 launcher with autonomous orbit raising. $Isp = 250$ s, $k = 0.25$. Orbit inclination: 28.5°.

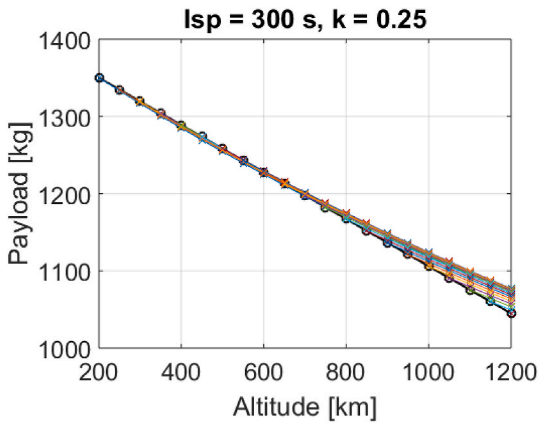

Fig. 25. Payload of the RS1 launcher with autonomous orbit raising. $Isp = 300$ s, $k = 0.25$. Orbit inclination: 28.5°.

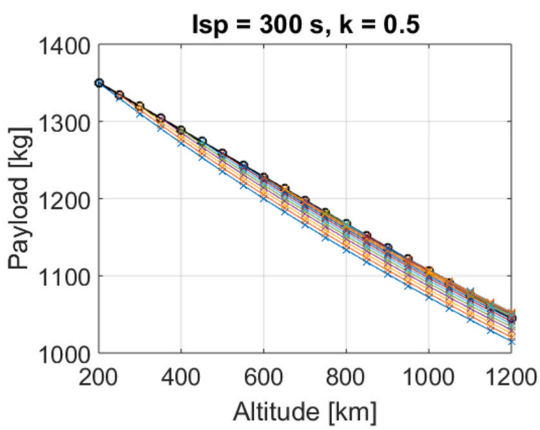

Fig. 26. Payload of the RS1 launcher with autonomous orbit raising. $Isp = 300$ s, $k = 0.5$. Orbit inclination: 28.5°.

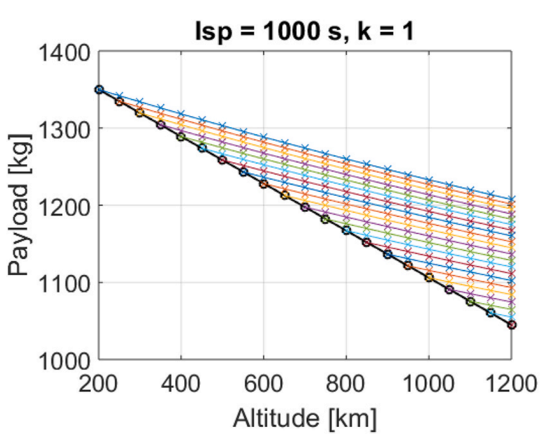

Fig. 27. Payload of the RS1 launcher with autonomous orbit raising. $Isp = 1000$ s, $k = 1$. Orbit inclination: 28.5°.

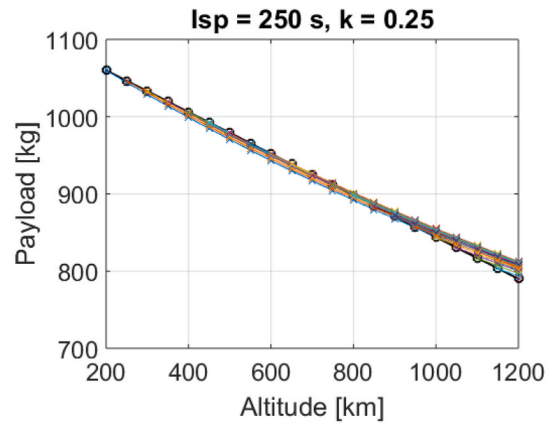

Fig. 28. Payload of the RS1 launcher with autonomous orbit raising. $Isp = 250$ s, $k = 0.25$. SSO Orbit.

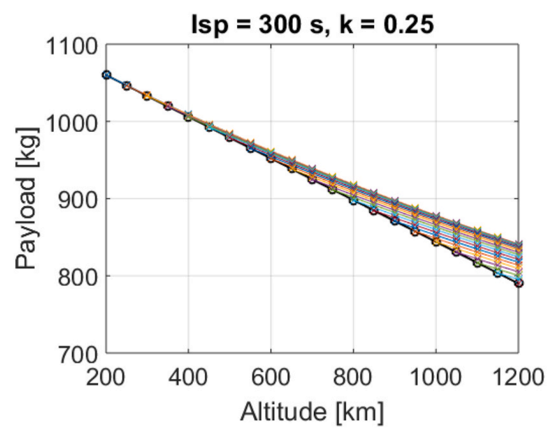

Fig. 29. Payload of the RS1 launcher with autonomous orbit raising. $Isp = 300$ s, $k = 0.25$. SSO Orbit.

