Impact of Controlled and Semi-Controlled Re-Entry on Spacecraft Design

Final Presentation

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11 December 2018

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Agenda

Morning: semi-controlled re-entry

09:30	Introduction and Flight Dynamics	[Kristen Lagadec]
11:30	System impacts	[Nicolas Leveque]
12:30	Lunch	

Afternoon: Controlled re-entry

13:30	Trajectory and mission drivers	[Nicolas Leveque]
14:15	Subsystems	[Nicolas Leveque]
15:15	Study cases	[Nicolas Leveque]
16:00	Questions & Discussion	
16:45	End of Meeting	

Semi-Controlled Re-entry

Introduction and Flight Dynamics

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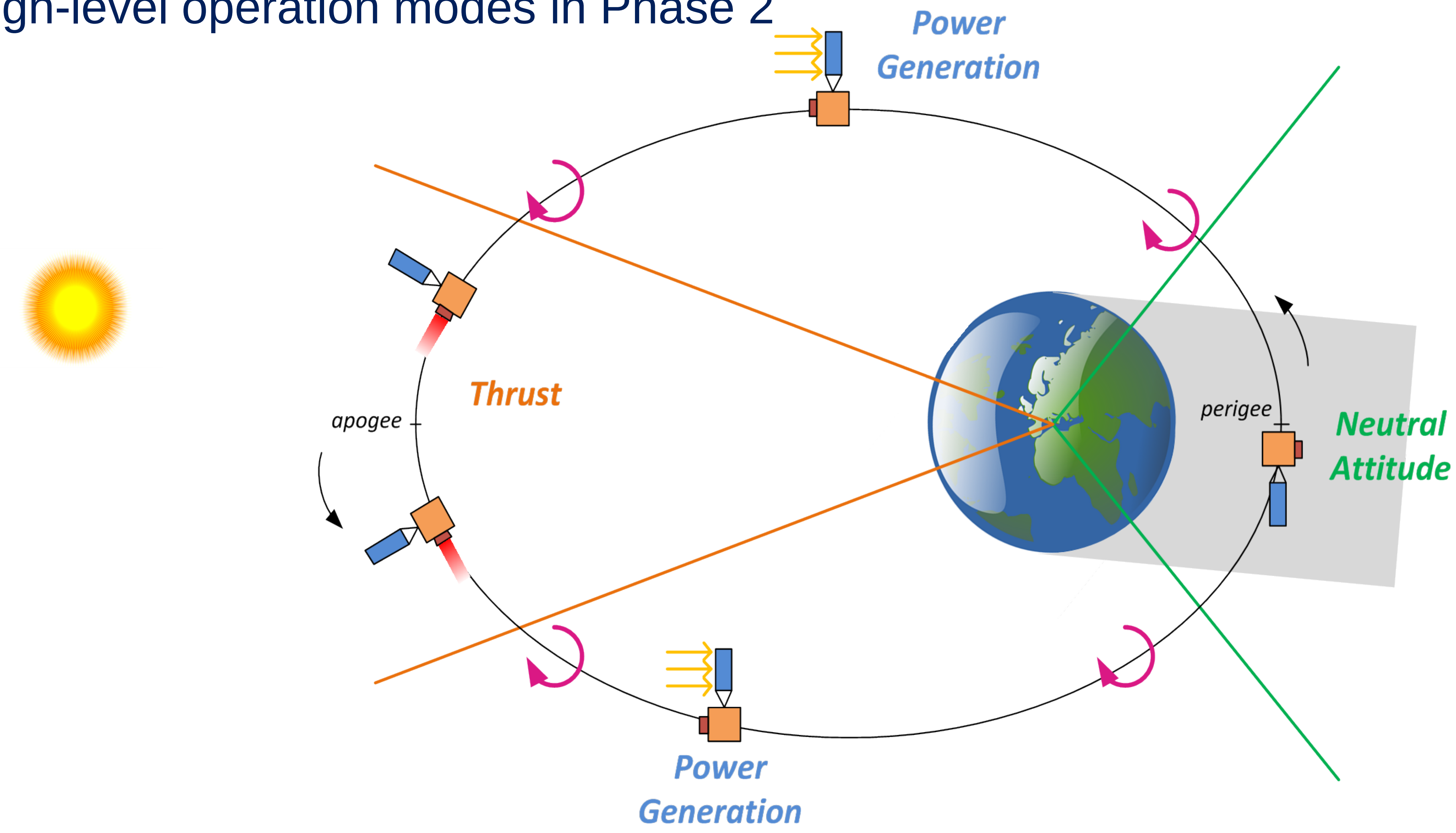
Semi-Controlled Re-entry

System Impacts

High-level operation modes in Phase 2



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Phase 2: Impact on Subsystems AOCS

Attitude Determination & Control

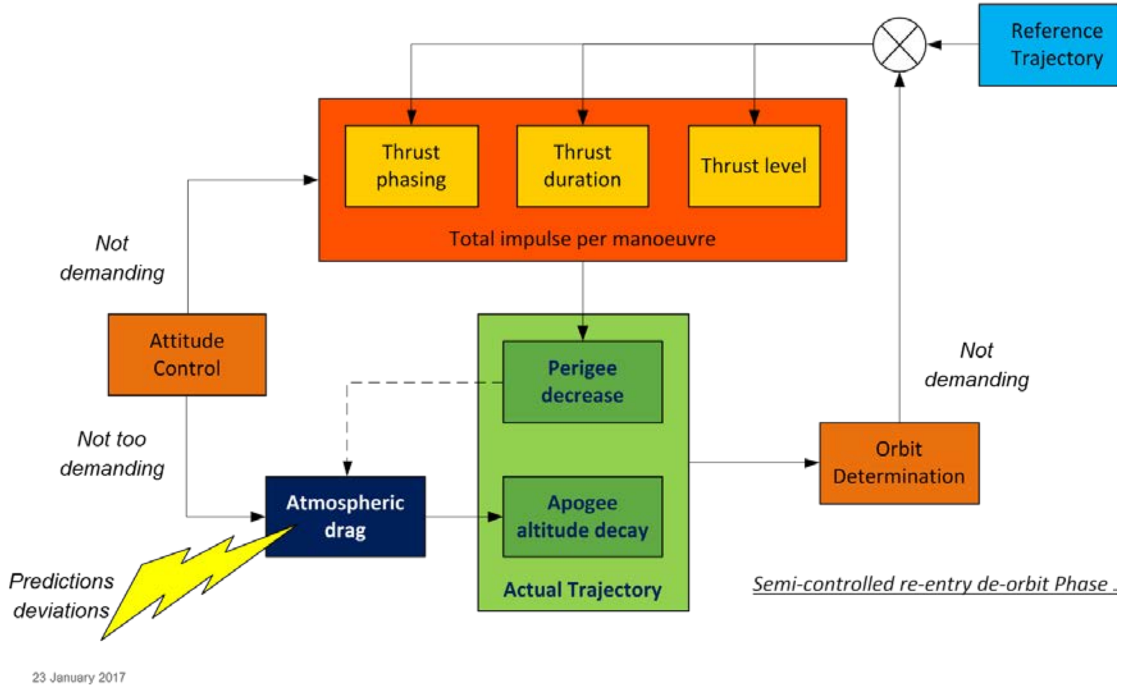
- Perigee Maintain the zero-aerodynamic-torque attitude
- Apogee Control thrust vector

In both cases, only a low pointing accuracy is required (in the order of ±1°)

- Thrust and zero-aerodynamic-torque both constrain 2 directions of the satellite
- The third direction can be clocked so as to ensure clear visibility of the star trackers

Acquisition of orbit determination data point is very relaxed, once an orbit is sufficient.

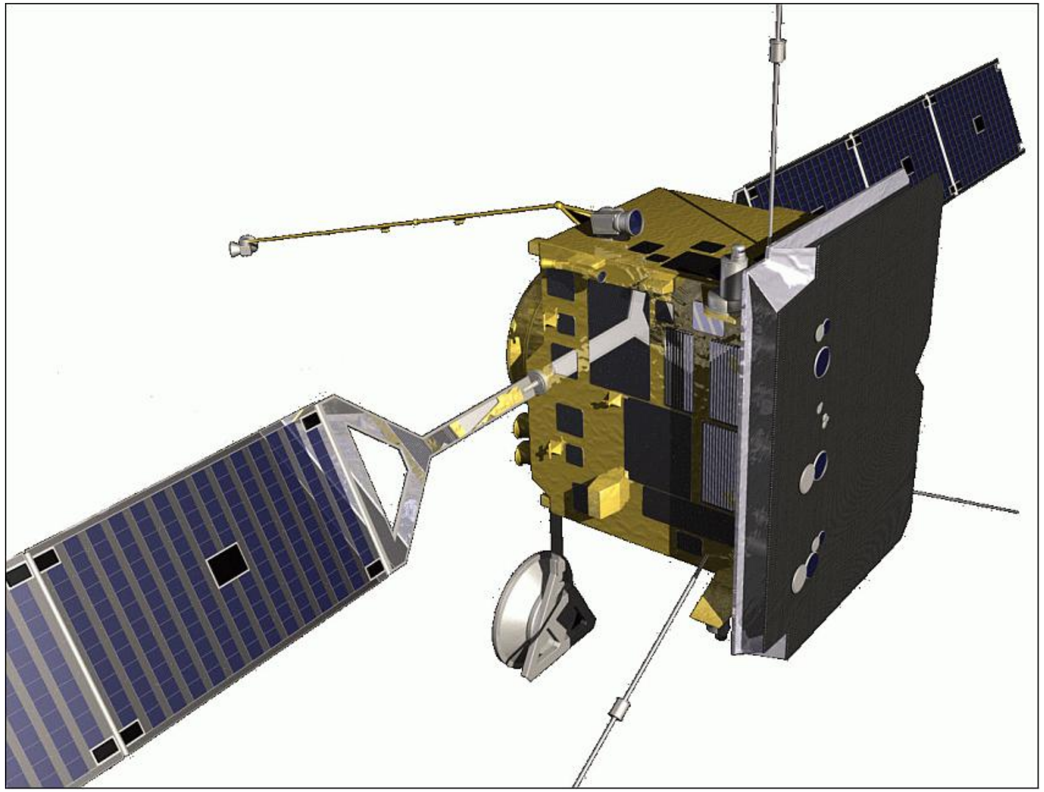
- No need for additional GPS antenna or for any special pointing requirement for acquisition than would normally exist.



