CAS-ESA Call for small mission proposals - Technical annex

Copenhagen workshop

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European Space Agency

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1. General considerations: Guidelines and Schedule

General guidelines:

- Spacecraft wet mass ≤ 300 kg (full space segment mass excluding launch adapter and any eventual propulsion module for orbital transfer)
- Payload mass ≤ 60 kg
- Payload average power consumption < 65 W (< 100 W peak)

Implementation schedule:

- Mission selection in 2015
- Joint definition phase < 2 years
- Space segment development < 4 years
- Launch in 2021
- Operational lifetime: 2-3 years

1. General considerations: Work breakdown and share

A Joint Mission is targeted with clear technical interfaces

- Must be jointly achieved:

Overall mission management
Science management and exploitation
Science payload

- To be provided by China, Europe, or jointly:

Mission architect	Spacecraft operations (MOC)
Platform	Ground stations
System Integration and Testing	Science operations (SOC)
Launch services	Science exploitation

1. General considerations: Technology Readiness and other constraints

Technology readiness:

- S/C development schedule ~3.5 years => re-use of available technologies (ISO TRL ≥ 6 for the payload and ≥ 7 for the platform elements).
- Payload must rely on heritage (but new development possible).
- Be careful with potential obsoleteness of components when referring to heritage.

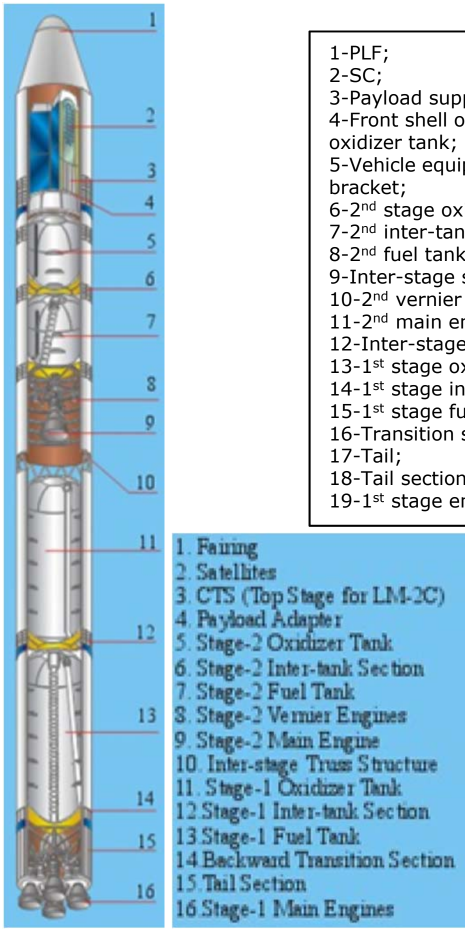
Export control constraints:

Compatibility with a launch from China is required. Therefore, the S/C must comply with any applicable export control regulation (e.g. US ITAR free)

2. Potential LVs and example of mission profiles

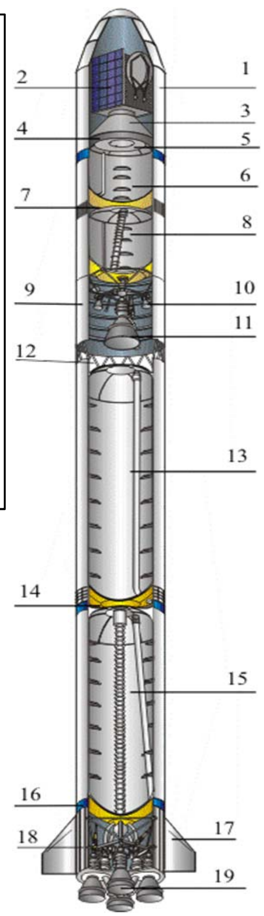
1. Long March LM-2C or LM-2D, launched from China.
2. European launchers (Vega/Soyuz/Ariane 5), launched from Kourou.
3. Possibility of additional upper stages (mostly for Earth escape orbits):
   a. LM-2C/CTS.
   b. LM-2D/TY-2.
   c. Liquid propulsion module with Vega (based on LPF example).
4. Only auxiliary/piggy-back passenger launches for Soyuz and Ariane 5. Note that an auxiliary passenger launch adds constraints on the potential orbits that can be reached (main passenger dictates the launcher ascent profile and burns) => highly specific and specialised orbits are unlikely, as opposed to more common orbits (e.g. SSO or GTO).
5. Detailed launcher performance curves are provided in Appendix (and full details in the User Manuals).

2. Potential LVs and example of mission profiles

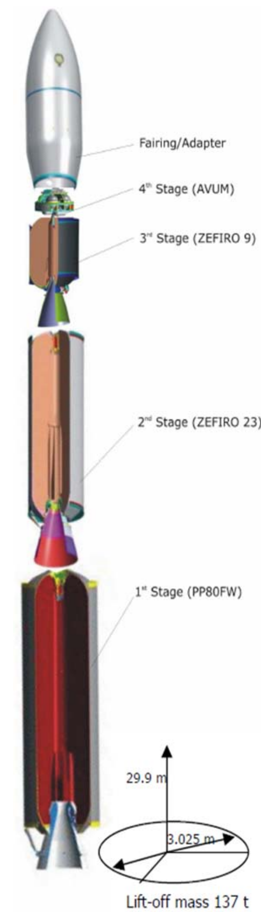


LM-2C/CTS

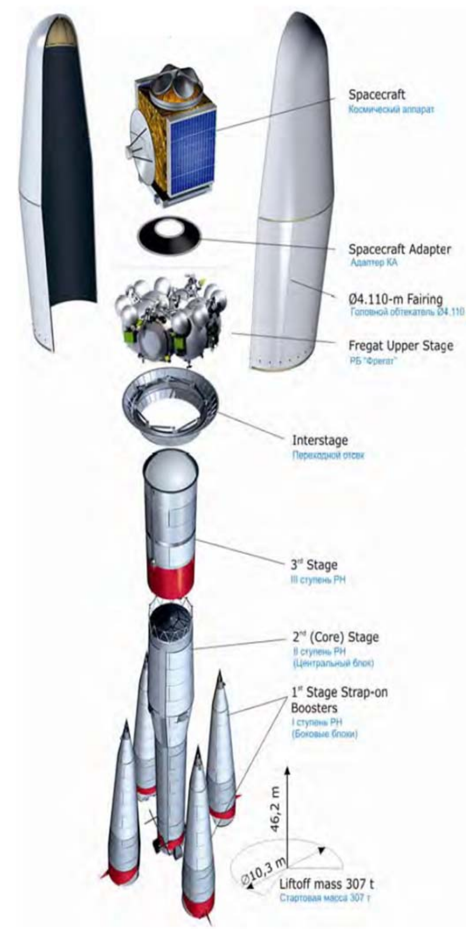
1-PLF;
2-SC;
3-Payload support bay;
4-Front shell of 2nd stage oxidizer tank;
5-Vehicle equipment disk-like bracket;
6-2nd stage oxidizer tank;
7-2nd inter-tank section;
8-2nd fuel tank;
9-Inter-stage section;
10-2nd vernier engine;
11-2nd main engine;
12-Inter-stage strut structure;
13-1st stage oxidizer tank;
14-1st stage inter-tank section;
15-1st stage fuel tank;
16-Transition section;
17-Tail;
18-Tail section;
19-1st stage engine.



LM-2D



Vega



Soyuz

2. Potential LVs and example of mission profiles

1. A preliminary assessment was made of potential mission profiles for all different launchers under consideration. Their performance to the typical science orbits are given in the next slide.
2. Some of the performances indicated rely on the launchers themselves, others imply the use of an additional upper stage or a propulsion module (this is indicated in "[…]").
3. Note that in most cases, the launchers' performances exceed the mass guidelines given earlier related to the programmatic constraint.
4. Therefore, two constraints have to be kept in mind:
   a. $M_{S/C\_wet} + M_{PM\_wet} + M_{adapter} \leq M_{launch}$
   b. $M_{S/C\_wet} \leq 300$ kg
5. Note the specific case of a mission to Venus, where a mass indication is given before/after insertion into orbit around Venus. In this case, the propulsion subsystem of the S/C has to take care of the orbital insertion.

2. Potential LVs and example of mission profiles

	Vega [with bi-liquid propulsion module]	Soyuz	LM-2C [LM-2C/CTS]	LM-2D [LM-2D/TY-2]
LEO	~ 2.300 kg @ 300 km (i=5°) 1.480 kg @ 400 km SSO 1.140 kg @ 1000 km SSO	4500 kg @ 700 km SSO	1850 kg @ 400 km SSO [1650 kg @ 700 km SSO with LM-2C/CTS]	2200 kg @ 400 km SSO [1550 kg @ 700 km SSO with LM-2D/TY-2]
HEO	1.963 kg @ 200 x 1500 km [~ 650 kg @ 300 x 36000 km] (both at equatorial i=5.4°)	3250 kg in GTO	3350 kg @ 200x1000 km (i=29°) [1250 kg in GTO with LM-2C/CTS]	3700 kg @ 200x1000 km (i=28.5°)
Sun Earth L1/L2 (C3 = 0 km^2/s^2)	[~ 420 kg]	2160 kg	[820 kg with LM-2C/CTS]	[380 kg with LM-2D/TY-2]
[1]Heading/trailing heliocentric orbits and [2]Sun-Earth L4/L5 (C3 > 0 km^2/s^2)	[1][≤ 400 kg] [2][~ 230-350 kg for L5 depending on transfer time]	< 2160 kg	[< 820 kg with LM-2C/CTS]	[< 380 kg with LM-2D/TY-2]
Venus, before orbit insertion (C3 ≈ 7.5 km^2/s^2)	[≤ 340 kg]	~1780 kg	[< 420 kg with LM-2C/CTS]	[< 200 kg with LM-2D/TY-2]
Venus, after insertion into 2-day HEO	[≤ 240kg after insertion]	~1250 kg	[< 290 kg with LM-2C/CTS]	[< 140 kg with LM-2D/TY-2]
Earth escape / interplanetary transfers	See performance as a function of C3 in Appendix.			