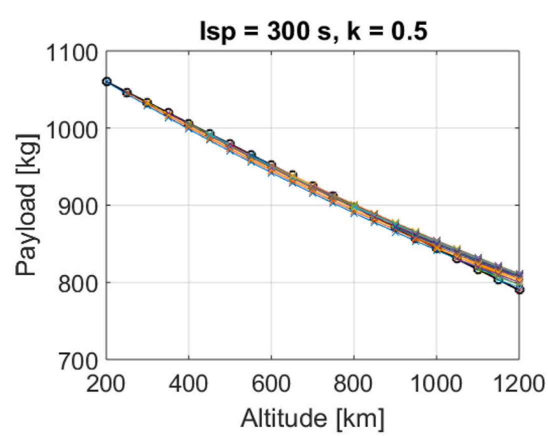

Fig. 30. Payload of the RS1 launcher with autonomous orbit raising. $Isp = 300$ s, $k = 0.5$. SSO Orbit.

timing of the respective starts of the orbit raising between different satellites. Another advantage is the possibility to perform a preliminary satellite commissioning at low altitude before the orbit raising in order to mitigate the orbital debris issue, as done by SpaceX Starlink constellation [69]. A further advantage is a generally more precise orbit injection due to smaller masses and thrusts involved, one of the reasons why several launch vehicles add a smaller kick stage [70]. A final benefit is the relaxation of time constraints, as the launcher upper stage can de-orbit while the satellites are still far away from reaching their final destination.

Another aspect to take into account is the total ΔV at which the launch vehicle is optimized. Let’s consider for the sake of simplicity a two-stage launch vehicle with equal performing stages ($Isp = 330$ s and $k = 0.075$). The launch vehicle is optimized in one case at 10 km/s of total ΔV and in the other at 14 km/s. For equal performing stages, the optimal point is an equal repartition of ΔVs, i.e. the total value divided

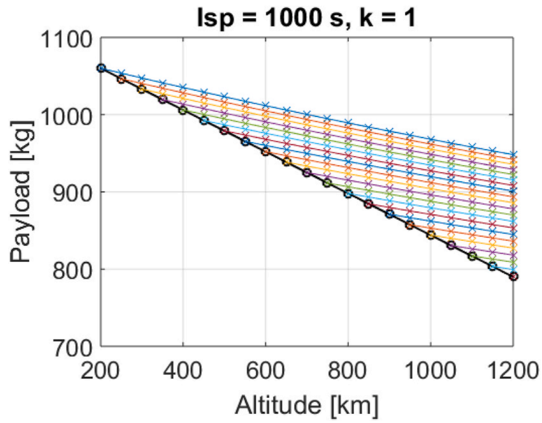

Fig. 31. Payload of the RS1 launcher with autonomous orbit raising. *Isp* = 1000 s, *k* = 1. SSO Orbit.

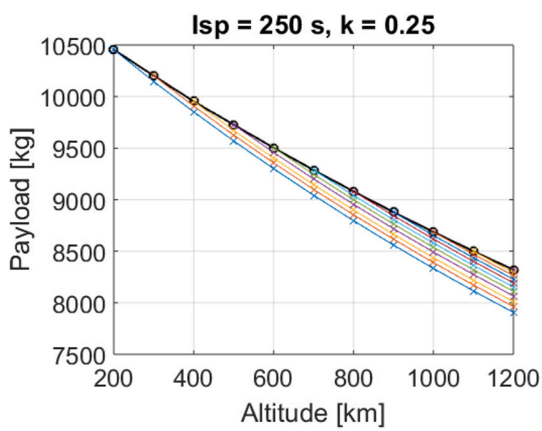

Fig. 32. Payload of the F9 launcher with autonomous orbit raising. *Isp* = 250 s, *k* = 0.25. Orbit inclination: 28.5°.

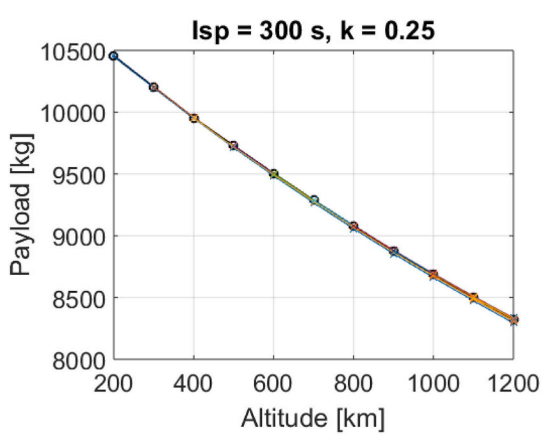

Fig. 33. Payload of the F9 launcher with autonomous orbit raising. *Isp* = 300 s, *k* = 0.25. Orbit inclination: 28.5°.