Phase 2: Impact on Subsystems
Power System

Objectives:
- Maintain energy balance
- Provides power to Electric Propulsion System

A few drivers:
- Solar array size compatible with energy consumption and satellite operations (pointing)
  – Including EP energy consumption
- Bus voltage and LCLs for Electric Propulsion
- Battery capacity (& SoC) during the critical phase, including non-nominal
- Solar array temperature
  – Epoxy
  – Solder (melting at ~200°C → ~400°C, SolO, BepiColombo)



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Phase 2: Impact on Subsystems
Thermal Control

Objective:

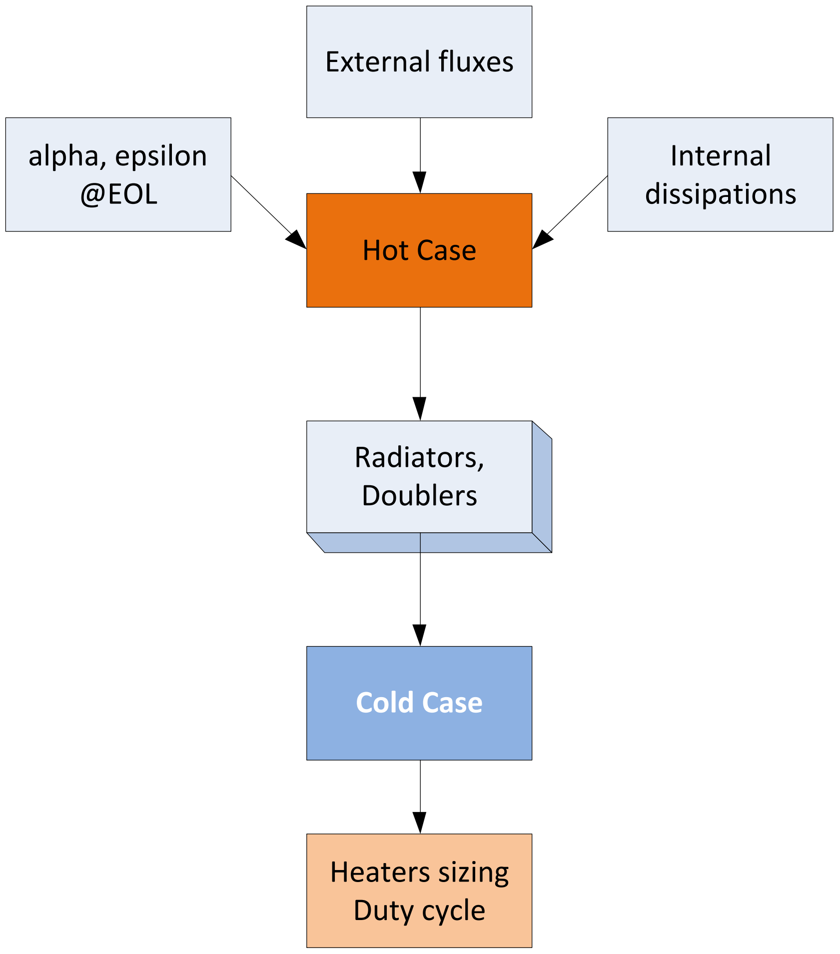
- Ensure that the platform equipment remain within their operational temperatures

Different environment compared to “classical” missions:

- Changes in view factors
- Introduction of aerothermodynamic heat fluxes
- Increased ATOX flux, degradation
- Different heat loads inside

The controlled descent may be a dimensioning hot case

- Driving the size of radiators & doublers
  - Potentially increasing power consumption in other modes (nominal mode, safe mode)



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Phase 2: Impact on Subsystems
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Casualty risk (uncontrolled) > Casualty risk (semi-controlled)
➔ Avoid interrupting the controlled descent

Replace “Fail Safe” by “Fail Operational” as much as possible, or “Fail Safe Operational”
- Thrust at apogee can be missed for a couple of successive apogees, and still be compensated for later.
  - E.g. priority to minimum bus voltage
- Need for high degree of autonomy → limit need for ground intervention / troubleshooting
  - Frequency of TM dumps during controlled descent?

Semi-Controlled Re-Entry

Case 1

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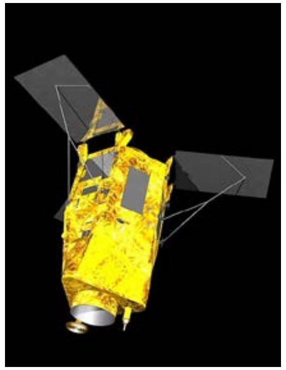
Mission Overview

- A small platform with an optical instrument which is responsible for most of the casualty area.
- Fixed deployable solar arrays, with a large cross-section area at perigee
  – hence a relatively higher-thrust level needed at apogee
- Thruster mounted on a side wall
  – Of interest for the nominal mission: potentially possible to thrust while imaging
    – Only if instrument has somewhat relaxed thermal requirements (not constraining attitude)
- AOCS:
  – Magnetic Safe Mode: no propulsion involved
    → no torque control, only thrust from the EP

	Case 1
Mass	750 kg
Initial orbit	800 km, SSO (10:00 LTDN)
Max Delta-V at apogee	90 mN (15 min)
Propulsion	HET (PPS-1350-G), 1650 s, 1.5 kW
AOCS capability	20 N.m.s
Casualty risk area	20 m^2

Contributor	Delta-V (m/s)
Orbit Injection correction and orbit acquisition	25
Orbit Maintenance	65
Collision Avoidance	10
De-orbit	160
Margin	40
Total budget	**300**



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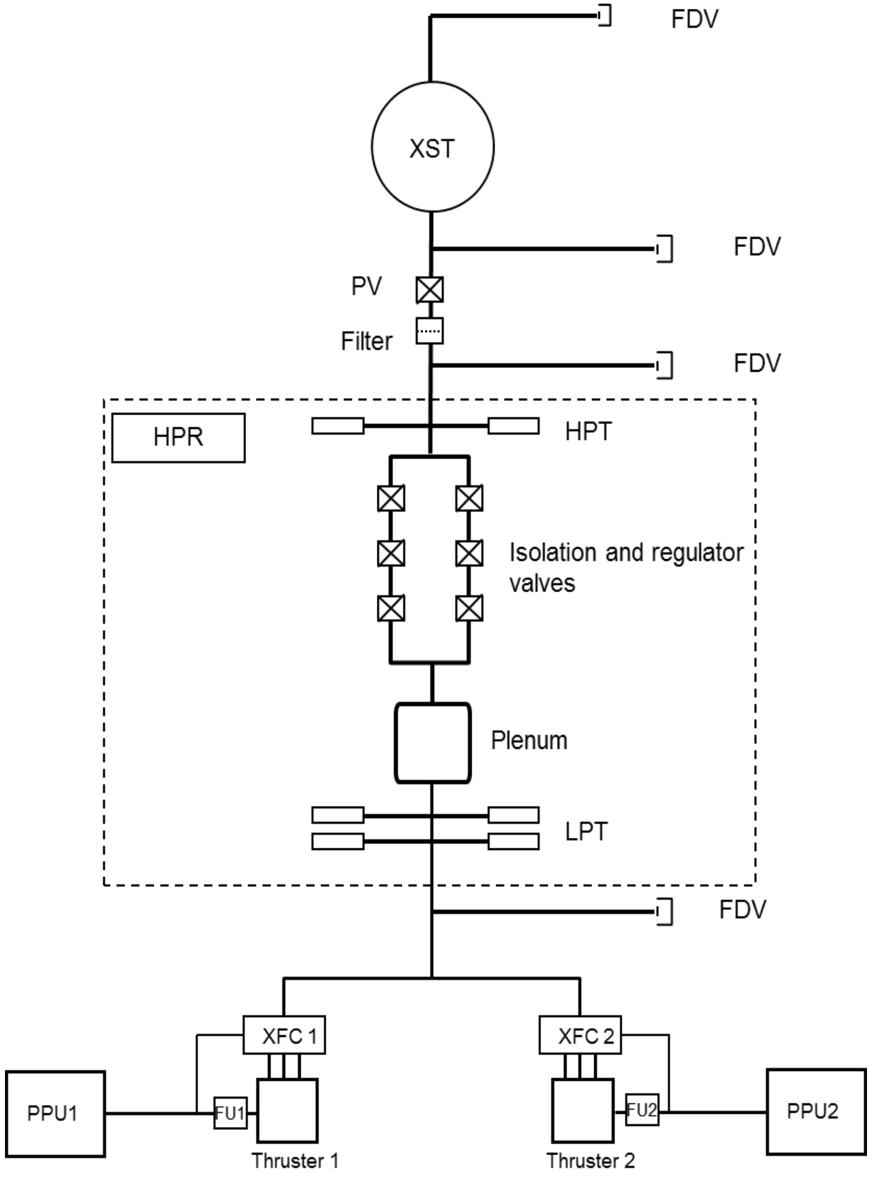
Satellite Design
Propulsion System

Thruster: Safran PPS-1350-G Hall Effect Thruster

Total impulse required ~240 kN.s
- An order of magnitude smaller than the thruster capability
- Mission Delta-V budget: 300 m/s

Architecture:
- One thruster needed (nominal) + one for redundancy
  - Each with dedicated PPU
    - OTS PPU for this thruster: TAS Mk2 (100 V Regulated, 2.5 kW max. output power)
- Single tank for 16.3 kg Xe → 13 L tank (10.2 L needed)
- Electronic High Pressure Regulator (Bepi)



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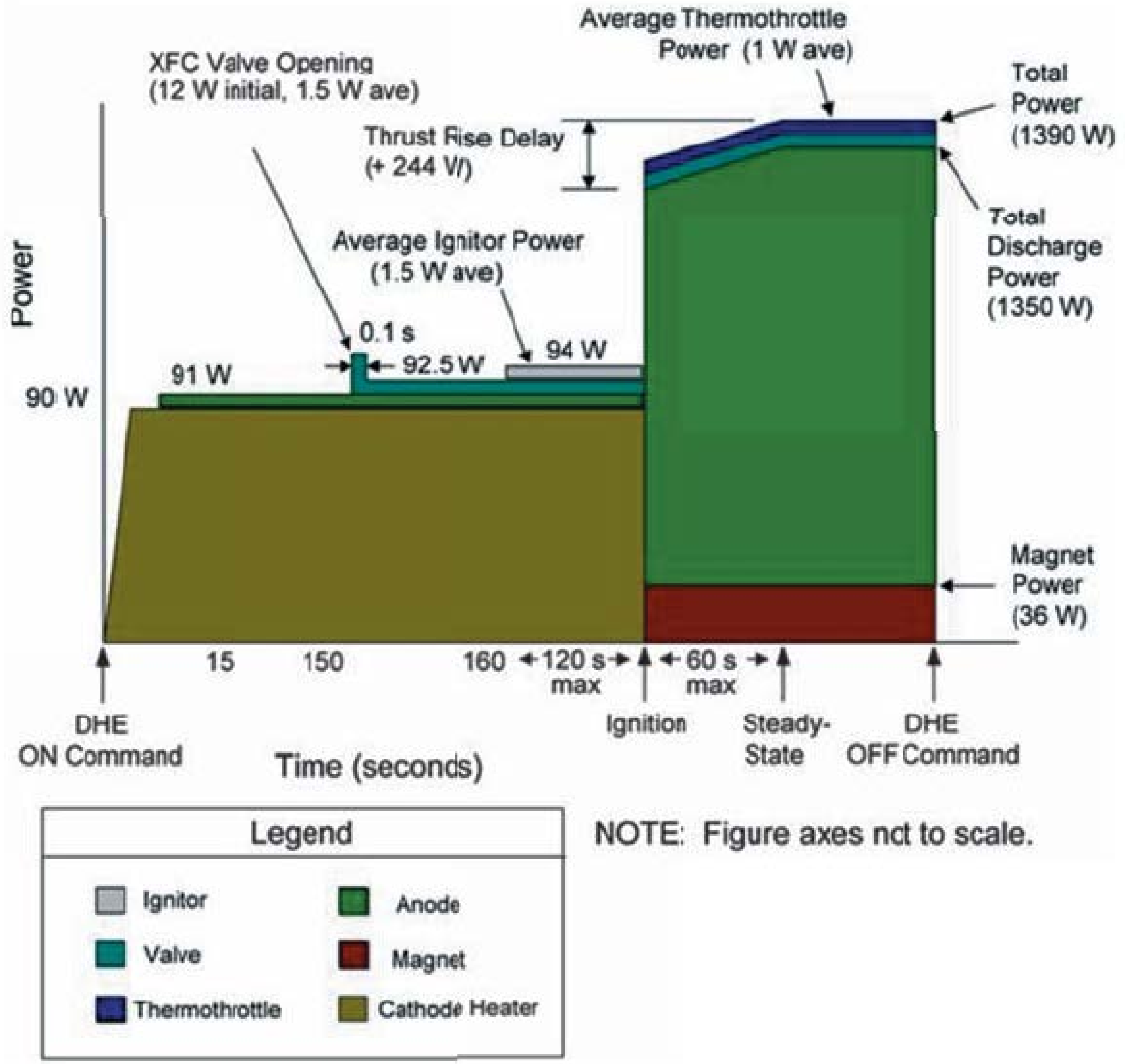
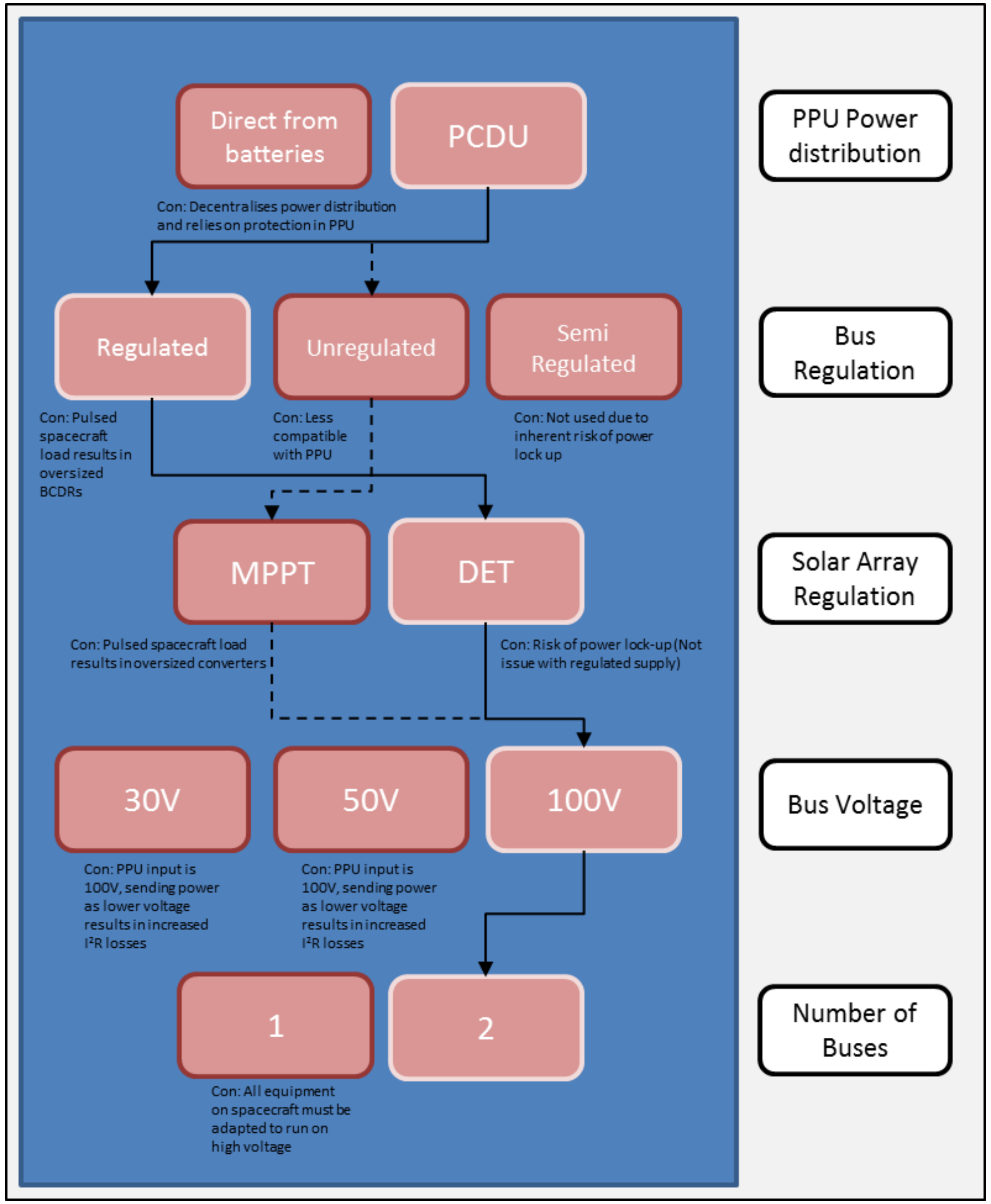
Hall-Effect Thruster Start-up Sequence