2. Potential LVs and example of mission profiles

1. Typical transfer durations, data rates and power generation capabilities are indicated below for the different orbits analysed:

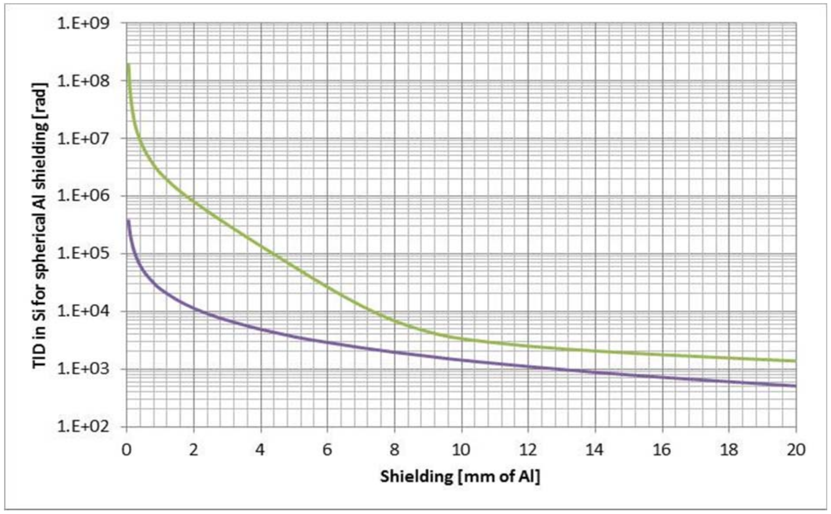
Orbit	Typical transfer duration	Typical science TM data rates	Power
LEO	< 1 day	X band: 20 – 200 Mbps S band: ~ 600 kbps	@ 1 AU Solar radiation: ~1300 W/m^2 Cosine loss for 36° off-pointing: 80% Cell efficiency: 28% System losses: 85% Cell packaging ratio: 70% Ageing: 86% (@ 3.75%/year for 4 years) ~150 W/m^2 at EoL
HEO			
Sun Earth L1/L2	~ 1 month	X band: 5-10 Mbps Ka band: 75 Mbps	
Heading/trailing heliocentric orbits and Sun-Earth L4/L5	14 – 50 months (in increments of 1 year, see details in Appendix)	Ka band: 150 kbps	
Venus	100 – 180 days (conj. transfer) 350 – 450 days (1.5 revolution transfer)	X band: 63 – 228 kbps (superior vs. inferior conjunction)	Approximately 1.9 times the value at Earth Higher temperatures may further reduce the solar cell efficiency.

3. Radiation environment

1. Models of the space environment and its effects can be found at https://www.spenvis.oma.be/.
2. The energetic particle radiation environment consists of trapped charged particles in the Van Allen radiation belts (electrons and ions trapped by the Earth magnetic field), Solar particles (mainly protons) and Galactic Cosmic Rays. The impact (performance and shielding requirement) is mission/orbit dependent.
3. As an example, Total Ionising Dose for missions operating in 2 different orbits:

- **The HEO orbit (green curve) is 1.8 x 7 R_{Earth}^2, argument of perigee 270 degree, inclination 63.4 degrees, 3 year lifetime => crosses both radiation belts 4 times every day.**

- **The orbit around L2 (purple curve) has a high amplitude of about 1.5 Mkm, is attained with a direct transfer strategy from launch, and also has a 3 year lifetime.**



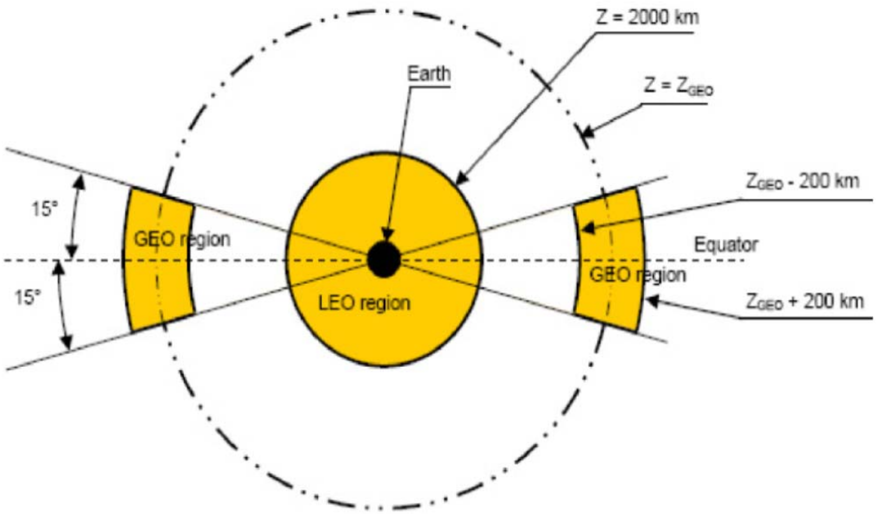
4. Data rate aspects

1. For small platforms in LEO/HEO in X band: ~ 20 to 200 Mbit/s, < 10 cm Low Gain Antenna, ≤ 10 W, 3 - 15 m ground antenna.
2. L2 orbit in X band: 10 Mbps, 30 cm High Gain Antenna, 30 W, 35 m ground antenna.
3. L2 orbit in Ka band: 75 Mbps, 50 cm HGA, 35 W, 35 m ground antenna.
4. Planetary mission at 1 AU in Ka band: 150 kbps, 1.1 m HGA, 35 W, 35 m ground antenna.
5. Planetary mission at Venus in X-band: 63 – 228 kbps (superior – inferior conjunction), 1.3 m HGA, 65 W, 35 m ground antenna.

Note: Existing small platforms will require large modifications to accommodate the communication subsystems detailed above for orbits beyond LEO.

5. Space debris mitigation and propulsion subsystem

1. Space debris mitigation [4] in LEO implies the need for a propulsion subsystem, to:
    - Move the S/C into a graveyard orbit at EoL, or
    - Ensure (passive) re-entry in the atmosphere within 25 years
2. This applies to the S/C + LV upper stage + S/C adapter(s) + ejectable covers etc. Further details (ΔV) are provided in Appendix.
3. Other ΔVs to consider:
    - Orbit transfer and/or insertion
    - Launcher dispersion correction manoeuvres
    - Orbit maintenance and specific manoeuvres



LEO and GEO protected regions [4]

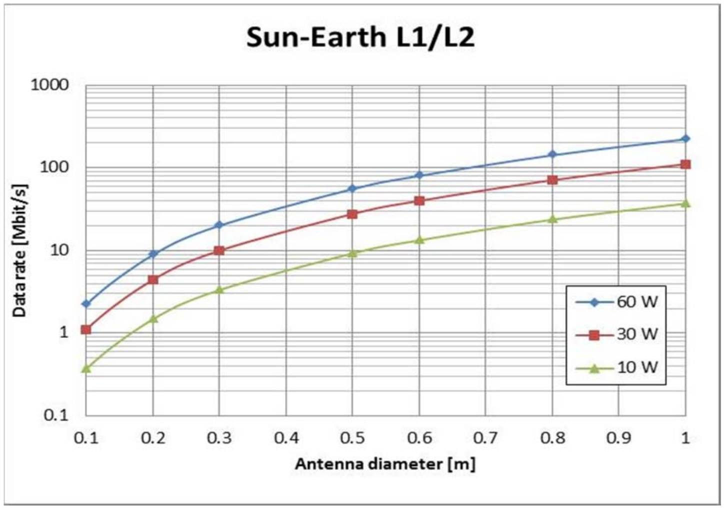
Appendices

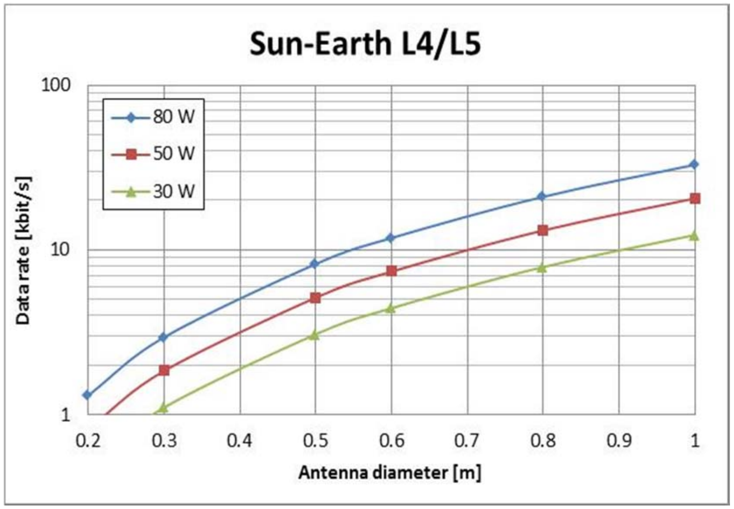
Data rate sensitivity analysis

$$\text{Data rate} \propto P_t.(D_t/\lambda)^2.\ (D_r/\lambda)^2.\ (\lambda/r)^2$$

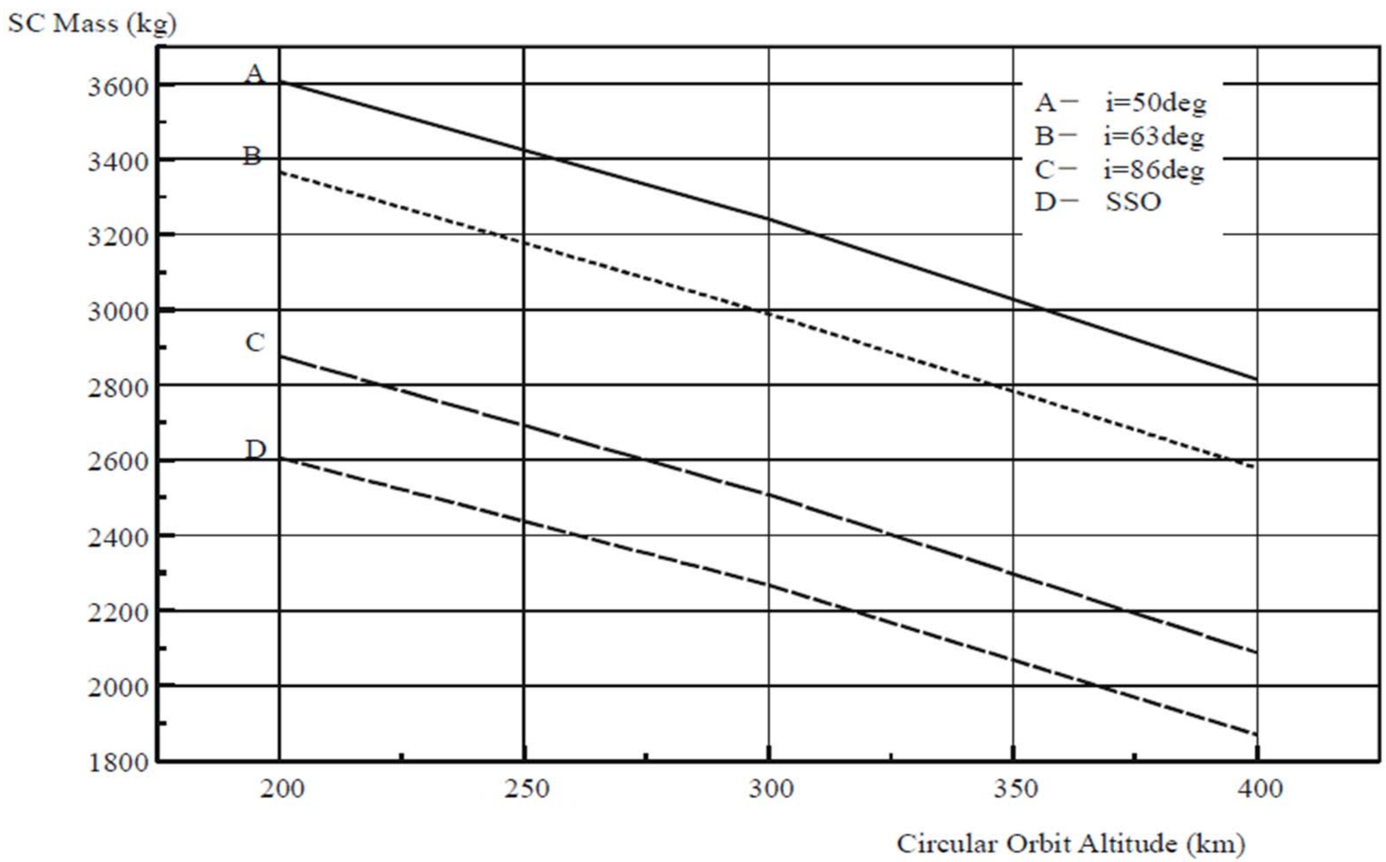
P_t is the communication subsystem transmitter power
D_t (resp. Dr) is the diameter of the transmitting (resp. receiving) antenna
λ is the communication wavelength
r is the distance between the spacecraft and the ground station