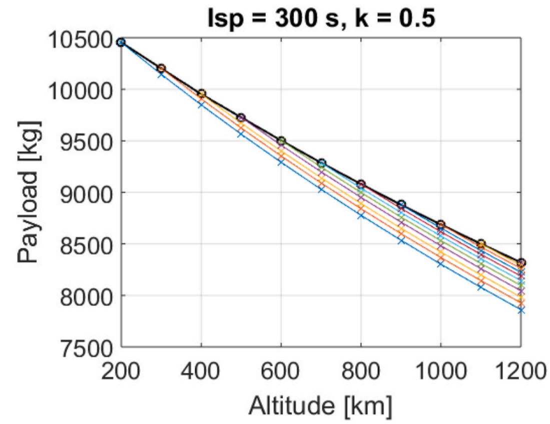

Fig. 34. Payload of the F9 launcher with autonomous orbit raising. *Isp* = 300 s, *k* = 0.5. Orbit inclination: 28.5°.

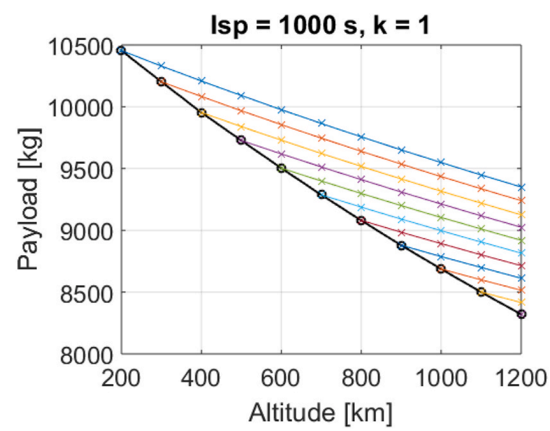

Fig. 35. Payload of the F9 launcher with autonomous orbit raising. *Isp* = 1000 s, *k* = 1. Orbit inclination: 28.5°.

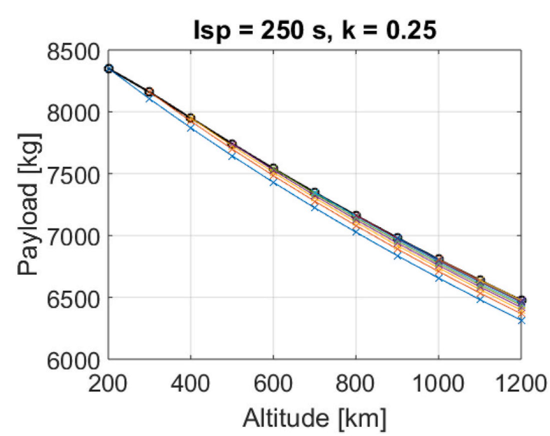

Fig. 36. Payload of the F9 launcher with autonomous orbit raising. *Isp* = 250 s, *k* = 0.25. SSO Orbit.

by the number of stages (dotted black line in Figs. 40 and 41). For a defined vehicle, when the total ΔV is increased the payload decreases (Fig. 42) and the repartition changes (red and blue lines in Figs. 40 and 41). The upper stage is always the most effected by the payload change. As shown in Fig. 42 under the standard curves, the low ΔV optimized launcher has a higher maximum payload but a lower maximum ΔV. An autonomous orbit raising is then considered starting from 10 km/s for both launchers. In this case the 2-stages ΔVs remain the same for any final payload mass and the total ΔV increases due to the added manoeuvre (yellow line).

Again, the payload mass is calculated as the initial mass minus the propulsion system mass. The propulsion mass is the sum of the propellant mass and the inert mass, which is $k = 0.25$ times the propellant mass. The specific impulse is 300 s.