Reasonable start-up duration, less than 5 minutes (SPT-100)



Satellite Design
Power System

- Two power bus
  - Primary bus: 100V (for high power demand of propulsion system)
  - Secondary bus with lower voltage for rest of satellite
- Power distribution via PCDU
- Bus regulation & Solar array regulation
  - DET with regulated bus
  - MPPT with unregulated bus
    - The former is preferable for regulated bus input to PPU



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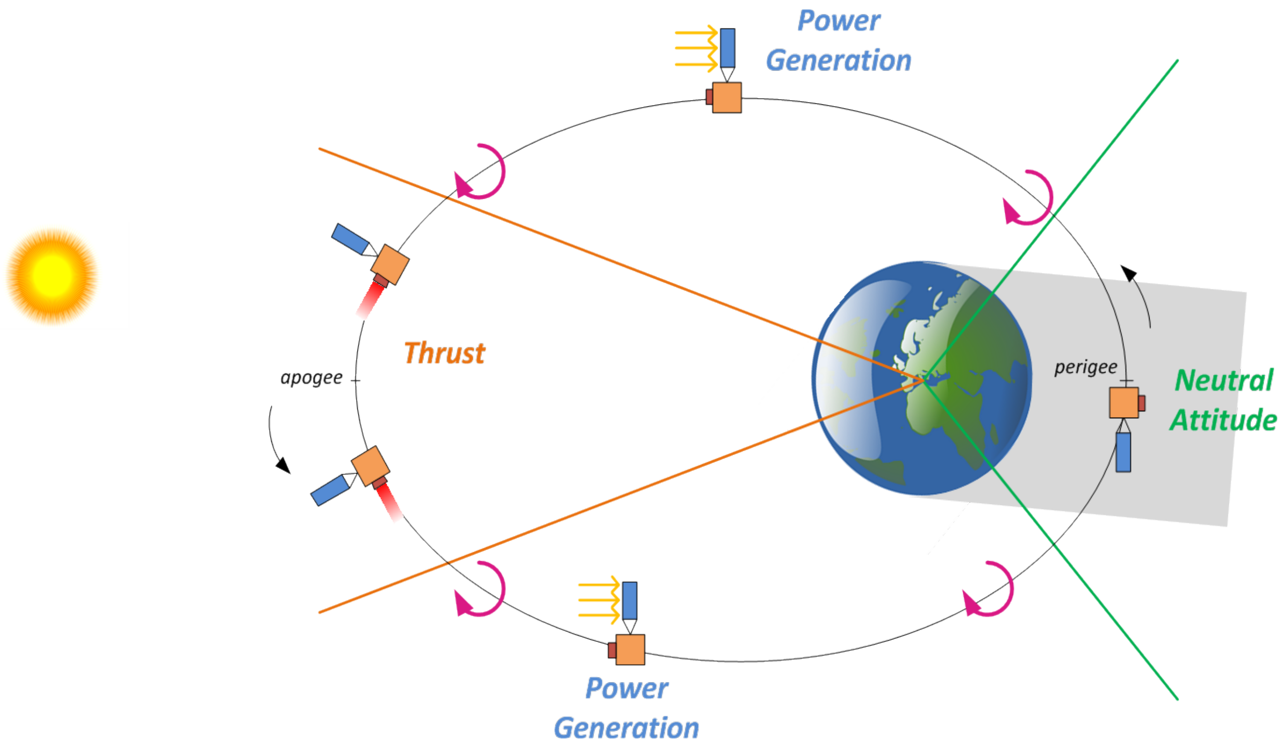
Satellite Design
AOCS

Based on the capability of the normal mode to target 3 main attitudes during the whole descent:

1. Attitude for the apogee boost to decrease the perigee.
   It may also be used for manoeuvres at perigee during Phase 1, it needed.
   Only one thruster is available to perform velocity increment.
2. Attitude that minimizes aerodynamic torque around perigee of Phase 2, to avoid reaction wheel saturation.
3. Attitude used to point the solar array toward sun direction to recharge the battery.



The main actuator is the reaction wheel cluster

- Its angular momentum capacity determines the transition between phase 2 and 3, and the minimum controlled apogee in phase 2.
- More generally it determines the altitude below which it is necessary to fly with a specific aerodynamic attitude.

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Satellite Design
Structure & Thermal

Structure based on S5p, Spot 6/7

- Thruster mounted on a dedicated structure on a lateral wall
- Aligned with CoM (vertical position)
- Tank located nearby, in a position minimising the shift of CoM
  - Possibly under the top floor (in Hydrazine systems, propulsion module seats on the bottom floor)

Thermal

- Based on the same thermal design (radiators, doublers, thermistor-controlled heaters, MLI)
- SCR will introduce some modifications
  - Mainly, Betacloth on the front face at passage of perigee for protection against ATOX
    - For Case 1, minor impact: the instrument would be pointed towards the flight direction
      - Degradation of optical surfaces, no longer an issue during disposal

Operations, Autonomy & FDIR

Mission Phases

- Nominal mission
  - Effect of low-thrust propulsion on the duration of manoeuvres and thus mission unavailability
  - For case 1, the orientation of the thruster w.r.t. instrument LoS would make it possible to operate the two together → little or no interruption of service.
- Disposal phase: as described before

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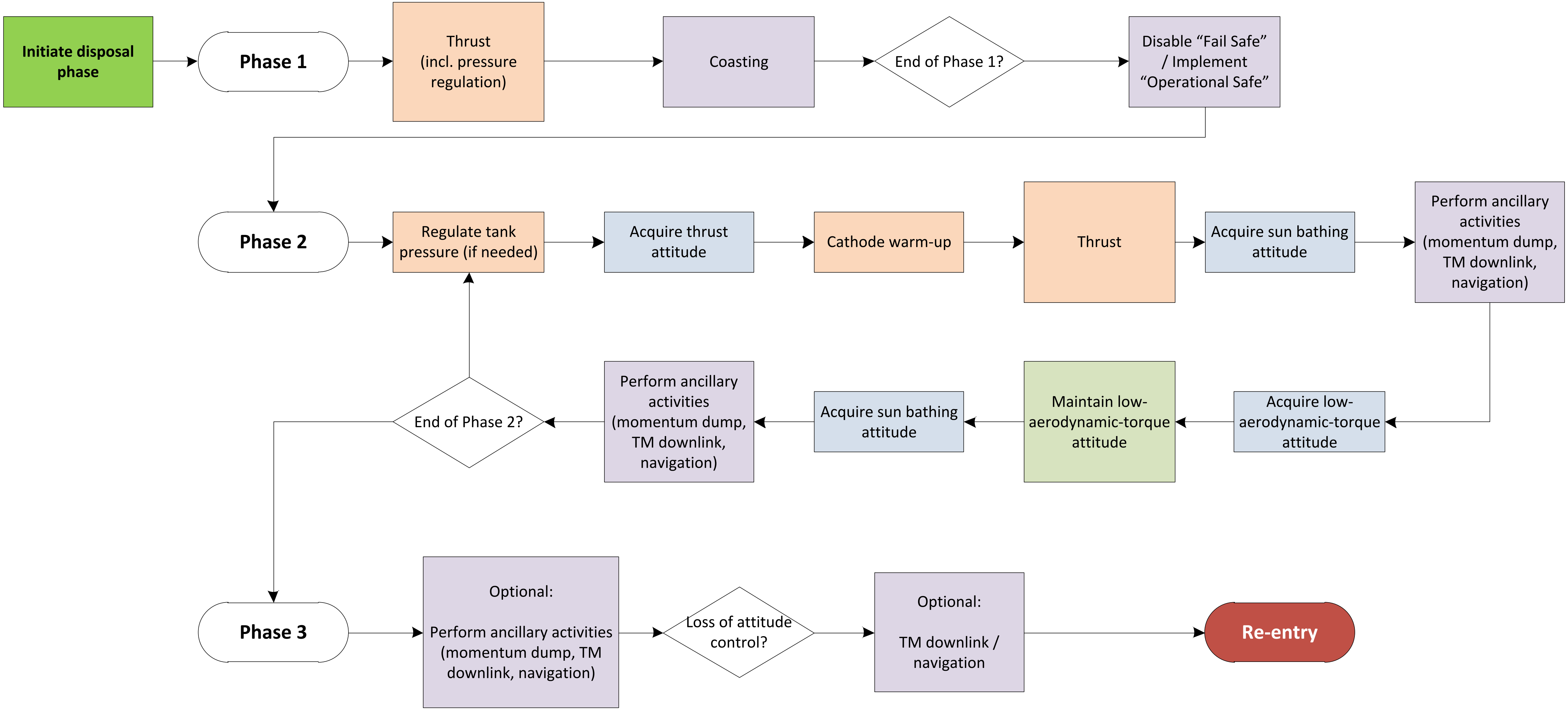
Disposal Phase 2