- Example of scaling the data rate as a function of antenna diameter, transmitter power and S/C to ground station distance based on the L2 orbit in X band example (slide 13 - 10 Mbps, 30 cm HGA, 30 W, 35 m ESTRACK):



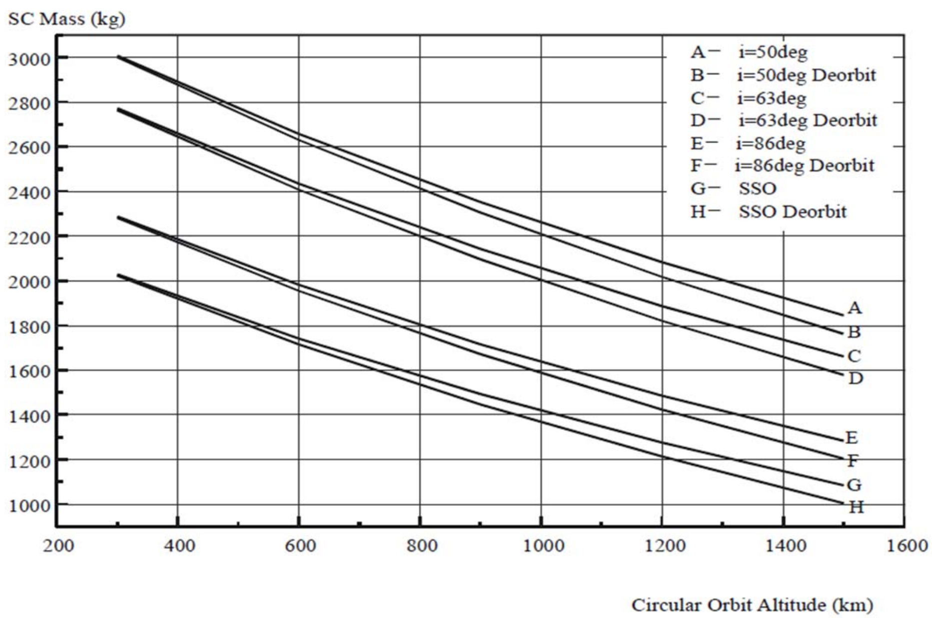


LM-2C to circular LEO



From JSLC

LM-2C with CTS upper stage to circular LEO



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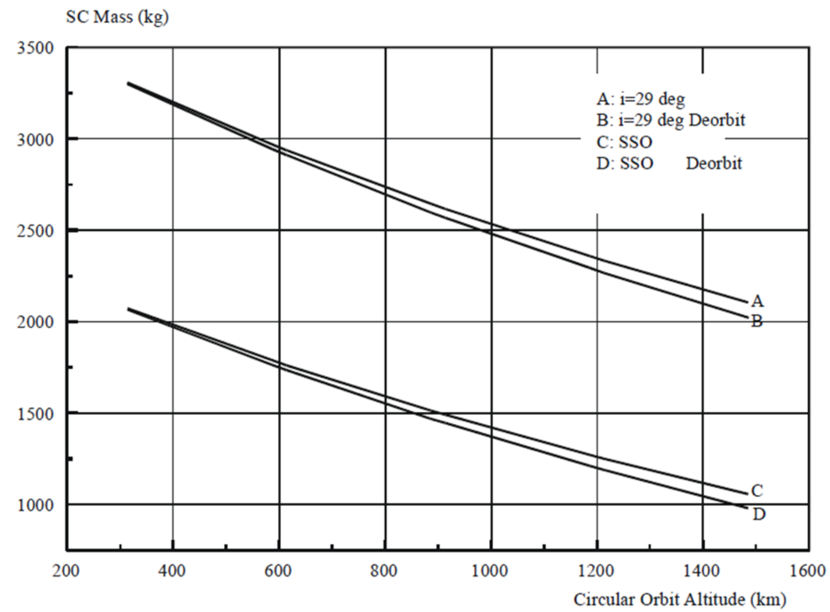
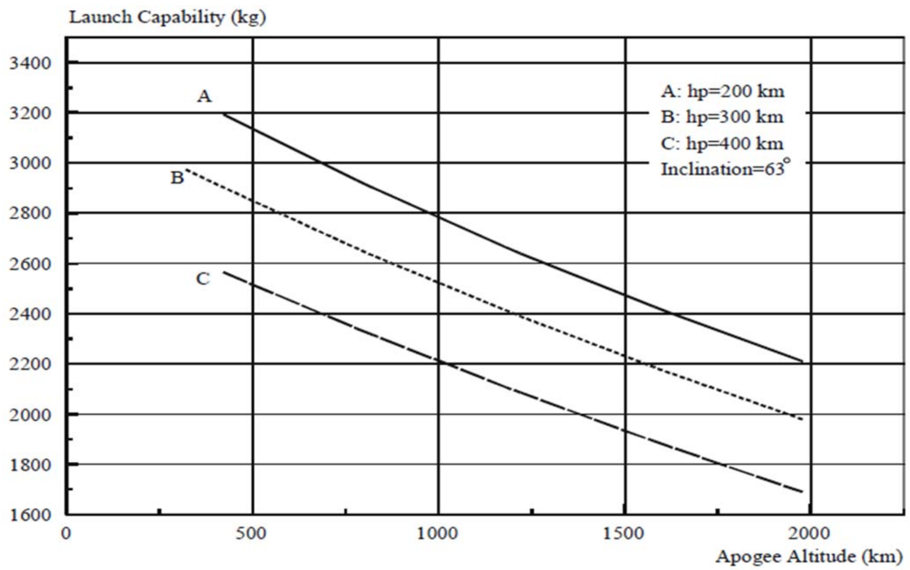


From XSLC

LM-2C to elliptical LEO



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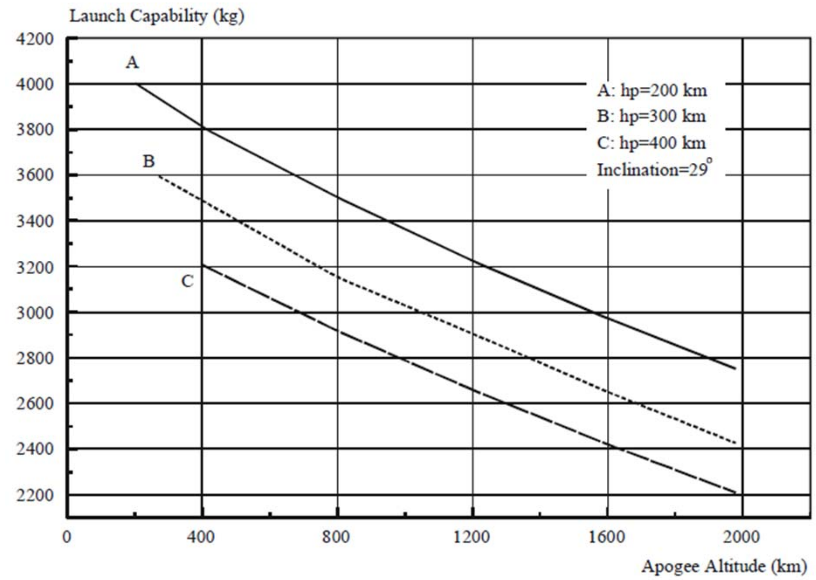
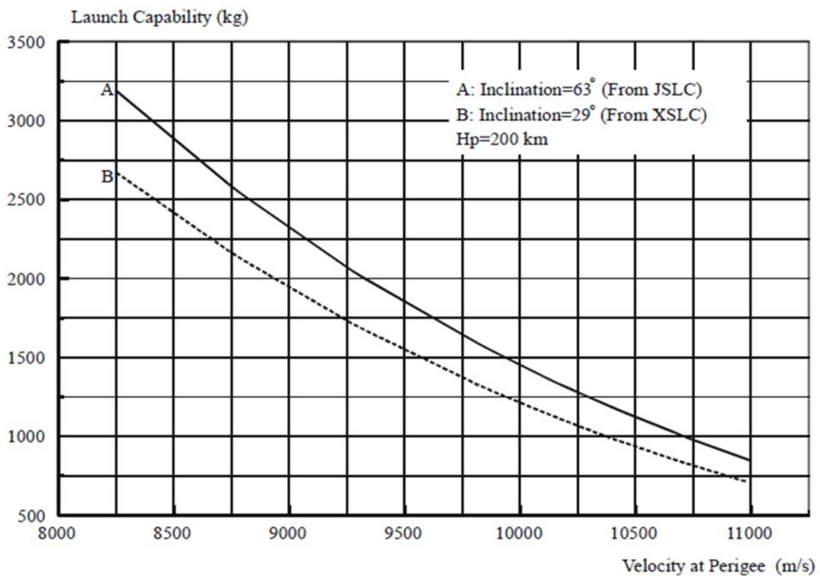
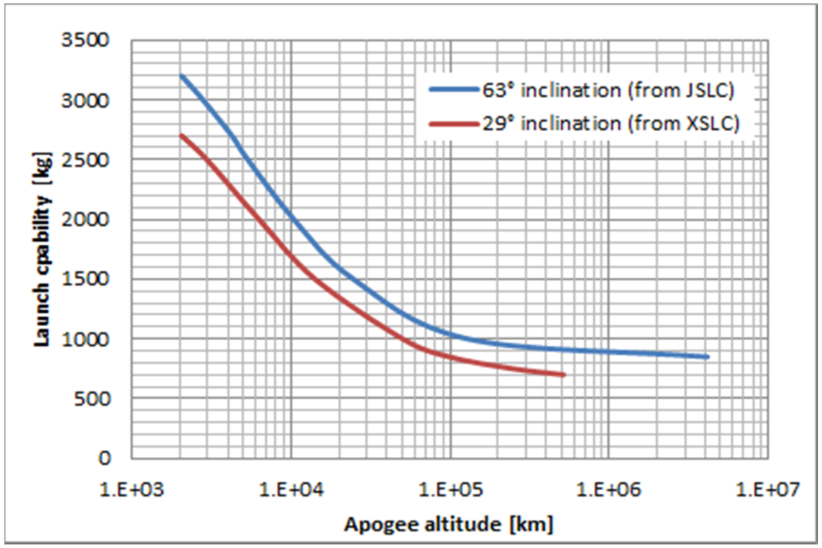