As expected, the autonomous orbit raising perform in the best way when the launch vehicle is optimized for the initial release altitude. The

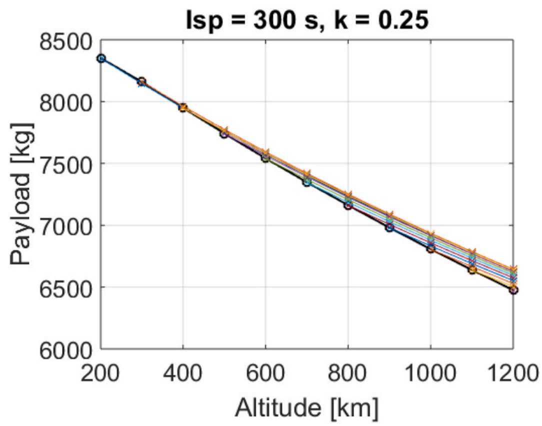

Fig. 37. Payload of the F9 launcher with autonomous orbit raising. *Isp* = 300 s, *k* = 0.25. SSO Orbit.

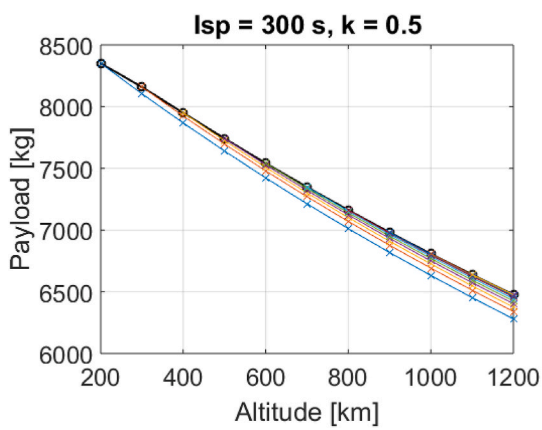

Fig. 38. Payload of the F9 launcher with autonomous orbit raising. *Isp* = 300 s, *k* = 0.5. SSO Orbit.

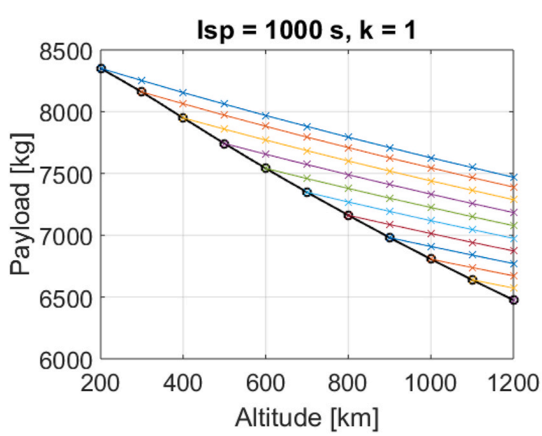

Fig. 39. Payload of the F9 launcher with autonomous orbit raising. *Isp* = 1000 s, *k* = 1. SSO Orbit.

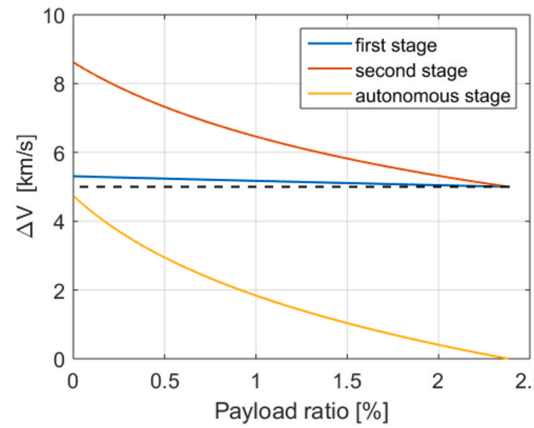

Fig. 40. Stage ΔV as a function of the payload ratio. *Isp* = 300 s, *k* = 0.25. Launcher optimized at 10 km/s.

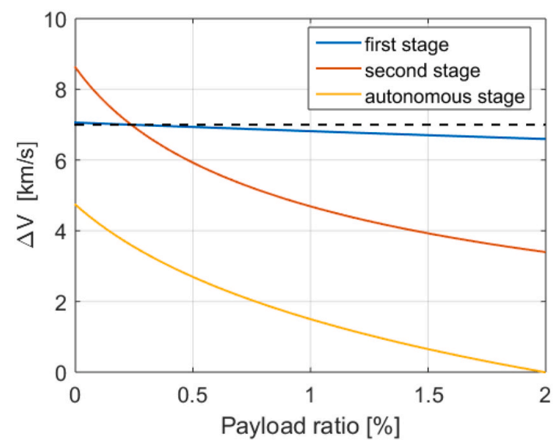

Fig. 41. Stage ΔV as a function of the payload ratio. *Isp* = 300 s, *k* = 0.25. Launcher optimized at 14 km/s.