- Continuation of descent is less risky than its interruption
  - Continuation: still a chance to achieve the semi-controlled re-entry
  - Interruption of descent would lead to uncontrolled re-entry
- Fail Operational rather than Fail Safe
  - MetOp-SG strategy:
    - First failure = Fail Operational
    - Second failure = Fail Safe

Operations



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Semi-Controlled Re-Entry

Case 2

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Mission Overview

- A medium-large platform with 3 payload instruments
  - At least one of these could have stringent thermal stability requirements during the nominal mission
- Single wing, rotating solar arrays, with a moderate cross-section area at perigee
- Thruster mounted on the bottom floor
- AOCS:
  - Magnetic Safe Mode: no propulsion involved
    → no torque control, only thrust from the EP

	Case 1
Mass	1500 kg
Initial orbit	800 km, SSO (10:00 LTDN)
Max Delta-V at apogee	240 mN (15 min)
Propulsion	Arcjet (MR-510), 600 s, 2 kW
AOCS capability	40 N.m.s
Casualty risk area	25 m^2

Contributor	Delta-V (m/s)
Orbit Injection correction and orbit acquisition	25
Orbit Maintenance	65
Collision Avoidance	10
De-orbit	175
Margin	25
Total budget	**300**

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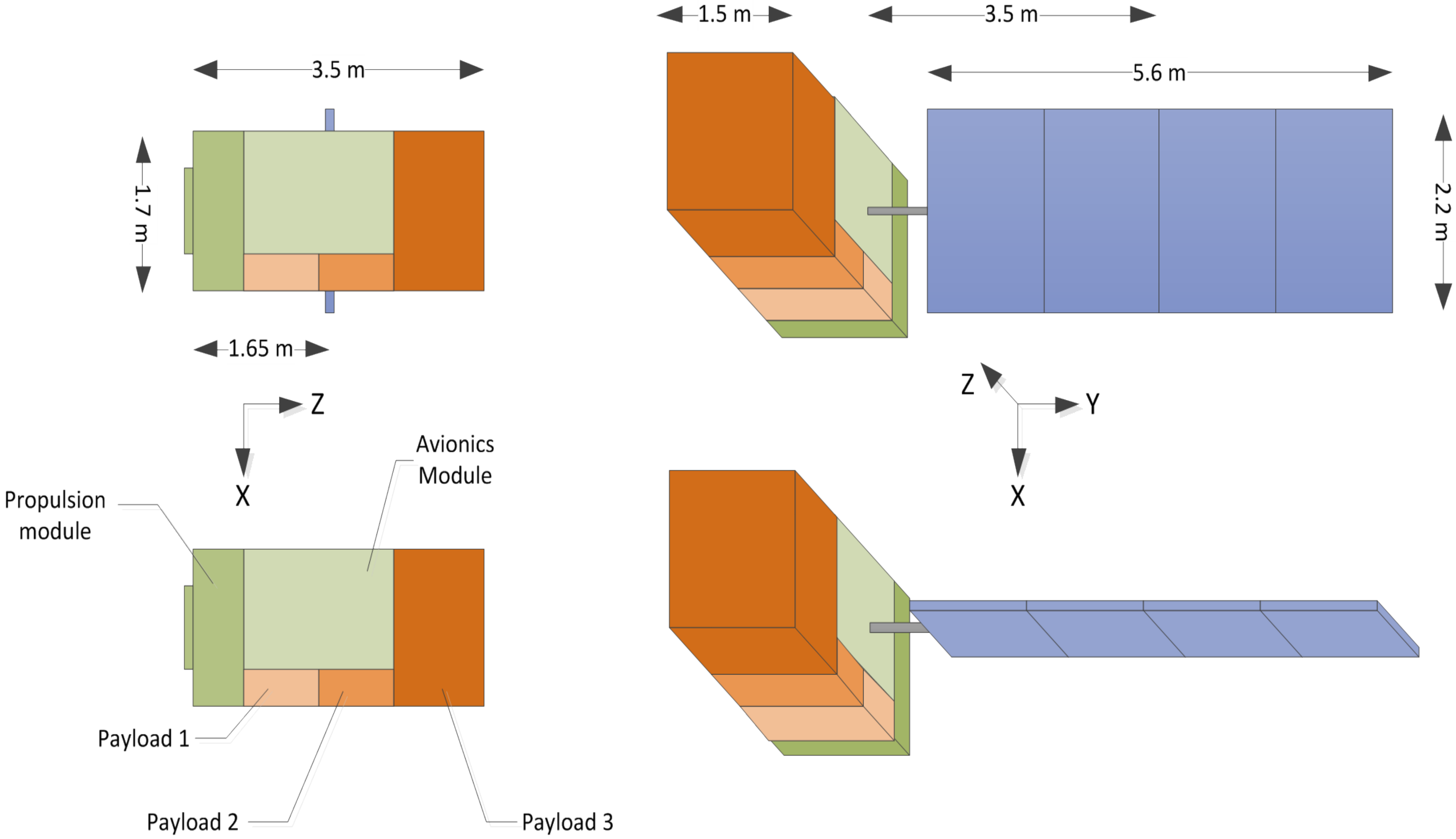
Case 2 configuration



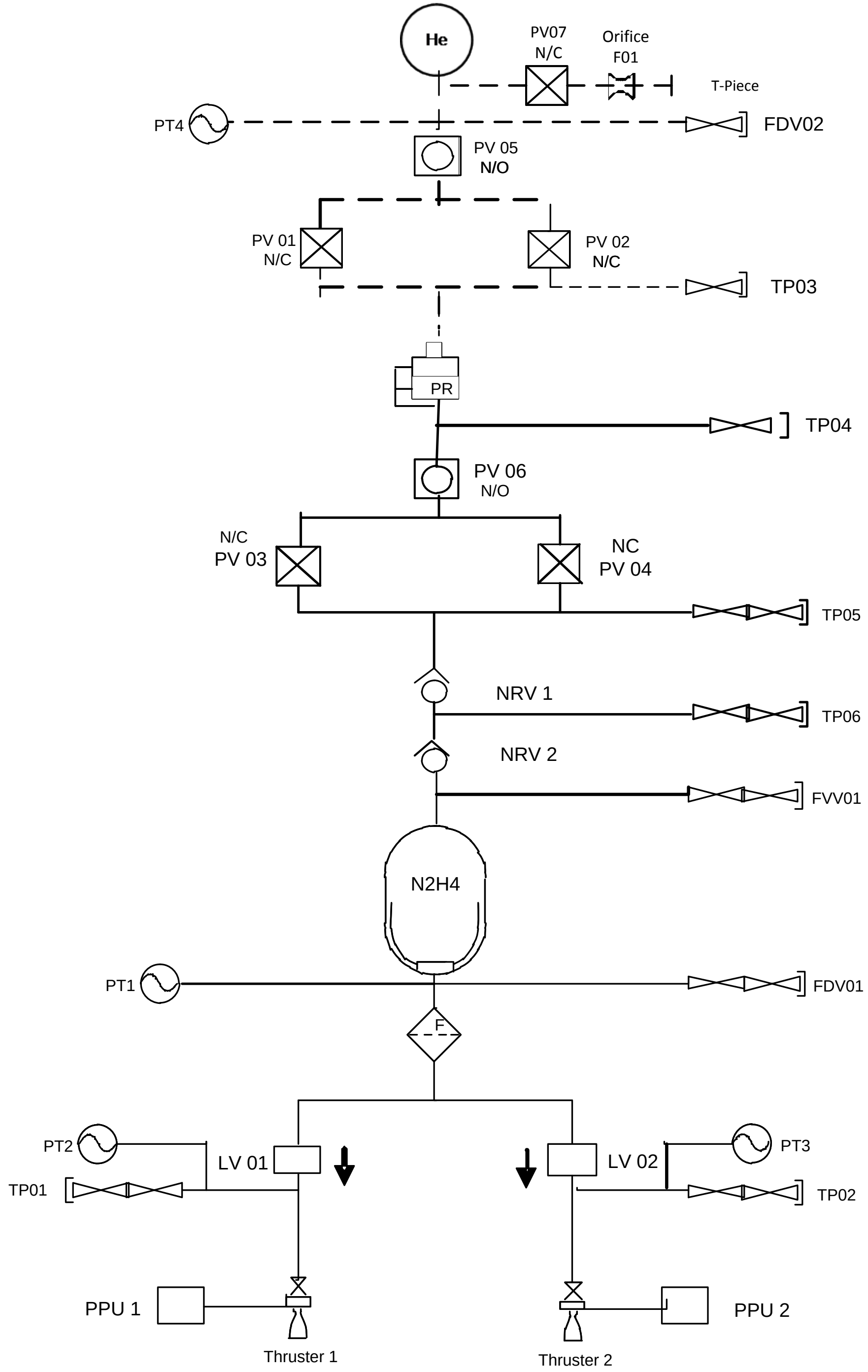
Satellite Design
Propulsion System

Thruster: Aerojet MR-510 hydrazine arcjet

Total impulse required ~450 kN.s
- About 1/3 of the thruster capability
- Mission Delta-V budget: 300 m/s

Architecture:
- One thruster needed (nominal) + one for redundancy
  - Each with dedicated PPU
- Single tank for 87.8 kg → 88 L tank (10.2 L needed)
- Pressurisation system
  - Regulated pressure of 17.5 bar assumed

- PPU (or PCU) from Aerojet
  - 69 V Regulated



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Power Budgets

		Nominal mission		Controlled descent	
Contributors	Power	duty cycle	Average power	duty cycle	Average power
Core Avionics	400	100%	400	100%	400
PDHT (average)	100	100%	100	0%	0
Payload 1	150	100%	150	0%	0
Payload 2	150	100%	150	0%	0
Payload 3	200	100%	200	0%	0
EP	2000	0%	0	20%	400
Sum			1000		800
Losses & margins		25%	250	25%	200
Average Power budget			1250		1000