From XSLC

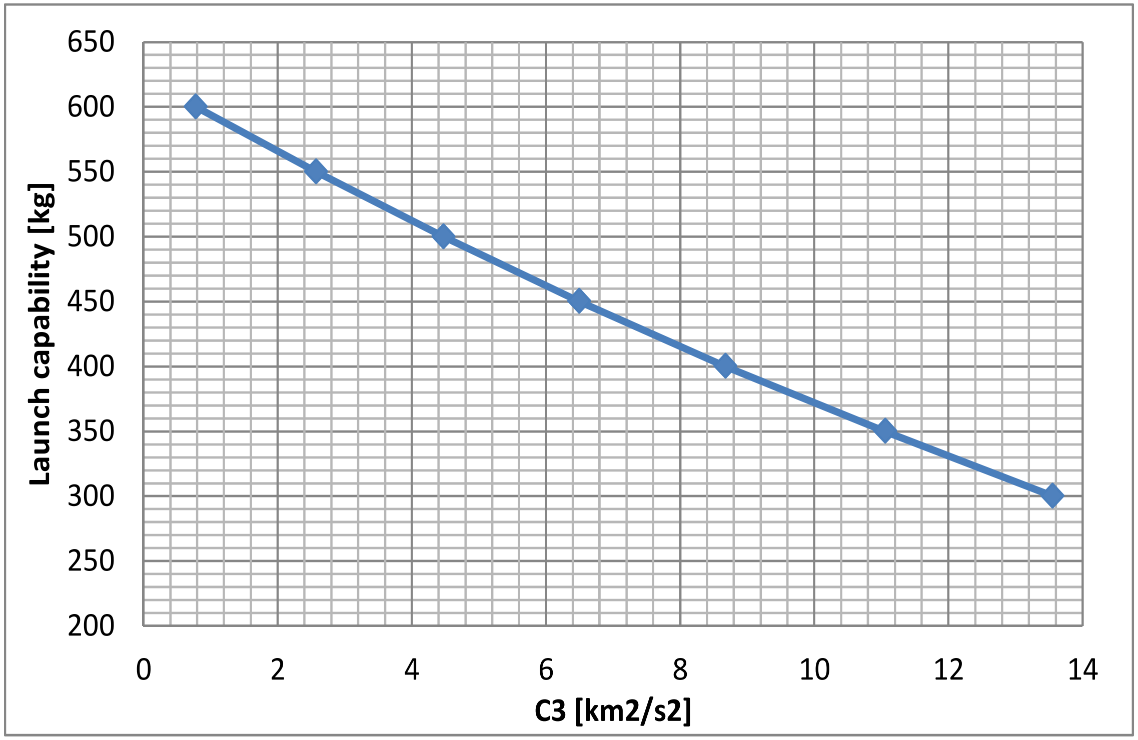
LM-2C to large elliptical orbits





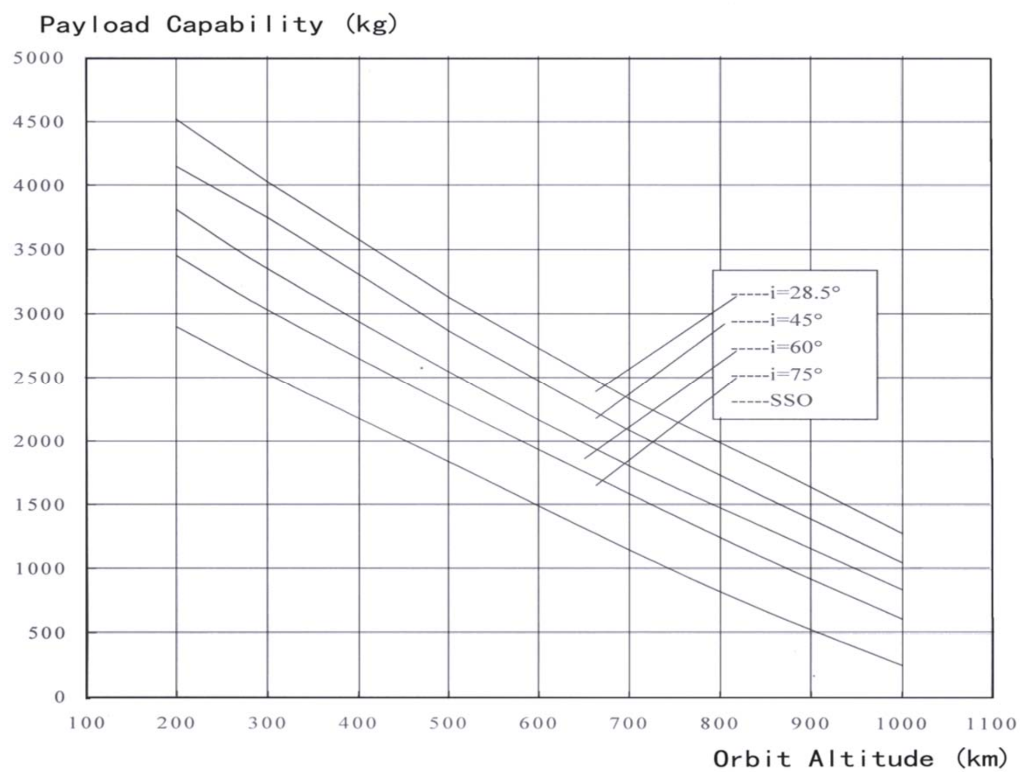
Note: Right figure gives the apogee that results from the velocity at perigee in the left figure, assuming a 200 km perigee altitude.

LM-2C with CTS upper stage to Earth escape orbits

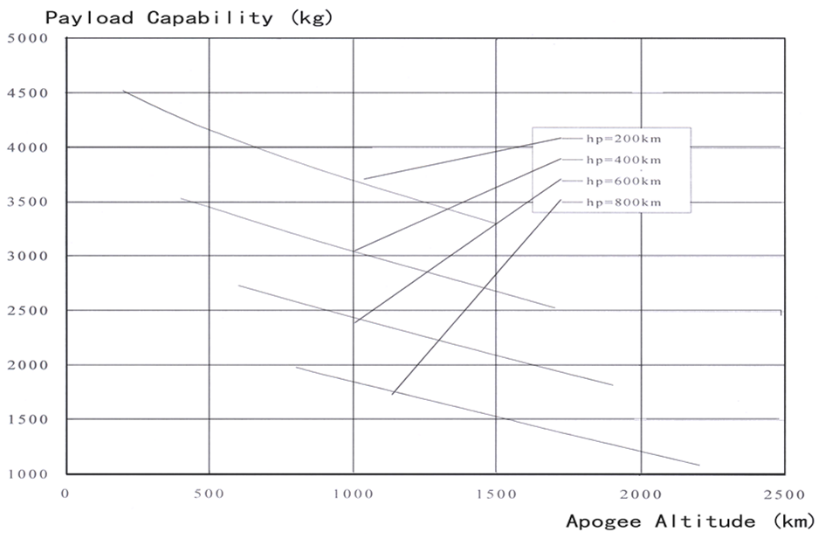


Note: C3 from a 200 km perigee, and with an inclination of 29°.