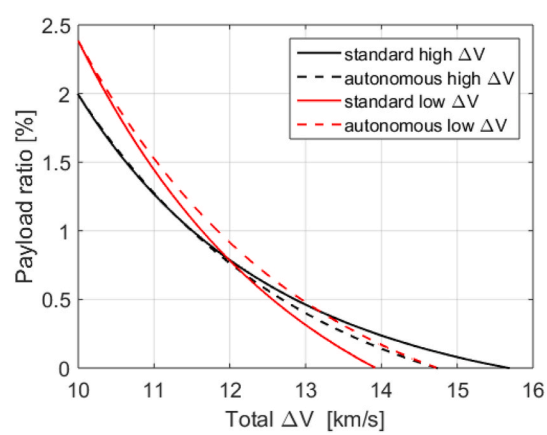

Fig. 42. Payload ratio as a function of the total ΔV. *Isp* = 300 s, *k* = 0.25.

maximum total ΔV is always the initial plus the liming case of the autonomous propulsion system, which depends on its specific impulse and inert mass fraction. To obtain higher total ΔVs is necessary to improve the performance of the autonomous system and/or increase the launcher ΔV at release (i.e. shifting up the ΔVs' optimization point and decreasing the launcher released mass).

The analysis just shown highlights that when operating with an autonomous orbit raising the complete access to space ΔV budget can often be suboptimal. This is probably not a huge concern for large systems in the category of the F9 where margins are higher. However, small launch vehicles are very sensitive to scale effects (already when expendable, even more if reusable) on costs and masses, and are more affected by the choice of the launch vehicle design point. Thus, trying to satisfy any possible situation (single dedicated launch, multiple satellite launch, satellite carrier launch, different orbits ad so on …) can result in a design that underperform (in terms of costs and/or performance) in the majority of cases. Consequently, it is possible to think that such small launchers could be favourably designed to be optimized for a relatively low ΔV that guarantees a reasonable minimum orbital altitude at each possible inclinations (with SSO as the general worst case) and leave the remaining burden on the satellites or on an ad-hoc optional space tug. This advantage can be further appreciated because very low altitudes can be still reached efficiently with a fast direct ascent instead of a slow two-burn Hohmann-like transfer, simplifying the launcher architecture (see the Pegasus rocket [67]). Anyway, some complexity is shifted on the payload, but this could be managed particularly if a standard is defined between the launch provider and the corresponding payloads or even better if the launch provider also supplies the satellite platforms (directly or in consortium) and provides a more comprehensive service. This could help to create synergy and integration between the propulsion needs for access to space and those for the rest of mission accomplishment, providing at the same time a higher degree of responsiveness and flexibility.

In fact, the possibility to create efficient specific standards could (should) be also pushed by governments as one of the main market for small launch vehicles is on demand, responsive, flexible (and independent for smaller nations) access to space of tactical payloads. This aspect will be also discussed in the next chapter on the costs analysis.

Somehow the Rocket Lab Photon platform based on the Electron launch vehicle Kick Stage with the Curie engine [70] provides an example of this potential path.

Standardization can also help to minimize the unfavourable inert mass related to multiple payloads, for examples using flat satellites with specific connections as in the case of Starlink or DiskSat [71,72].

Another point to highlight is that VLEO missions are an interesting option for small launchers, not only for mitigating the space debris problem and to increase satellite performance for Earth observation [73–78], but also because they allow increasing the mass to orbit of the launch vehicle, reducing specific costs. On the opposite, if a VLEO mission is launched on a conventional vehicle that is directed to a more conventional altitude there is a double loss, the energy to reach the parking orbit and the energy to drop the altitude up to operational level. Moreover, there is also the risk that in case of initial satellite malfunctioning the nominal debris mitigation advantage is lost.

5. Cost analysis

The final consideration to be done on the topic of small satellite access to space is a cost comparison between the different options. For this purpose, a very simple model has been conceived in order to help having some useful insights. The model compares a dedicated launch with a rideshare solution considering the added propulsive capabilities needed to reach the final destination.

The mass that the conventional launch vehicle has to lift up to orbit is the sum of the mobile mass and the inert mass of the payload adapters that remains on the upper stage:

$$m_{lv} = m_i(1 + k_1) \quad (48)$$

In some complex configurations this mass is significantly higher than that of a conventional single-satellite payload adapter, like for example in the Vega SSMS. The reference launch vehicle, SpaceX Falcon 9 provides a direct price based on the payload mass, so, unless farther adapters are attached to its ports, in this case this coefficient has been set to 0. The initial mobile mass is the sum of the propellant mass and the final mass:

$$m_i = m_f + m_{prop} \quad (49)$$

The final mass is the sum of the payload (i.e. from our point of view the satellite itself) and the inert mass of the sat carrier or the satellite propulsion system:

$$m_f = m_{pay} + m_{inert\_tot}$$

This inert mass has a component proportional to the propellant mass and one proportional to the payload (i.e. fixed with respect of ΔV):

$$m_{inert\_tot} = m_{inert\_fix} + m_{inert\_pr} \quad (51)$$

$$m_{inert\_fix} = k_2 m_{pay} \quad (52)$$

$$m_{inert\_pr} = k_3 m_{prop} \quad (53)$$

The propellant mass is calculated rearranging the usual Tsiolkovsky equation. The inert mass coefficients have been selected to simulate MOOG satellite carriers [31–33].