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Case 3

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Mission Overview

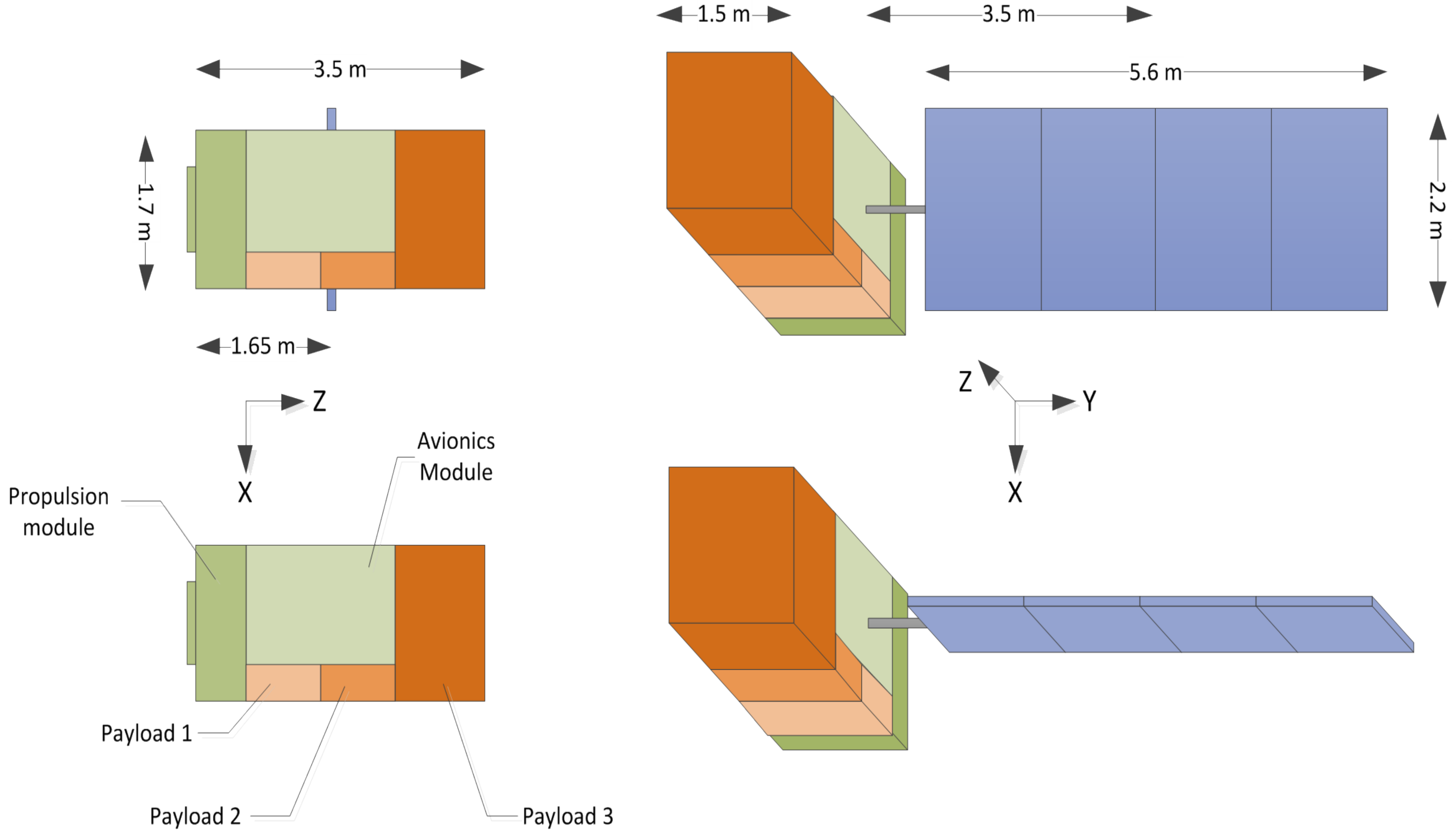
- A medium-large platform with 3 payload instruments
  - At least one of these could have stringent thermal stability requirements during the nominal mission
- Single wing, rotating solar arrays, with a moderate cross-section area at perigee
- Thruster mounted on the bottom floor
- AOCS:
  - Magnetic Safe Mode: no propulsion involved → no torque control, only thrust from the EP

	Case 1
Mass	1500 kg
Initial orbit	800 km, SSO (10:00 LTDN)
Max Delta-V at apogee	120 mN (20 min)
Propulsion	HET PPS-1350-E
AOCS capability	40 N.m.s
Casualty risk area	25 m^2

Contributor	Delta-V (m/s)
Orbit Injection correction and orbit acquisition	25
Orbit Maintenance	65
Collision Avoidance	10
De-orbit	175
Margin	25
Total budget	**300**

Case 3 configuration

Same as Case 2



Summary of the Study Cases

- The three scenarios presented are all clearly feasible (Cases 2 and 3 are in effect identical), with high to very high reductions in casualty risk.

- The estimates for the manoeuvre impulse have been found to be oversized, and not constraining the feasibility domain.
  - Low-thrust propulsion may be more constraining on the nominal mission than on the disposal phase

- Semi-controlled re-entry is generally feasible for a wide range of satellites with low-thrust propulsion systems.

Controlled Re-entry

Introduction

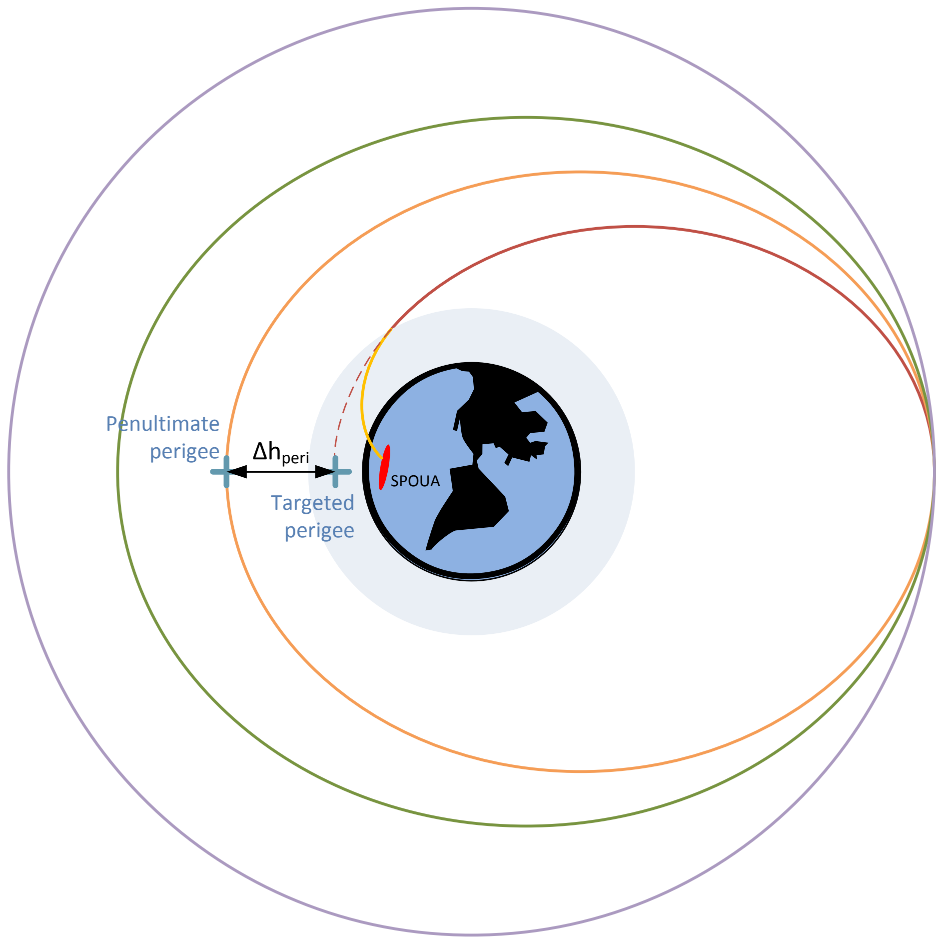
Definitions

Targeted perigee: perigee of the last, incomplete orbit.

- typically below 80 km. Due to the high atmospheric density and associated drag force, the satellite will fall down on the surface of the Earth.

Penultimate perigee: last perigee before the final manoeuvre, thus preceding the targeted perigee.

- Typically exceeds 150 km, and less than 450 km.



Controlled Re-entry: Key requirements

The key is to target a “well-defined impact footprint”

- Most LEO missions have inclinations between 35°-145°
- Hence, the South Pacific Ocean Uninhabited Area (SPOUA) is the most suitable disposal area

6.3.1	**Probability of successful disposal**
6.3.1.1	The probability of successful disposal of a spacecraft or launch vehicle orbital stage shall be at least 0.9 at the time disposal is executed.
6.3.1.2	The probability of successful disposal, as discussed in Annex A, shall be evaluated as conditional probability weighted on the mission success, i.e. P(D|M).
6.3.1.3	The start and end of the disposal phase, as illustrated in Annex B, shall be chosen so that all disposal actions are completed within a period of time that ensures compliance with 6.3.1.1.
6.3.3	**LEO disposal manoeuvres**
6.3.3.1	A spacecraft or launch vehicle orbital stage operating in the LEO protected region, with either a permanent or periodic presence, shall limit its post-mission presence in the LEO protected region to a maximum of 25 years from the end of mission.
6.3.3.2	After the end of mission, the removal of a spacecraft or launch vehicle orbital stage from the LEO protected region shall be accomplished by one of the following means (in order of preference): a) retrieving it and performing a controlled re-entry to recover it safely on the Earth, or b) manoeuvring it in a controlled manner into a targeted re-entry with a well-defined impact footprint on the surface of the Earth to limit the possibility of human casualty, or c) manoeuvring it in a controlled manner to an orbit with a shorter orbital lifetime that is compliant with 6.3.3.1, or d) augmenting its orbital decay by deploying a device so that the remaining orbital lifetime is compliant with 6.3.3.1, or e) allowing its orbit to decay naturally so that the remaining orbital lifetime is compliant with 6.3.3.1, or manoeuvring it in a controlled manner to an orbit with a perigee altitude sufficiently above the LEO protected region that long-term perturbation forces do not cause it to re-enter the LEO protected region within 100 years.
6.3.4	**Re-entry**
6.3.4.1	For the re-entry of a spacecraft or launch vehicle orbital stage (or any part thereof), the maximum acceptable casualty risk shall be set in accordance with norms issued by approving agents.
6.3.4.2	The re-entry of a spacecraft or launch vehicle orbital stage (or any part thereof) shall comply with the maximum acceptable casualty risk according to 6.3.4.1.

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Probability and Reliability

A word of caution:

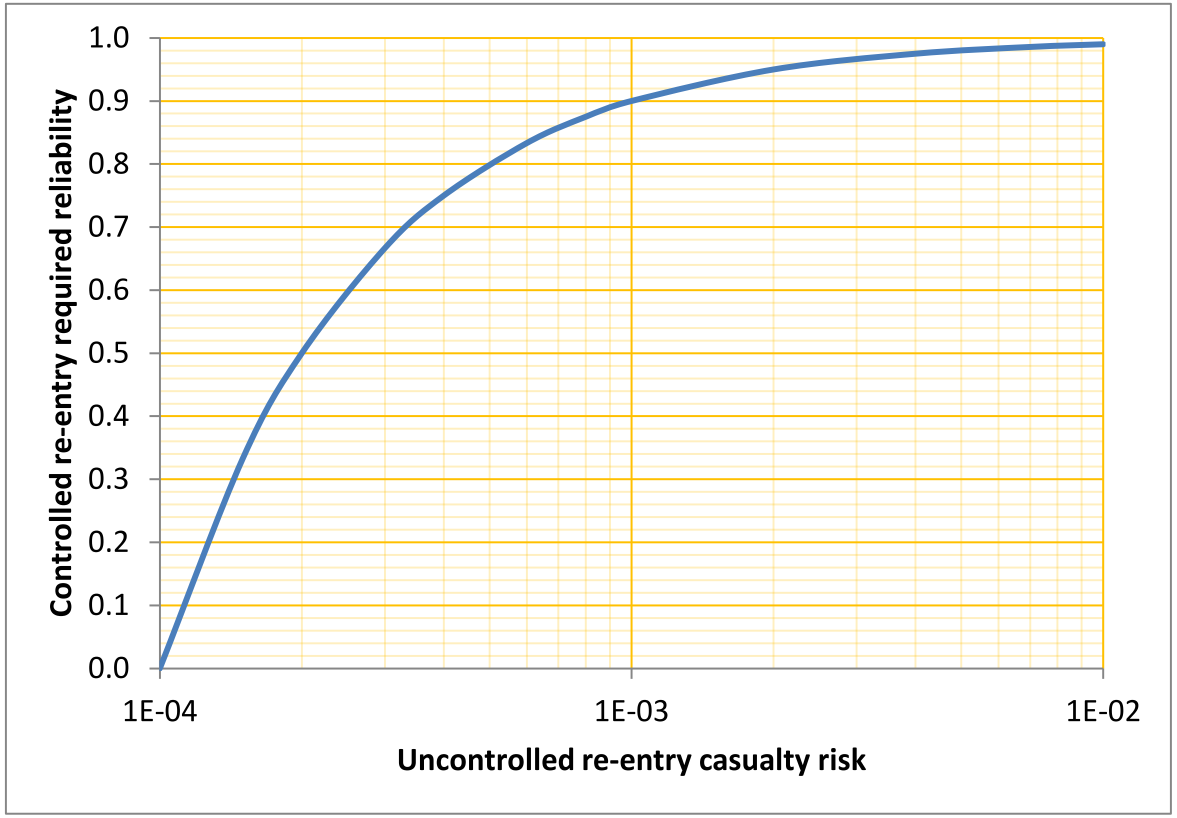
- Requirement 6.3.1.1 from ISO-24113 is only about successful disposal with respect to the LEO Protected Region after the end of the nominal mission.