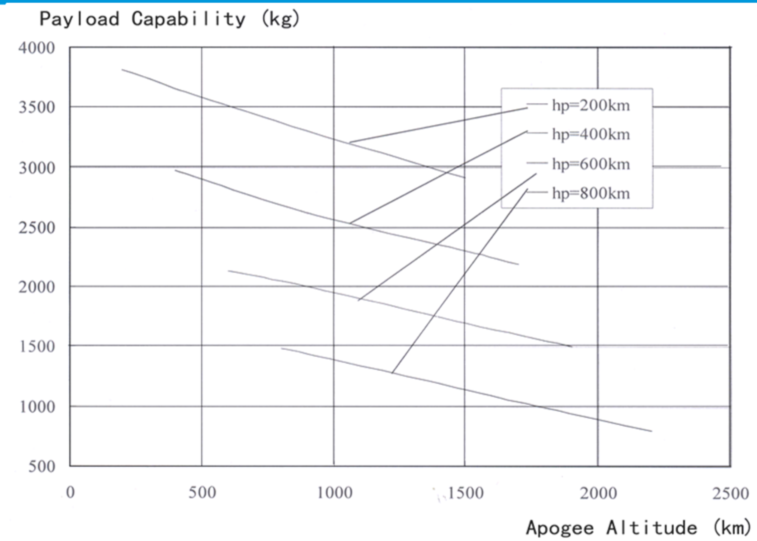
LM-2D to circular LEO



LM-2D to elliptical LEO

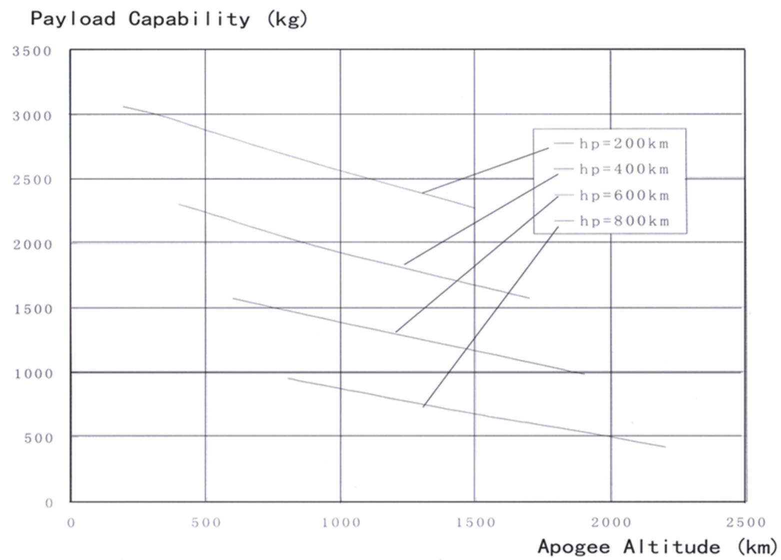


i=28.5°

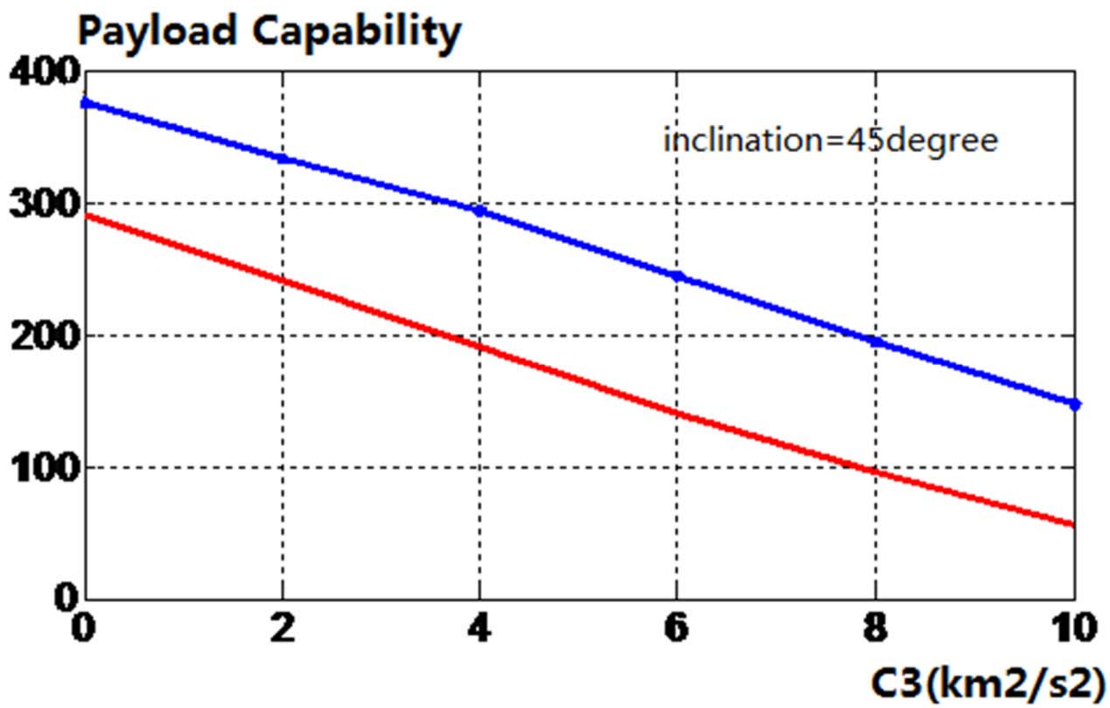


i=60°

i=90°

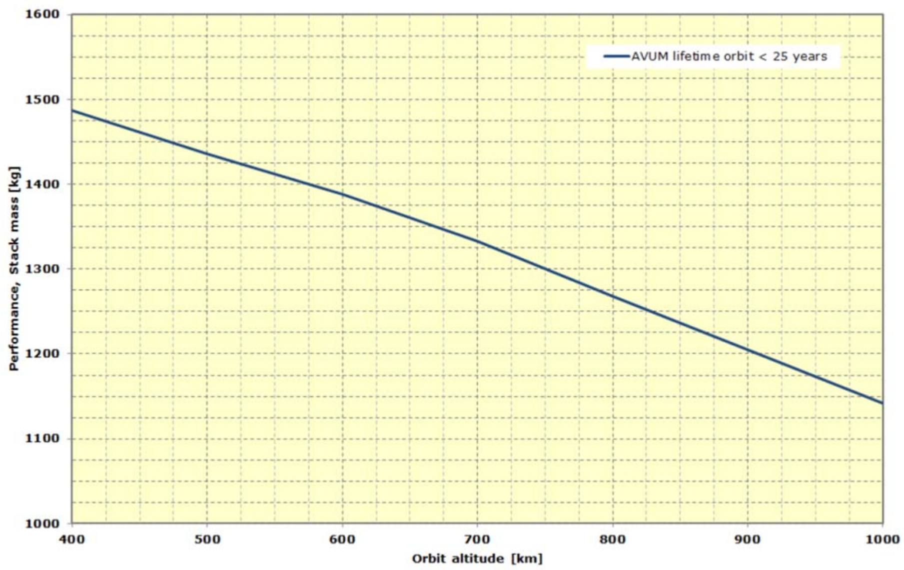


LM-2D with TY-2 upper stage to Earth escape orbits

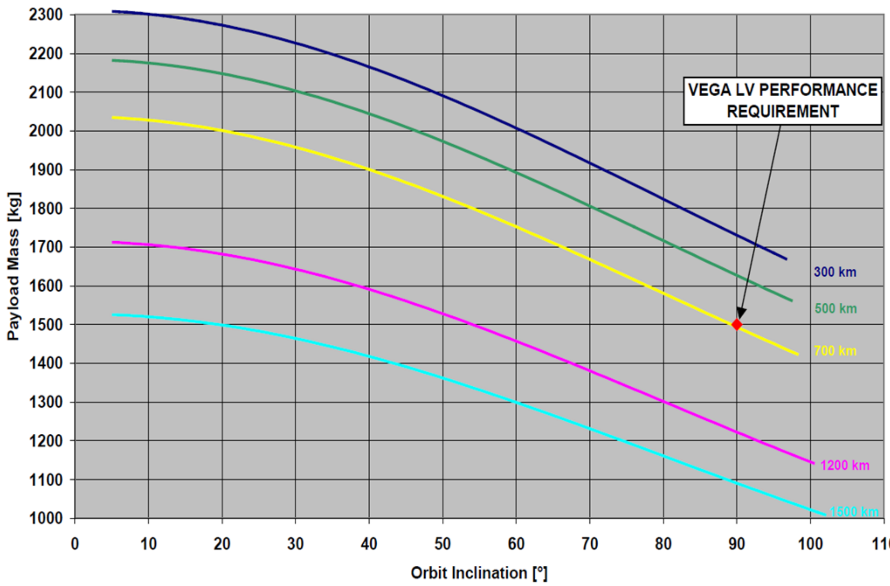


Escape from a 200 x 900 km parking orbit (blue curve) and a 200 km circular parking orbit (red curve)

Vega to LEO



At SSO inclination

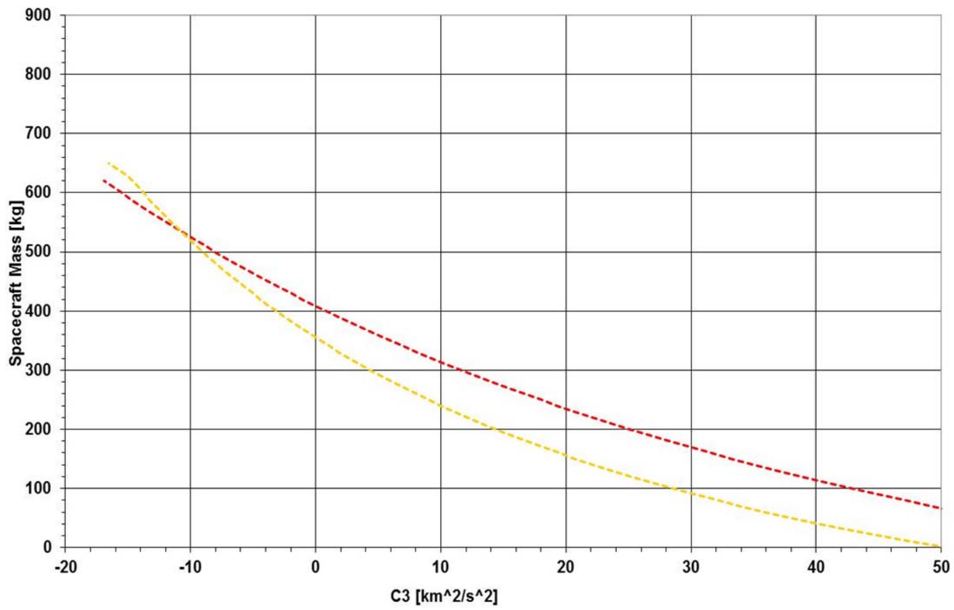


To other inclinations. Note: 5% mass margin should be included with respect to these curves.

- Performance into the low-eccentricity, near equatorial orbit planned for the upcoming ESA LISA-Pathfinder mission: 1963 kg (200 x 1500 km, i=5.4°).

Vega with a bi-liquid propulsion module

1. Possibility to add a bi-liquid propulsion module with a Vega launch, based on the LPF approach.
2. Vega insertion into a generic 300 km circular LEO.
3. Escape into any direction through multiple burns for apogee raise, followed by a final burn for eventual insertion in an Earth escape hyperbola trajectory.



- **Yellow curve with Eurostar 2000 “short” tank (from LPF). It does not take full advantage of the Vega launch performance (~2300 kg), as its propellant capacity is limited to 1200 kg.**
- **Red curve with a possible extension to a longer tank with a higher capacity.**

Vega with a bi-liquid propulsion module

1. To GEO: 650 kg (C3 ≈ -16 km^2/s^2).
2. To L1/L2: 420 kg (@ C3 ≈ 0 km^2/s^2).
3. To L4/L5: L5 is less demanding to reach than L4 + allows observations of the solar surface before the observed regions will have rotated onwards so they can affect the Earth. The transfer is performed with the propulsion module as in previous slide + an arrival manoeuvre is required by the S/C's propulsion system.

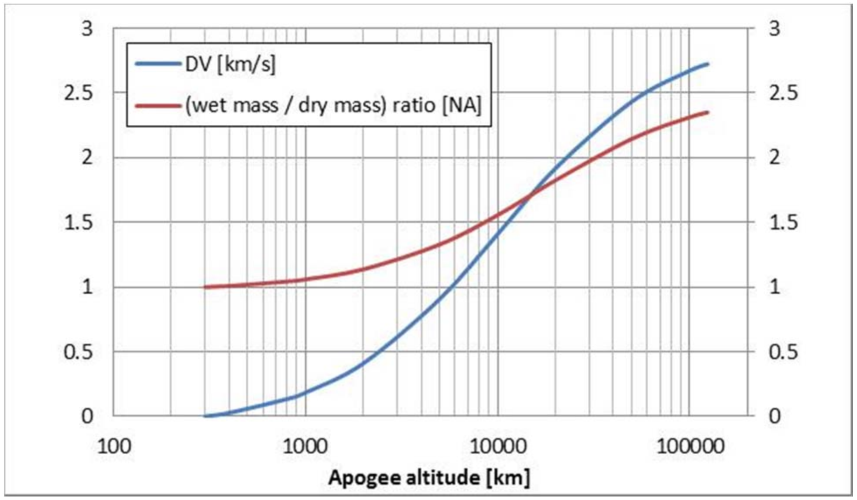
Transfer duration [months]	Escape from 300 km LEO [km/s]	Departure C3 [km^2/s^2]	Estimated spacecraft mass into heliocentric orbit incl. prop system for final insertion [kg]	Arrival manoeuvre [km/s]	Prop. fraction for arrival manoeuvre [%]
14	3.292	2.016	~ 230	1.419	37
26	3.227	0.582	~ 310	0.763	22
38	3.213	0.272	~ 335	0.521	16
50	3.207	0.157	~ 350	0.396	12

4. For drifting, Earth leading/trailing orbits, there are no constraints such as discrete transfer intervals and no arrival manoeuvre is required. The only ΔV to consider is the one required to reach Earth escape velocity, with a C3 ≥ 0 km^2/s^2.