The total cost of the mission is divided in three elements, namely the launch costs, the sat carrier/propulsion costs and all the other mission costs (the satellite, the ground segment, project management and so on …):

$$c_{tot} = c_{lv} + c_{pr} + c_m$$

The launch costs have been fixed equal to 5500 $/kg as from SpaceX advertising. The mission costs are related to the payload mass by two coefficients: one is the specific cost and the other is a penalty coefficient that take into account the time lost during deployment. For example, a one year deployment due to a RAAN change can correspond to 1/3 to 1/5 of a smallsat lifetime. However, depending on the type of platform/mission, the satellite could be partially operational (e.g. taking images) even before reaching its final position. For simplicity this coefficient has been set to 1. To perform a sensitivity analysis, the total mission costs excluded access to orbit costs have been set to three values: Low (L) 25 k$/kg, Medium (M) 50 k$/kg and High (H) 100 k$/kg. The order of magnitude is in line with the Aerospace Corporation Small Satellite Cost Model (SSCM) [79] and data from the news/literature.

$$c_{tot} = k_4 m_{lv} + c_{pr} + k_8 k_9 m_{pay} \quad (55)$$

$$c_{pr} = k_5 m_{inert\_fix} + k_6 m_{inert\_pr} + k_7 m_{prop} \quad (56)$$

The satellite carrier costs are proportional to its inert mass. The specific cost has been set to 25 k$/kg. the propellant cost is set to 100 $/kg. All data are listed in Table 4.

Fig. 43 shows the mass evolution as a function of the in-orbit ΔV to reach the final position. As the ΔV is increased the propulsion mass increases exponentially.

This increase in total mass directly influences the launch costs (Fig. 44). The propellant mass has only this indirect effect because its direct cost is negligible. On the opposite, the inert mass, while having roughly 1/3 of the indirect effect, has also a direct contribution due to its significant specific cost. The mission costs other than launch and propulsive costs tend to be always dominant except for very large ΔVs.

The total cost in case of a dedicated launch with a corresponding small launch vehicle can be calculated by the same equations, but in this case the ΔV and the propulsion mass (related to the journey to orbit) is set to zero. Again, to perform a sensitivity analysis, in this case the specific launch costs have been set to three values: Low (L) 12.5 k$/kg, Medium (M) 25 k$/kg and High (H) 50 k$/kg. The order of magnitude is in line with the declarations from small launcher providers. The High

Table 4
Value of parameters used in the cost model.

Parameter	Value
k_1	0
k_2	0.2
k_3	0.333
k_4	5.5 k$/kg
k_5	25 k$/kg
k_6	25 k$/kg
k_7	0.1 k$/kg
k_8	25 or 50 or 100 k$/kg
k_9	1

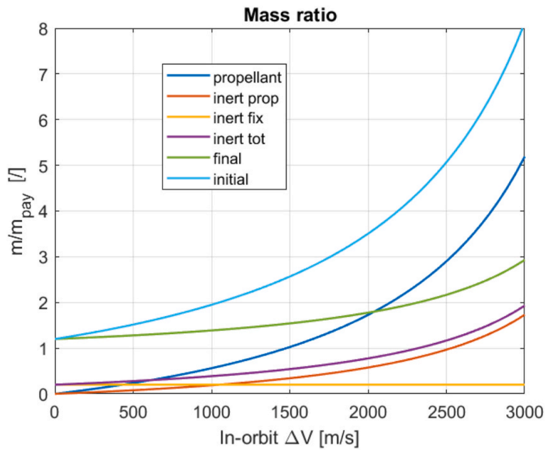

Fig. 43. Mass ratio as a function of the in-orbit ΔV.