- The reliability required for successful controlled re-entry is driven by the acceptable casualty risk
  - Casualty expectancy for a combined case of nominal (controlled re-entry) and non-nominal (uncontrolled re-entry)

$$E_{C,combined} = E_{C,nom} R_{nom} + \sum_{k=1}^{N} E_{C,off-nom,k} P_{off-nom,k}$$

$$R_{nom\ (CRe)} = 1 - P_{off-nom\ (URe)} = 1 - \frac{10^{-4}}{E_{C,off-nom\ (URe)}}$$

6.3.1.1	The probability of successful disposal of a spacecraft or launch vehicle orbital stage shall be at least 0.9 at the time disposal is executed.



Mission Drivers

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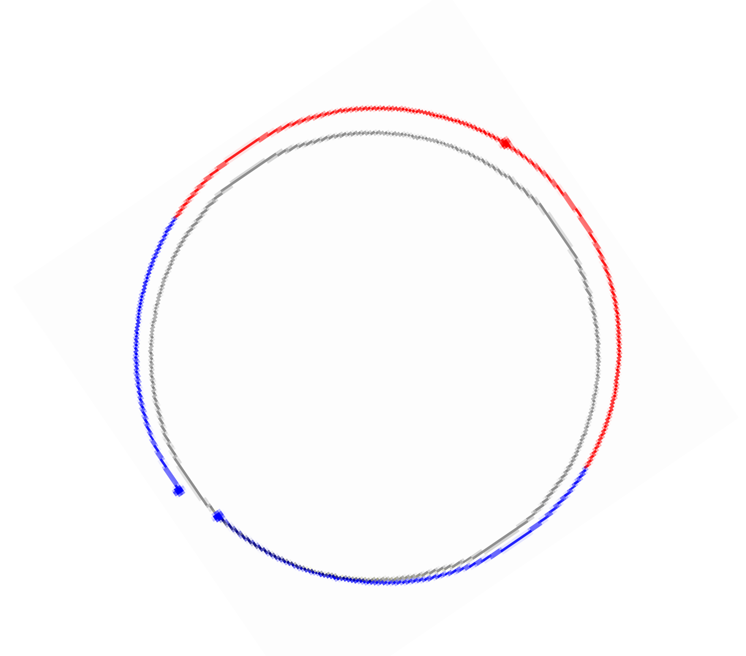
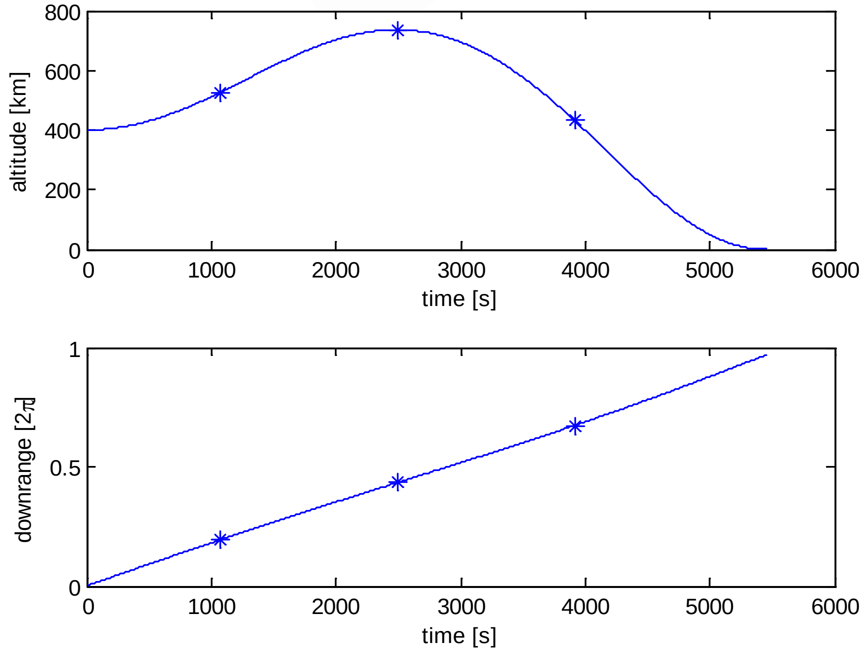
Trajectory Requirements

Downrange of the centre of the debris cloud from the penultimate perigee, expressed in number of revolutions:
- > 0.88 for h_{fp} > 0 km
- ~ 1 for h_{fp} = 60 km

The centre of the SPOUA is ~45°S
If we aim for the centre of the SPOUA with 30 km targeted perigee, then the argument of perigee would be:
- -63° latitude → ω = 243° if travelling from North to South
- -27° latitude → ω = 333° if travelling from South to North

The choice of the argument of perigee has significant effects on the performances of the controlled re-entry.
- Both the mean length and the uncertainty in the length of the debris dispersion cloud are minimum for ω = 180°/360°
- For ω = 270°, they increase respectively of 71% and 64-times with respect to their minimum values, for the worst-case of high spacecraft ballistic coefficient.





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Altitude of the targeted perigee

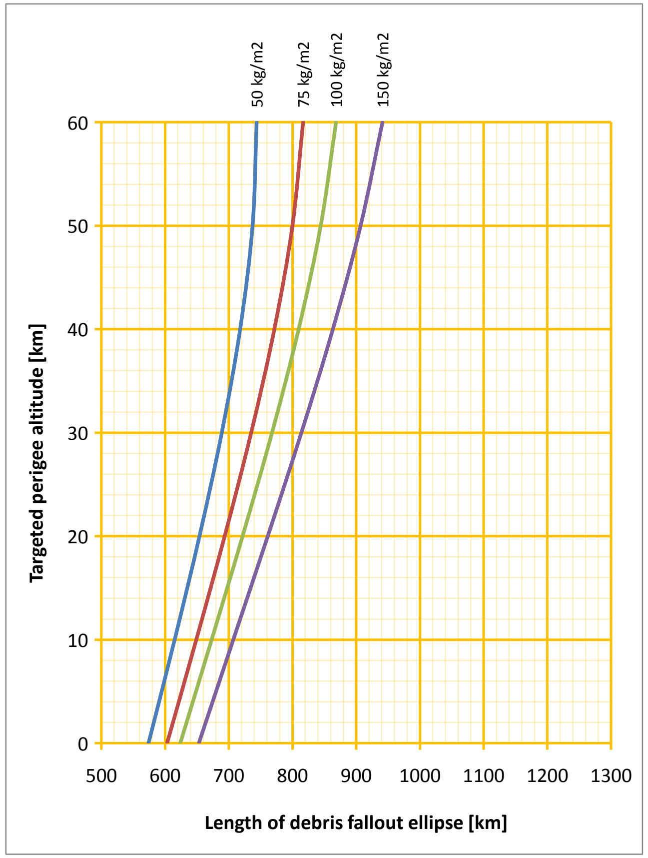
Its choice affects the main realised performance defining the casualty risk of the re-entry phase:

- Size of the SRA (Properties of the surviving debris:)
- Realised dispersions

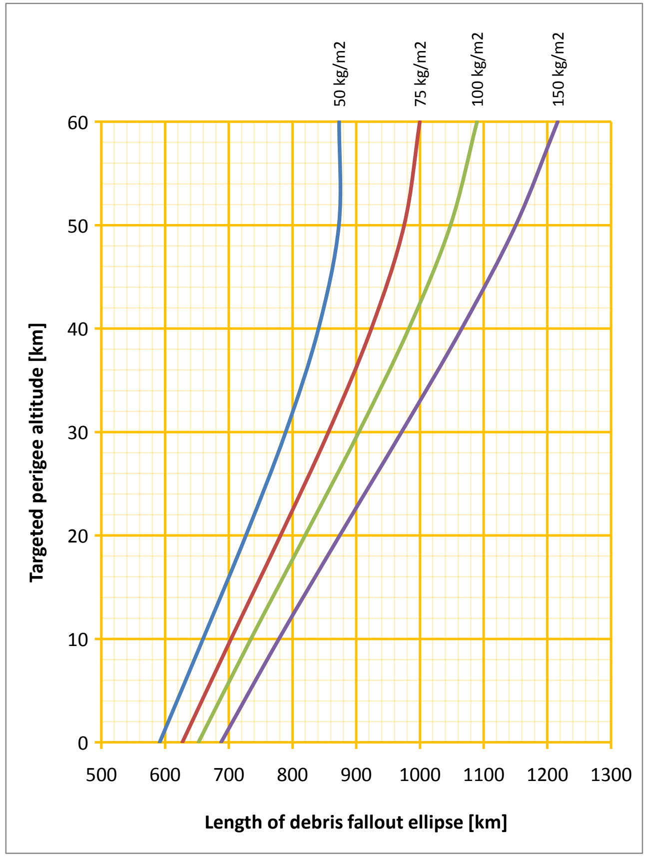
For ω = 180°/360°, the debris cloud length:

- Is dominated by the mean, with low impact from uncertainty
- Becomes asymptotic with thrust-to-mass ratio >100 N/ton
- Is mainly driven by two parameters:
  - It increases (logarithmically) with $BC_{S/C}$
  - It increases (exponentially) with the altitude of the targeted perigee

For ω = 270°, the dispersion is not dominated by the mean but there is a large impact from the uncertainty.



ω = 180° / 360°



ω = 270°

Impulse of the last manoeuvre (1/2)

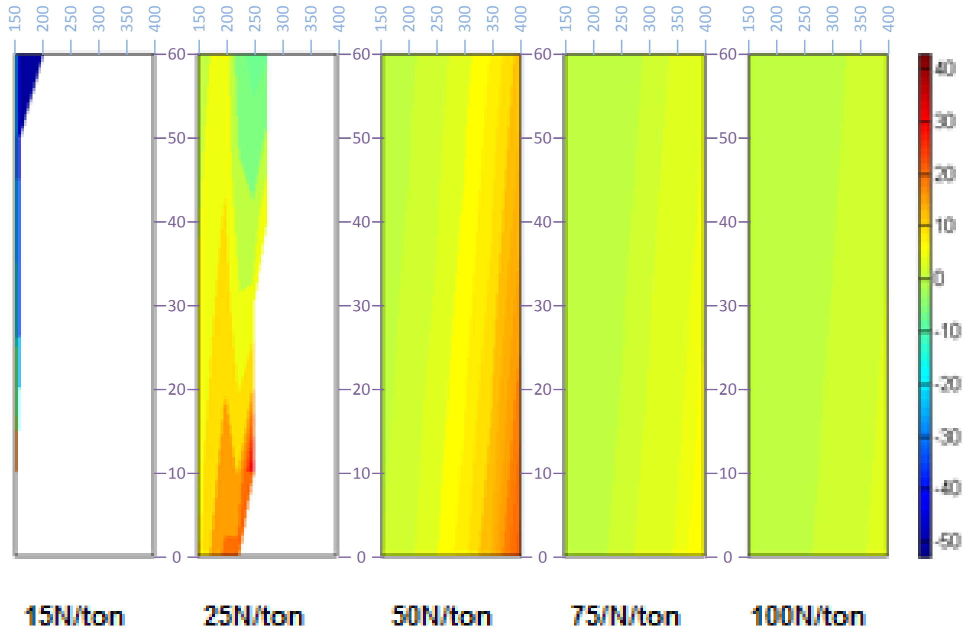
Recommended thrust authority (thrust-to-mass ratio)

- Threshold ~ 50 N/ton (acceptable for penultimate perigee below ~300-350 km)
- Goal > 75 N/ton