Venus example with Vega + bi-liquid propulsion module

Venus orbital insertion into 2-day HEO orbit example (left) and Delta-V required for apogee reduction down to a 300 km circular orbit (right) with Vega + bi-liquid propulsion module:

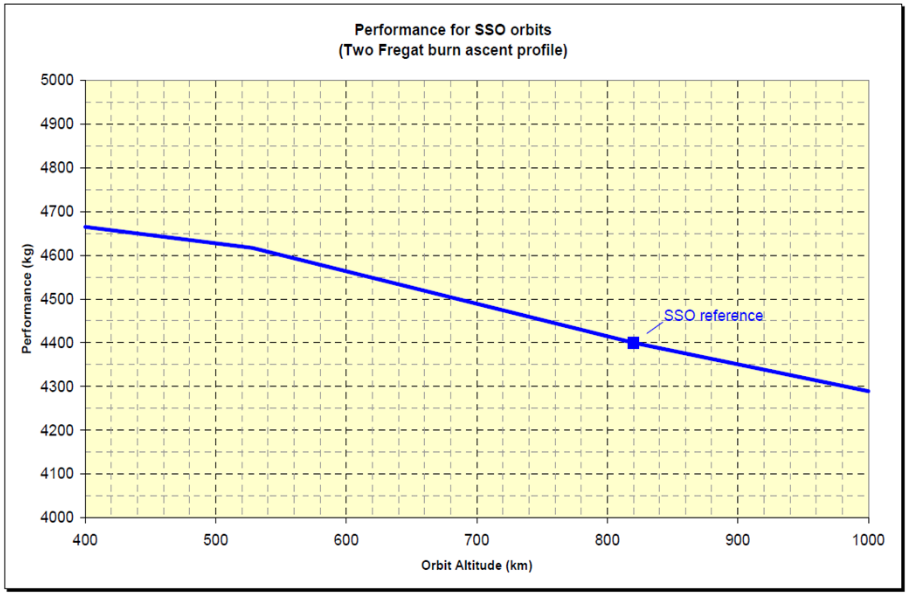
Launch date	06/11/2021	07/06/2023
Esc. Velocity [km/s]	3.608	3.127
Esc. Declination [degree]	5	-3.3
S/C wet mass [kg]	292	324
Venus arrival	24/02/2022	26/10/2023
Venus Orbit Insertion (including gravity losses) [m/s]	879	863
S/C mass in 2 day HEO [kg]	180	240



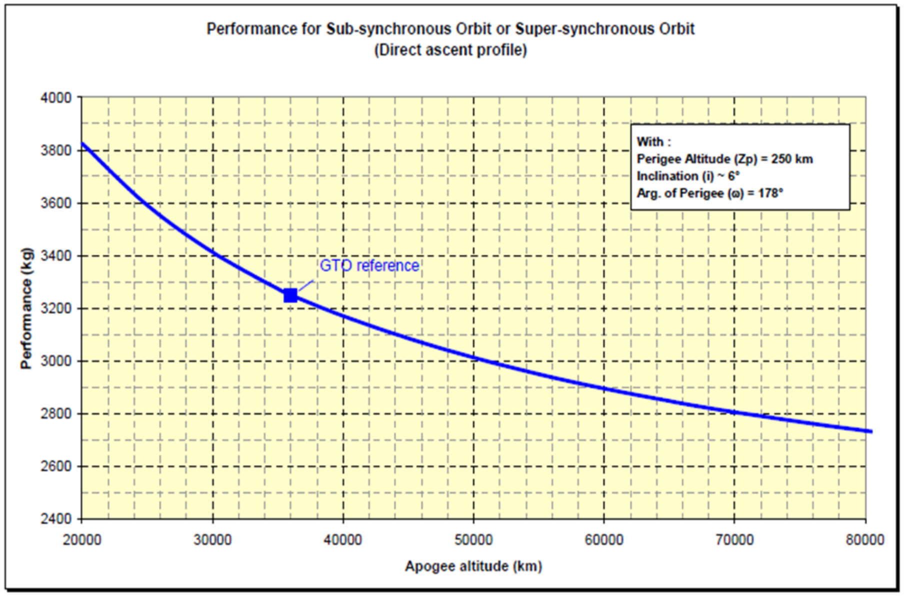
Note:

- The mass ratios (before/after Venus orbit insertion and for apogee reduction) are independent of Vega and can be re-used with the other launchers.
- Venus 2-day HEO = 300 x 123863 km. Right figure gives ΔV (and resulting wet/dry mass ratio) to reduce the apogee until circularisation at 300 km.
- Aero-braking can reduced the ΔV for orbit insertion and circularisation.