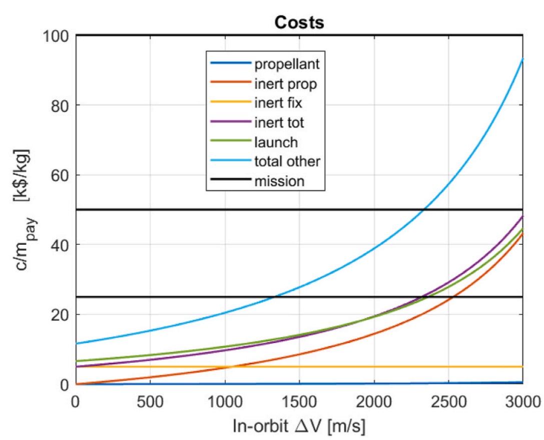

Fig. 44. Specific costs as a function of the in-orbit ΔV.

value is more probable for micro-launchers with a payload on the order of 10^2 kg while the Low value could be achieved by small launchers with payload around 10^3 kg.

It is important to note that the cost per kg is a useful metric to make comparisons, however, particularly in case of a dedicated launch, the total launch costs tend to be fixed and the cost per kg depends on the filling level of the launcher capacity. Anyway, this can still be taken into account in the current model by dividing the nominal cost per kg by the launcher payload capacity utilization. For example, a launch vehicle of the Medium category filled at 50 % will fall into the High category. Moreover, for the same reason, the cost per kg is also not a fixed number at full capacity because the latter changes with the final orbital parameters, particularly the altitude.

Fig. 45 shows, as expected that the rideshare option is the most affordable, but the advantage deteriorates as the final destination is far away from the release orbit. Depending on the cost level of the small launch vehicle at a certain point a dedicated launch becomes cheaper overall.

Considering the actual capabilities of small satellites and satellite carriers propulsion systems the right part of the graph is generally unfeasible. In fact, currently, rideshare options do not move far away from the original orbit. As seen in chapter 2, in-plane manoeuvres do not require more than 500 m/s of ΔV in LEO. Higher ΔV are related with changes in the orbital plane. Consequently, the rideshare option is bounded to the nominal orbital plane, and different ones can be reached only with a dedicated launch. For this reason, it is interesting to zoom in the region up to 500 m/s as shown in Fig. 46.

Considering a micro-launcher, the red lines represent the most

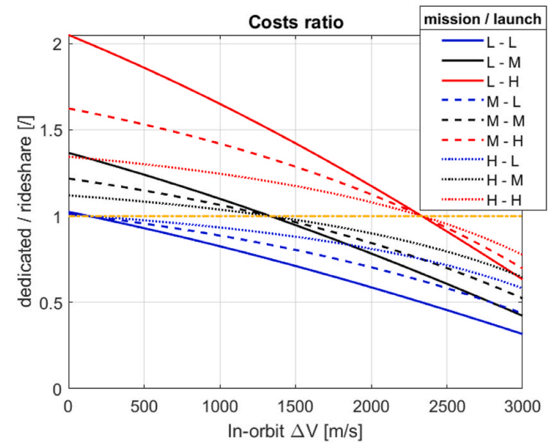

Fig. 45. Ratio between total costs of a dedicated launch vs rideshare. $k_2 = 0.2$.

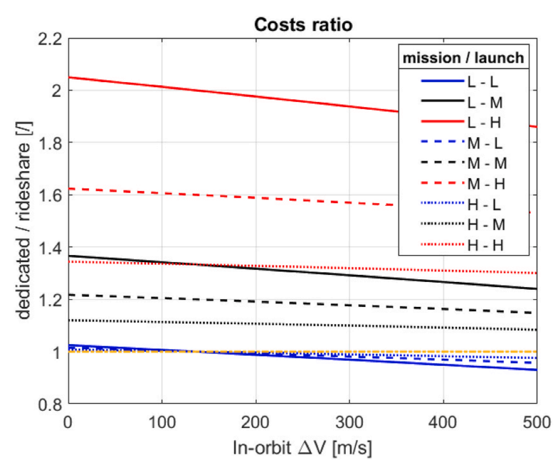

Fig. 46. Ratio between total costs of a dedicated launch vs rideshare (zoom in). $k_2 = 0.2$.

probable situation, with the black lines as an optimistic projection. Even if the micro-launcher cost is 10 times the one of the F9, the total mission costs are roughly between 40 % and two times higher because of the flattening effect of mission costs, and this penalty could be considered acceptable when other aspects are paramount. The advantage of the rideshare is reduced as the mission cost are increased. Consequently, the market for micro-launcher is clear, not an untenable price competition against rideshare options but a dedicated premium service for relatively sophisticated missions that require a responsive access to space, particularly for underserved orbits.

Typical but not exclusive customers for such a service are the governments. With the rise of small satellites, a lot of countries now have emerging space capabilities but cannot afford a (traditional) independent access to space (e.g. Brazil, Australia, Turkey, Spain and many more) unless it is a small launch vehicle. But also traditional players like the US, France, China, India etc. are tremendously interested in the possibility of deploying tactical satellites on demand.