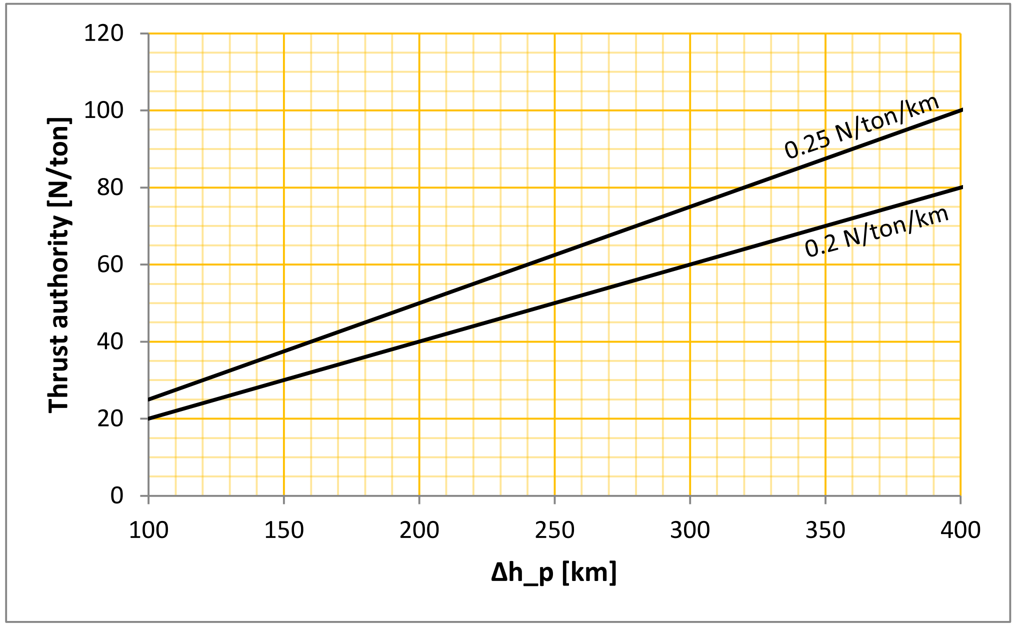
NOTE: while we make these recommendations, you will see that for the study cases, in some instances, we stretch these (by choosing a low penultimate perigee)

Recommended thrust-to-mass-to-perigee-drop ratio:

- Threshold ~ 0.2 N/ton/km
- Goal ~ 0.25 N/ton/km



Gravity losses
(% of impulsive ΔV)



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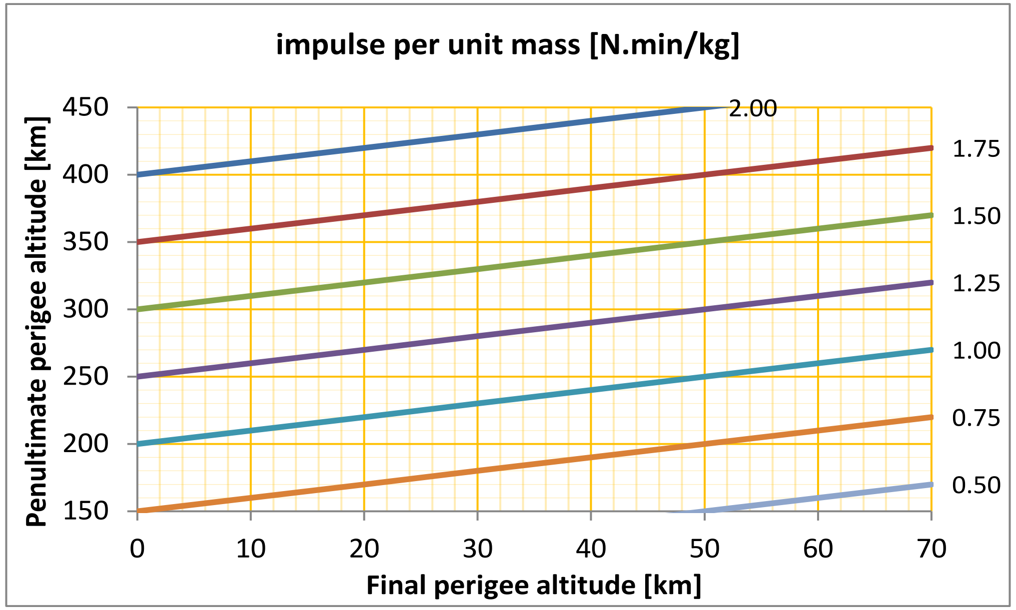
Impulse of the last manoeuvre (2/2)

Also, the perigee drop is approximately given by:

$$\Delta h_{peri}^{[km]} \cong 200 \times \frac{F_T^{[N]} \Delta t^{[min]}}{m_{sat}^{[kg]}}$$

This links the drop in perigee altitude with the duration of the burn, for a given thrust-to-mass ratio.



Accuracy of the manoeuvre
Accuracy of the position of the debris ellipse centroid

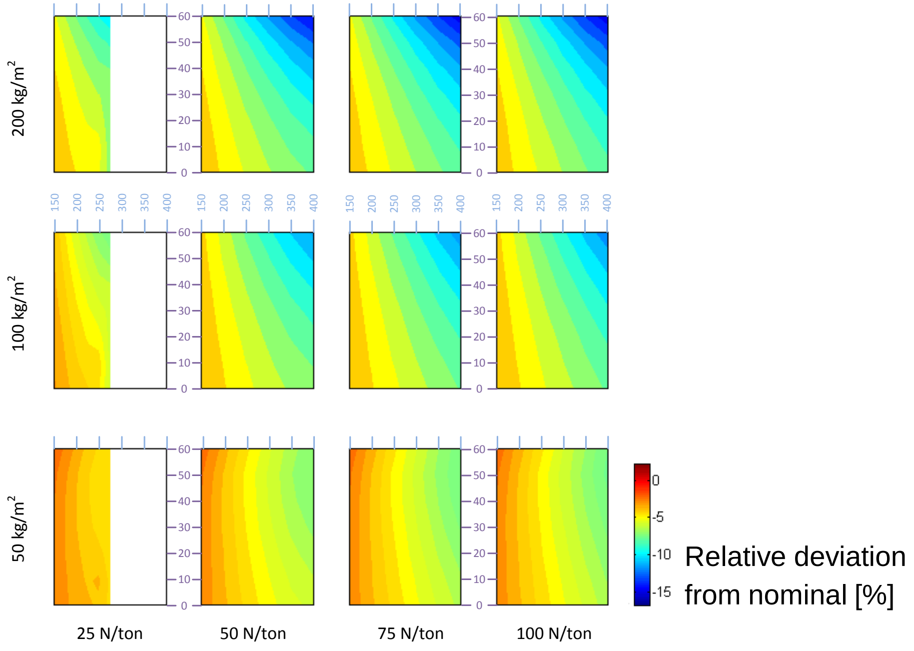
- The satellite ballistic coefficient effect is small
- Centroid position error due to a Delta-V error increases with:
  - Targeted perigee: changes the trajectory
  - Penultimate perigee: increases the required Delta-V (and thus the magnitude of the 5% error)
- For a 5% Delta-V error → the drift is close to the size of the nominal debris-cloud
- No significant differences on LoA location

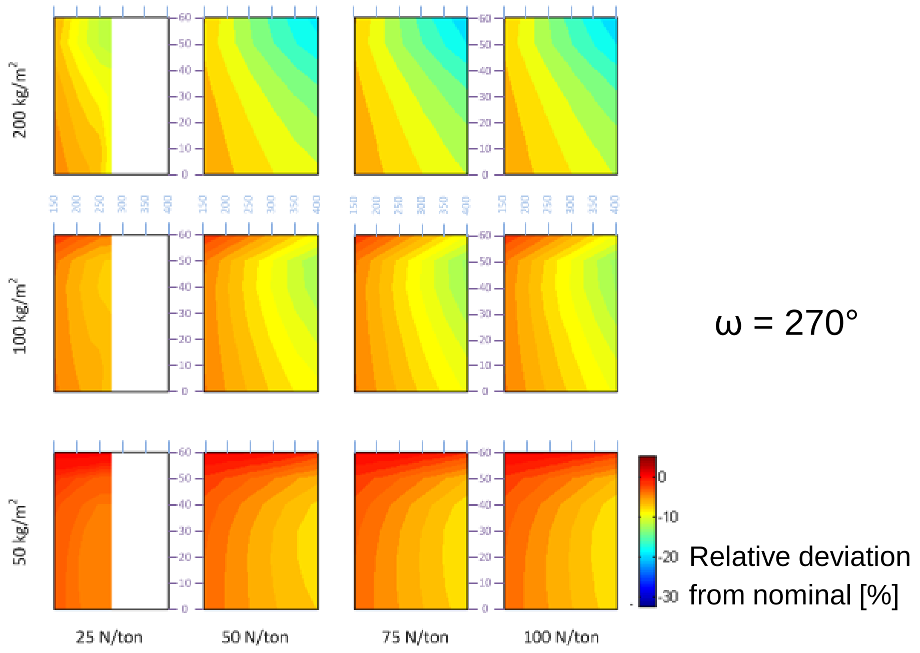
Deviation from
nominal [deg]

Accuracy of the manoeuvre debris ellipse length

- Dispersion in debris ellipse length increases with:
  - Penultimate perigee
  - Targeted perigee
  - Spacecraft ballistic coefficient

ω = 180° / 360°





ω = 270°

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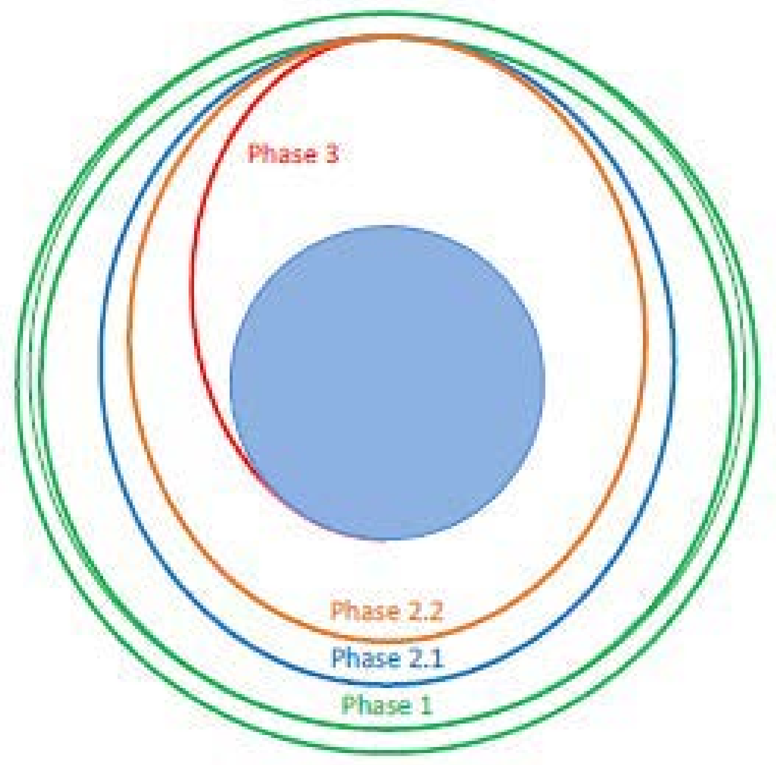
Controlled Re-entry Manoeuvre Strategy
1/ Multi-step impulsive manoeuvre strategy

The strategy of reference
- Particularly well suited for initial orbits above 500 km
- Less demanding in terms of the Delta-V magnitude of any manoeuvre compared to a direct re-entry from the nominal altitude orbit
- Suitable for regular monopropellant thrusters.