Soyuz in Earth orbits

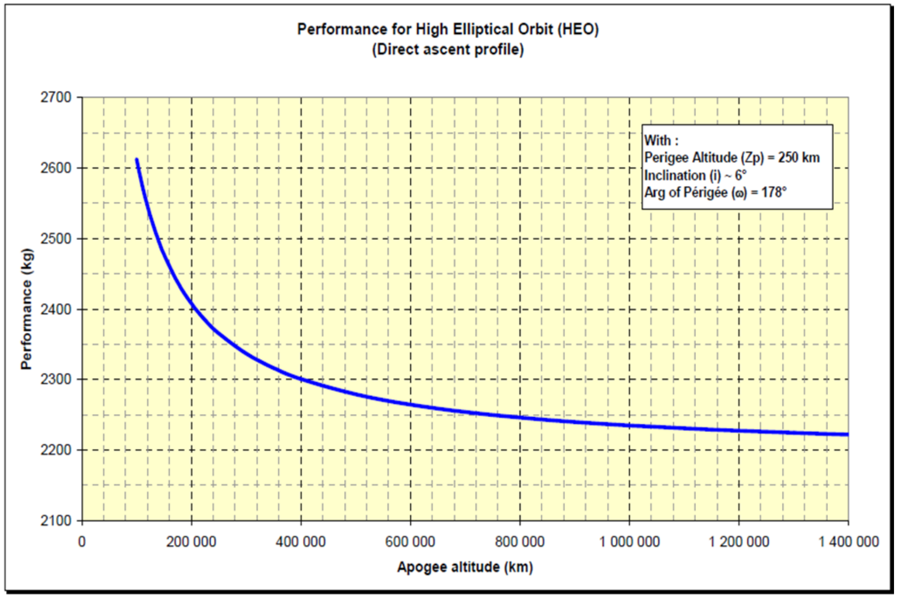


SSO orbits

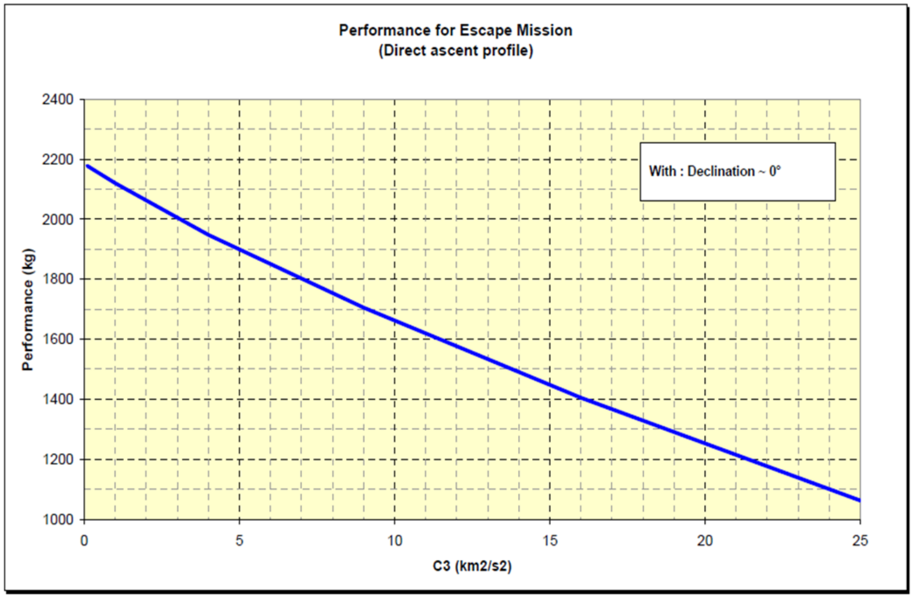


Elliptical orbits

Soyuz to HEO and Earth escape

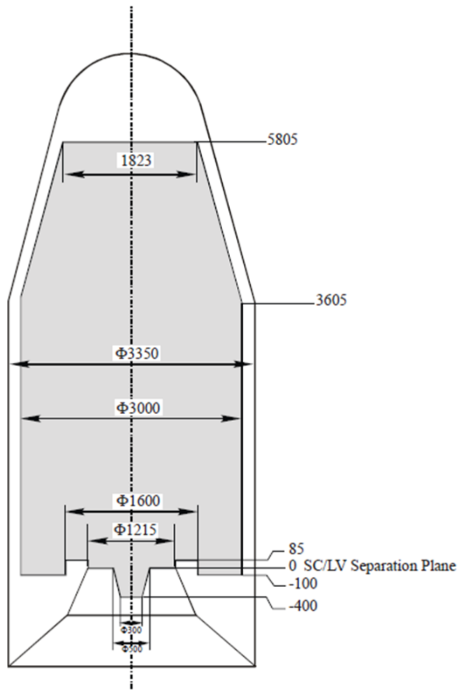


HEO orbits

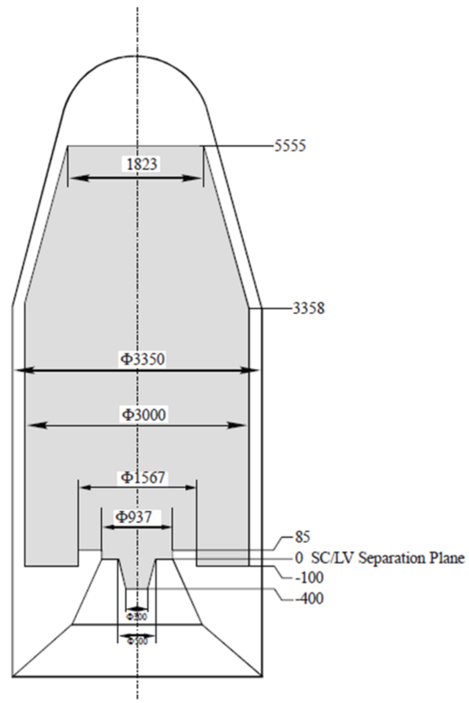


Earth escape missions

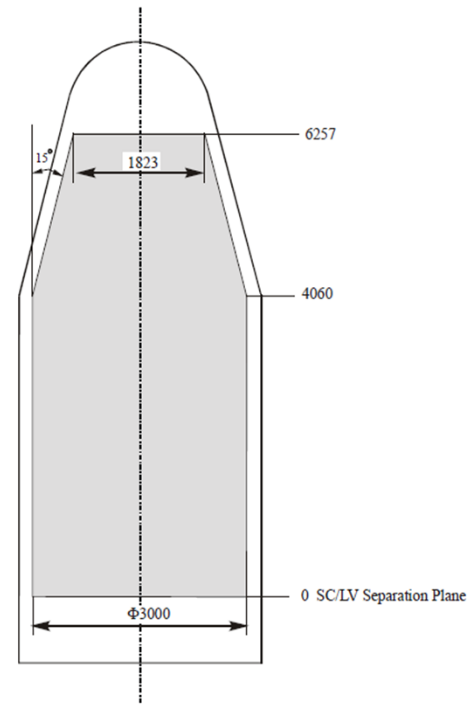
LM-2C fairing and adapters



LM-2C static envelope with 1194A interface

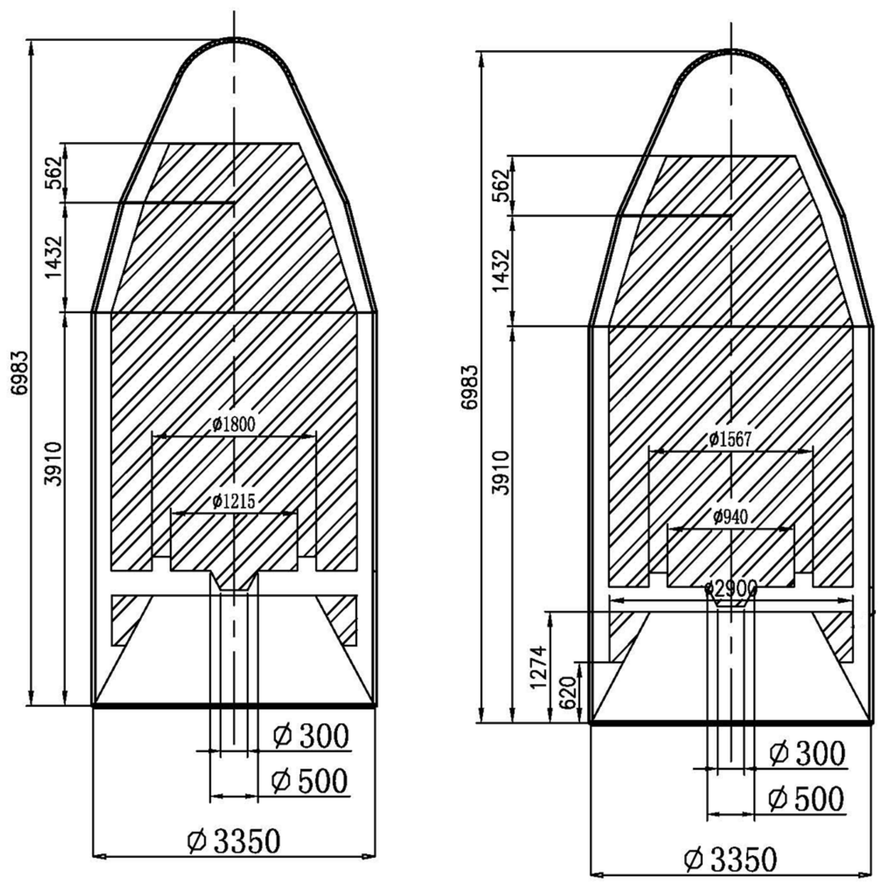


LM-2C static envelope with 937B interface



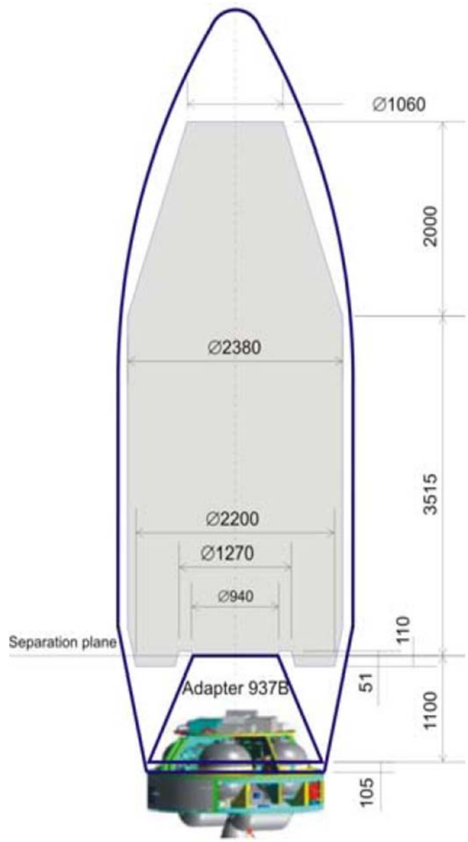
LM-2C/CTS static envelope with explosive bolt interface

LM-2D fairing and adapter



LM-2D fairing static envelope with 1194 (left) and 937 (right) interfaces

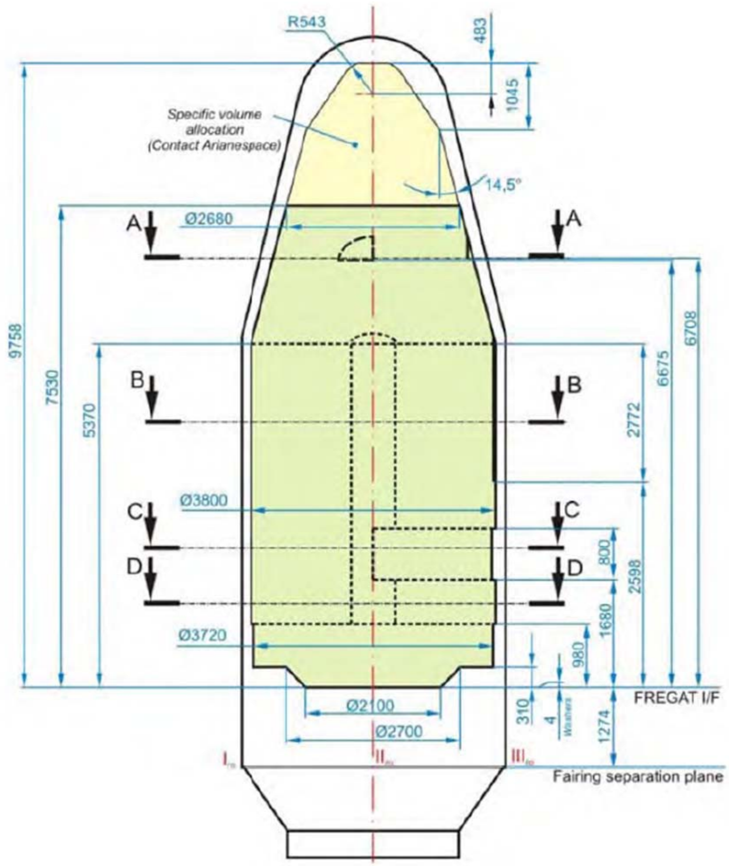
Vega fairing and adapters



Vega fairing dimensions with 937B adapter

Note: The Vega specific VESPA adapter is available for dual missions. The upper position allows passengers up to 1000 kg, while a 600 kg S/C can be accommodated inside the VESPA cavity.

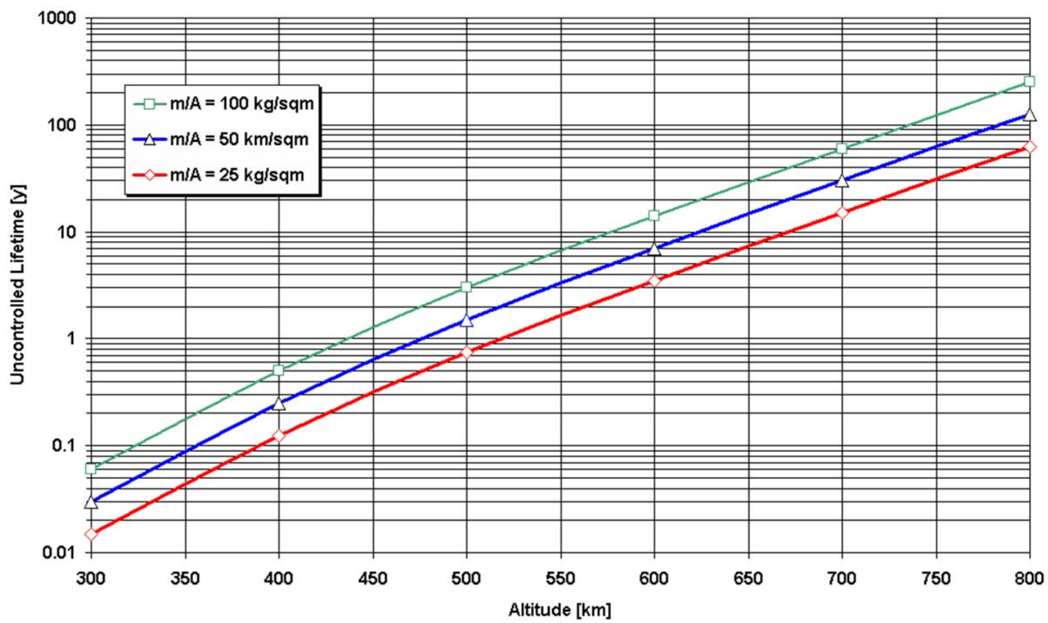
Soyuz fairing and adapters



Soyuz fairing dimensions

Note: Soyuz proposes standard adapters for multiple S/Cs. Those include adapters designed for the Globalstar 2 and Galileo missions, but also the SYLDA-S (under qualification) and the ASAP-S (designed for 1 main S/C of ~400 kg in the central position, and 4 external satellites in the 200 kg class).

Lifetime in LEO

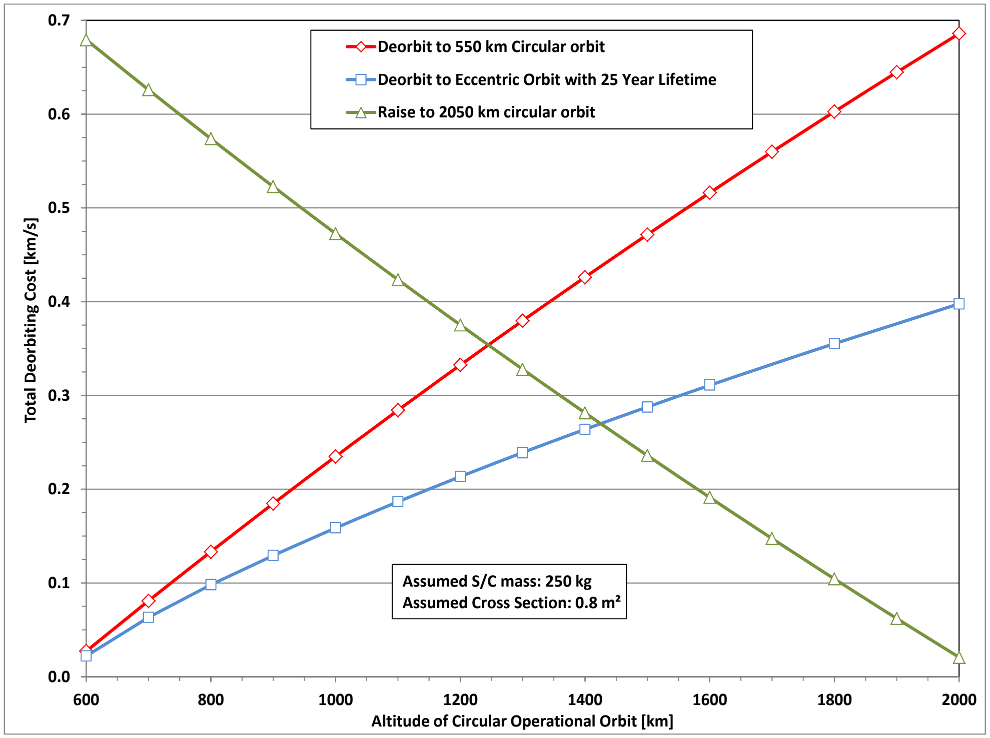


1. For the small-class mission under consideration here, a worst case mass to area ratio of ~ 300 kg for 1 m^2 is possible, meaning the LEO lifetime could be as high as 3 times the green curve above. With these characteristics, such a mission would re-enter within 25 years as long as it is in a circular orbit below ~ 550 km.