Consequently, any development of a micro-launcher should necessarily focus on the capability to launch "anytime, anywhere", preferably on short notice. In this regard, air launch is an interesting option, but take off location and available flight time should be carefully considered, while on the opposite a ground launch exclusively from spaceports like Andoya (Norway), Kiruna (Sweden) and Sutherland (Scotland, UK) is very limiting in terms of inclination achievable.

Shifting the interest to mini-launchers with payloads on the order of 10^3 kg, the situation becomes more favourable, with the results portrayed by the black (pessimistic) and blue (optimistic) lines as their specific launch costs are lower than those of the previous micro-launchers. Particularly at the low end of price, the small launcher becomes very competitive. However, for such level of payload it is generally not practically possible to have a fully dedicated launch for small satellites, so a certain number of payloads have to be launched together. Anyway, the need to aggregate almost ten times less mass in a single launch guarantee a better chance to find multiple payloads with similar orbits. As in this case the launch is not fully dedicated, in order to make the comparison more fair and not too optimistic Fig. 46 has been modified eliminating the penalty mass for the large launch vehicle and replotted as Fig. 47.

With this hypothesis, at the low end of launch price, the small launch vehicle can provide a total mission cost less than 25 % more than the larger vehicle. This penalty can be more than acceptable if it can solve the bottleneck in launch opportunities. Therefore, the development of a small launcher should focus on a good balance between costs and responsiveness, with less extreme requirements compared to a micro-launcher in terms of lead times and flexibility but lower prices, which should be also easier to attain at this scale.

In any case, also the small launch vehicle should operate on one or more locations that altogether allow to reach multiple inclinations. In fact, currently many SmallSat LEO missions are often launched in SSO but this is only partially due to the inherent benefits of this kind of orbits, as the rideshare options are also targeted to those orbits limiting availability for the others in a sort of chicken and egg problem. Different inclinations can have important advantages too, like more payload to orbit and more frequent passages at low inclinations, also considering that the majority of population in every continent excluding Europe lives between±45° of Latitude. Another possibility is to use 63.4° of inclination on an elliptical VLEO for Earth observation in order to fix the perigee (minimum altitude, max resolution) at the latitude of interest and increase the apogee altitude to decrease the average drag.

Launch frequency should be relatively high, at least one per month if not one per week. Increasing launch frequency has also a double beneficial effect of reducing launch prices and increasing the number of opportunities and consequently catching a larger share of the market.

6. Conclusions

The small satellite market is booming right now. There are three possibilities to put a satellite in its foreseen orbital position: by a dedicated launch with a small lunch vehicle, by the satellite own propulsion after getting in orbit with a rideshare option or by a satellite carrier performing the same function.

The paper showed that the sat carrier is inherently more inefficient than autonomous propulsion, particularly as the ΔV and the number of satellites increase. However, it could still be the preferable solution for micro-satellites with limited propulsion capabilities when it is too complex, long and expensive to design the satellite platform to cope with all the uncertainties in parking orbits provided by different possible launch opportunities.

Small launch vehicles are inherently more expensive than rideshare options, but their costs impact only a part of total mission costs and can provide a higher responsiveness and frequency of launch opportunities, particularly for unserved but still interesting orbits.

CRediT authorship contribution statement

Francesco Barato: Conceptualization, Data curation, Formal analysis, Investigation, Methodology, Software, Validation, Visualization, Writing – original draft, Writing – review & editing. **Elena Toson:** Data curation. **Fabiana Milza:** Data curation. **Daniele Pavarin:** Supervision.

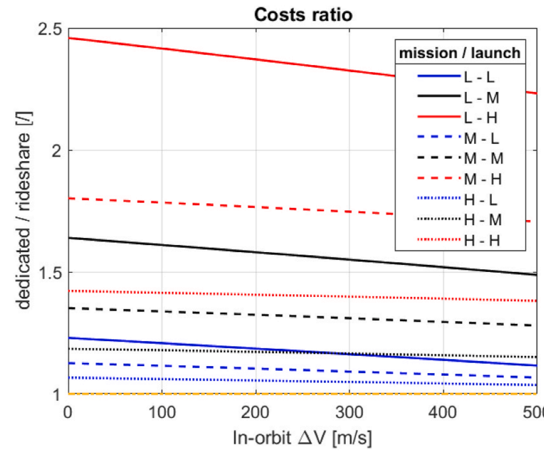

Fig. 47. Ratio between total costs of a dedicated launch vs rideshare (zoom in). $k_2 = 0$.

Declaration of competing interest

The authors declare that they have no known competing financial interests or personal relationships that could have appeared to influence the work reported in this paper.

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