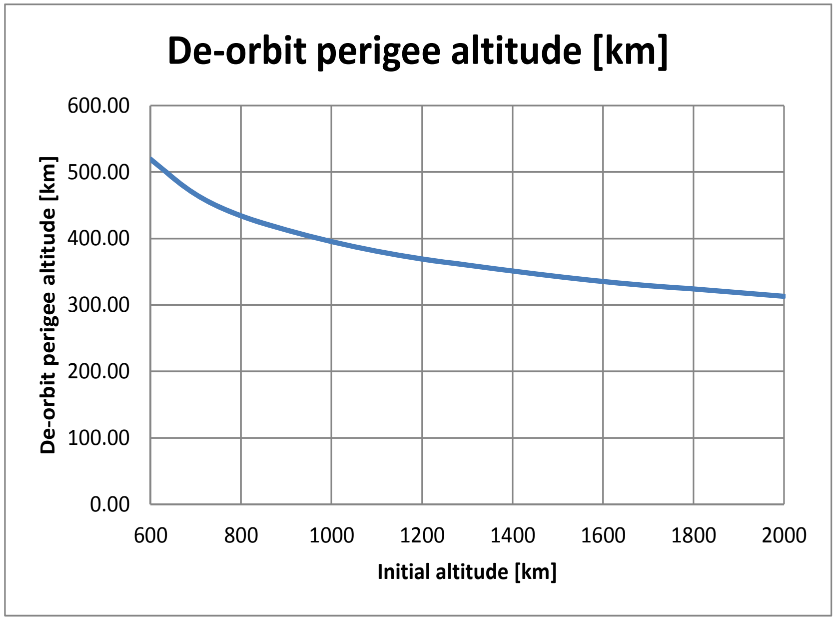
LEO de-orbiting strategy

1. Based on the previous slide, there are 3 possible strategies for orbital debris mitigation:
    a. Lower the altitude below 550 km, from which an un-controlled re-entry will follow within 25 years. This is achieved with a first manoeuvre to reduce the perigee, followed by a second manoeuvre to circularise the orbit.
    b. De-orbit to an eccentric orbit with a 25 years lifetime. Unlike the first option above, the second manoeuvre to circularise the orbit is not necessary: keeping the S/C in an eccentric orbit should ensure de-orbiting, as long as the perigee is low enough (lower than the 550 km for the circular orbit in the first option, see figure in next slide).
    c. Raise the altitude above 2000 km, outside of the LEO protected region. This is beneficial in terms of ΔV only if the initial altitude is already high enough.
2. The required ΔVs for all 3 options are shown in the next slide:
    a. Eccentric orbit solution is more advantageous for orbits with altitudes below 1400 km (and re-entry will occur in less than 25 years if the mass to area ratio of the S/C is equal to or lower than 250 kg for 0.8 mm^2)
    b. Raising the altitude above 2000 km is more advantageous for altitudes above ~1400 km. When retaining the most favourable case, the ΔV ranges from 20 to ~260 m/s.

LEO de-orbiting strategy



De-orbiting ΔVs for 3 options



Required perigee as a function of initial apogee altitude for eccentric de-orbiting solution (corresponds to the blue curve in the left figure)

ISO TRL table

Technology Readiness Level	Milestone achieved for the element	Work achievement (documented)
TRL 1: Basic principles observed and reported	Potential applications are identified following basic observations but element concept not yet formulated.	Expression of the basic principles intended for use. Identification of potential applications.
TRL 2: Technology concept and/or application formulated	Formulation of potential applications and preliminary element concept. No proof of concept yet.	Formulation of potential applications. Preliminary conceptual design of the element, providing understanding of how the basic principles would be used.
TRL 3: Analytical and experimental critical function and/or characteristic proof-of-concept	Element concept is elaborated and expected performance is demonstrated through analytical models supported by experimental data/characteristics.	Preliminary performance requirements (can target several missions) including definition of functional performance requirements. Conceptual design of the element. Experimental data inputs, laboratory-based experiment definition and results. Element analytical models for the proof-of-concept.
TRL 4: Component and/or breadboard functional verification in laboratory environment	Element functional performance is demonstrated by breadboard testing in laboratory environment.	Preliminary performance requirements (can target several missions) with definition of functional performance requirements. Conceptual design of the element. Functional performance test plan. Breadboard definition for the functional performance verification. Breadboard test reports.
TRL 5: Component and/or breadboard critical function verification in a relevant environment	Critical functions of the element are identified and the associated relevant environment is defined. Breadboards not full-scale are built for verifying the performance through testing in the relevant environment, subject to scaling effects.	Preliminary definition of performance requirements and of the relevant environment. Identification and analysis of the element critical functions. Preliminary design of the element, supported by appropriate models for the critical functions verification. Critical function test plan. Analysis of scaling effects. Breadboard definition for the critical function verification. Breadboard test reports.

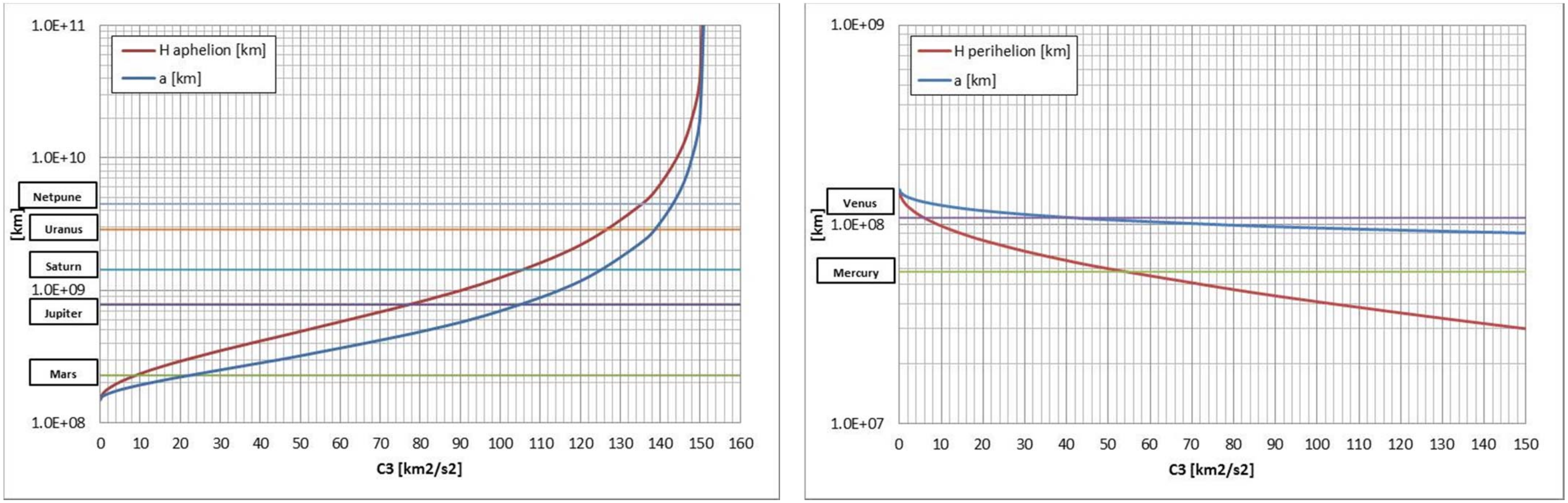
ISO TRL table

TRL 6: Model demonstrating the critical functions of the element in a relevant environment	Critical functions of the element are verified, performance is demonstrated in the relevant environment and representative model(s) in form, fit and function.	Definition of performance requirements and of the relevant environment. Identification and analysis of the element critical functions. Design of the element, supported by appropriate models for the critical functions verification. Critical function test plan. Model definition for the critical function verifications. Model test reports.
TRL 7: Model demonstrating the element performance for the operational environment	Performance is demonstrated for the operational environment, on the ground or if necessary in space. A representative model, fully reflecting all aspects of the flight model design, is built and tested with adequate margins for demonstrating the performance in the operational environment.	Definition of performance requirements, including definition of the operational environment. Model definition and realization. Model test plan. Model test results.
TRL 8: Actual system completed and accepted for flight ("flight qualified")	Flight model is qualified and integrated in the final system ready for flight.	Flight model is built and integrated into the final system. Flight acceptance of the final system.
TRL 9: Actual system "flight proven" through successful mission operations	Technology is mature. The element is successfully in service for the assigned mission in the actual operational environment.	Commissioning in early operation phase. In-orbit operation report.

C3 definition

1. Orbital velocity: $V = \sqrt{\mu.\left(\frac{2}{r} - \frac{1}{a}\right)}$
2. C3 parameter: $C_3 = -\frac{\mu}{a} = V^2 - \frac{2.\mu}{r}$
3. C3/2 is the specific energy of the orbit: therefore, C3<0 for elliptical orbits, C3 = 0 for the parabolic orbits and C3>0 for hyperbolic orbits.
4. For hyperbolic orbits, we also have $C_3 = {V_\infty}^2$, where $V_\infty = \lim_{r \to \infty} V$ is the velocity at infinity ($V_\infty = 0$ for the parabolic limit).
5. When applying the above formulas to the two-body system defined by the Earth and the spacecraft, C3 provides the escape velocity in the Earth referential frame. at infinity ($V_\infty = 0$ for the parabolic limit). For obtaining the spacecraft velocity in the heliocentric referential frame, the Earth orbital velocity must be added to V_∞.

C3 definition



C3 values required to reach the external (left) and inner (right) planets. The semi-major axes of the orbit of the external planets are indicated.

Assumptions:

- **Two-body approximation**
- **Direct Hohmann transfer**
- **Orbital insertion will require another ΔV**
- **Planet locations indicated on the Y axes correspond to their semi-major axes**

Note: For Mercury, Jupiter and beyond, typical transfers will involve gravity assists manoeuvres (e.g. JUICE and BepiColombo missions), to reduce the DV budget for the space segment.

References

- [1] LM-2C User's manual, issue 1999
- [2] Vega User's manual, issue 4.0, 2014
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- [4] Requirements on space debris mitigation for ESA projects, IPOL(2008)2 Annex 1
- [5] ISO/CD 16290, Space system – Definition of the TRL and their criteria of assessment, 2012
- [6] ESTRACK facilities manual (EFM), issue 1.1, 2008
- [7] http://sci.esa.int/home/51549-missions/