Split in three phases:
- Phase 1: Initial Hohmann transfer to lower circular orbit to free operational orbit (e.g. 20-50 km lower altitude)
  - Allows nominal S/C operation during preparation for controlled re-entry.
- Phase 2 (.1 to .n): One to several perigee lowering manoeuvers to reach pen-ultimate perigee
  - Number of manoeuvres depends on spacecraft capabilities
  - Possibility for validation of correct disposal system function and correction in case of manoeuvre errors
- Phase 3: Final re-entry boost leading to re-entry and break up in the atmosphere
  - One thruster left open to initiate tumbling motion of the spacecraft (optional)

Also interesting for:
- Keeping the duration of the overall disposal phase short
- AND with the possibility of monitoring the orbit evolution
  - Apply corrections where necessary



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Controlled Re-entry Manoeuvre Strategy
2/ Combined low-thrust + impulsive manoeuvre strategy

Again, three phases:

1. clear the operational orbit if required.
   it may be better / faster to perform this manoeuvre via a Hohmann transfer
2. Using low-thrust, high-specific impulse propulsion systems, the second step can be achieved in two ways:
   1. Reducing the perigee only, as per the previous strategy;
   2. Continuous firing, dropping into a circular orbit down to the penultimate “perigee”.

Once the penultimate perigee altitude is reached, the final boost leading to re-entry is performed with a high-thrust propulsion system.

	Option 1	Option 2	Comments
Propellant usage (I_{sp} = 500 s)	2.2%	4.5%	Of dry mass
Phase 2 duration	4-8 weeks	2-5 weeks	Subject to thrust-to-mass ratio, and thrust duty cycle per orbit (for Option 1)

Option 2 is:

- Faster
- May be limited by the electrical energy available in each orbit;
- Makes it more demanding on the AOCS, especially as the altitude gets smaller;
- Has a small penalty (~2-3%) on the final de-orbit Delta-V prior to re-entry because of the lower apogee.

➔ Option 1 is preferable

Propulsion Systems

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Propulsion Systems
Re-cap of requirements

Recommended thrust-to-mass ratio per unit perigee altitude drop:

- Preferred: 0.25 N/ton/km
- Acceptable: 0.2 N/ton/km

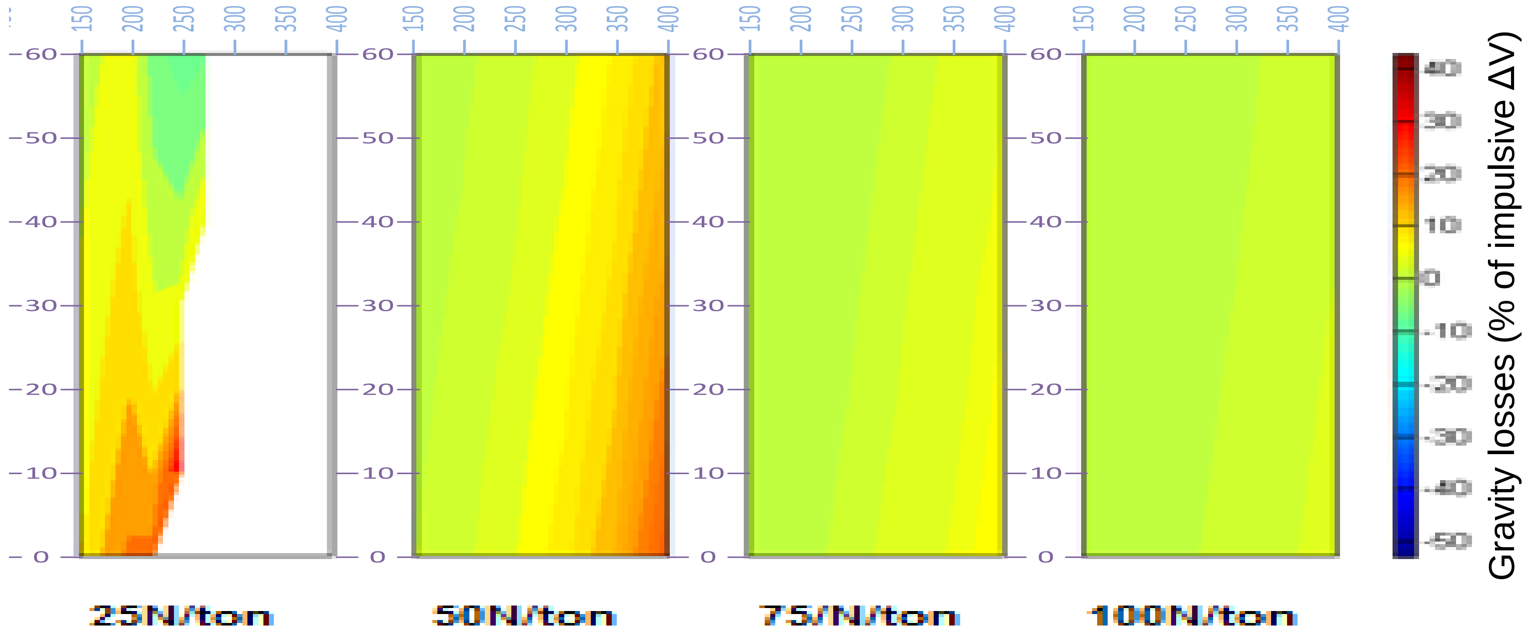
manoeuvre duration:

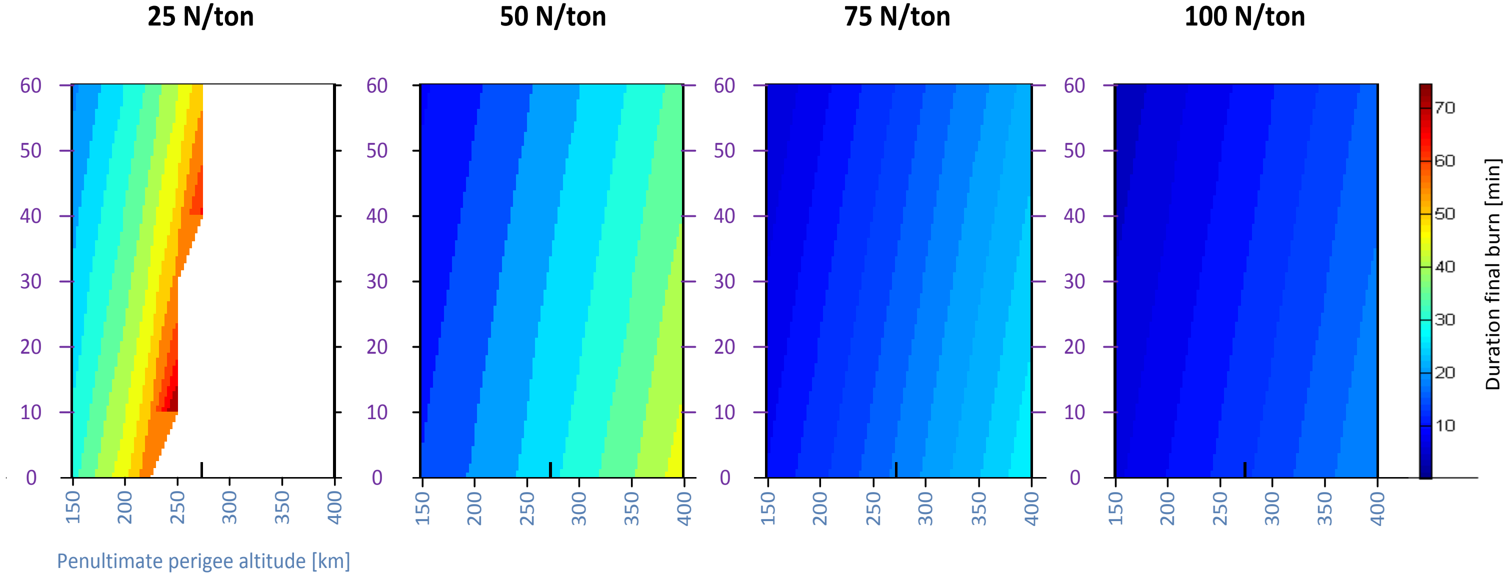
A value of 20 min is generally used.

To minimise gravity losses:

- Preferred: 20 minutes
- Acceptable: 30 minutes
  - Gravity losses <10% approximately
- Stretched: 40 minutes
  - Gravity losses of ~20%

However, what is acceptable in terms of gravity losses is very much subjective!





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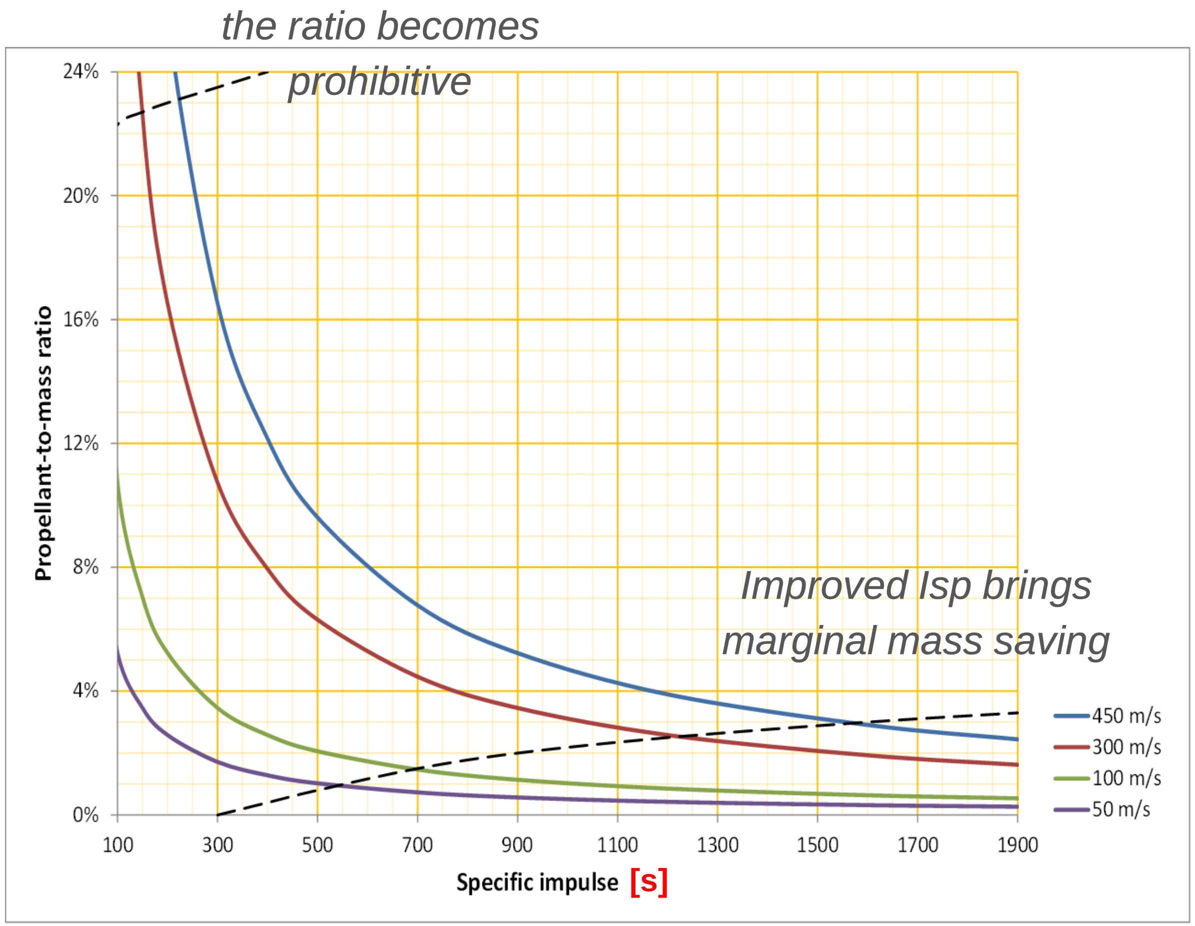
Propulsion Systems
Drivers and Constraints

- The launcher performance will dictate the maximum wet mass of a satellite
  - Thus how much propellant can be embarked by the satellite, for a given dry mass.
  - The propellant-to-dry-mass ratio is highly dependent on:
    - specific impulse
    - total Delta-V required

The advantage of targeting propulsion technologies with higher specific impulses is centred on the “elbows” of the curves, resulting in potentially essential mass savings (depending on the dry mass of the satellite).

The dotted lines are indicative.



Propulsion Systems
Drivers and Constraints (ctng)

Thruster throughput

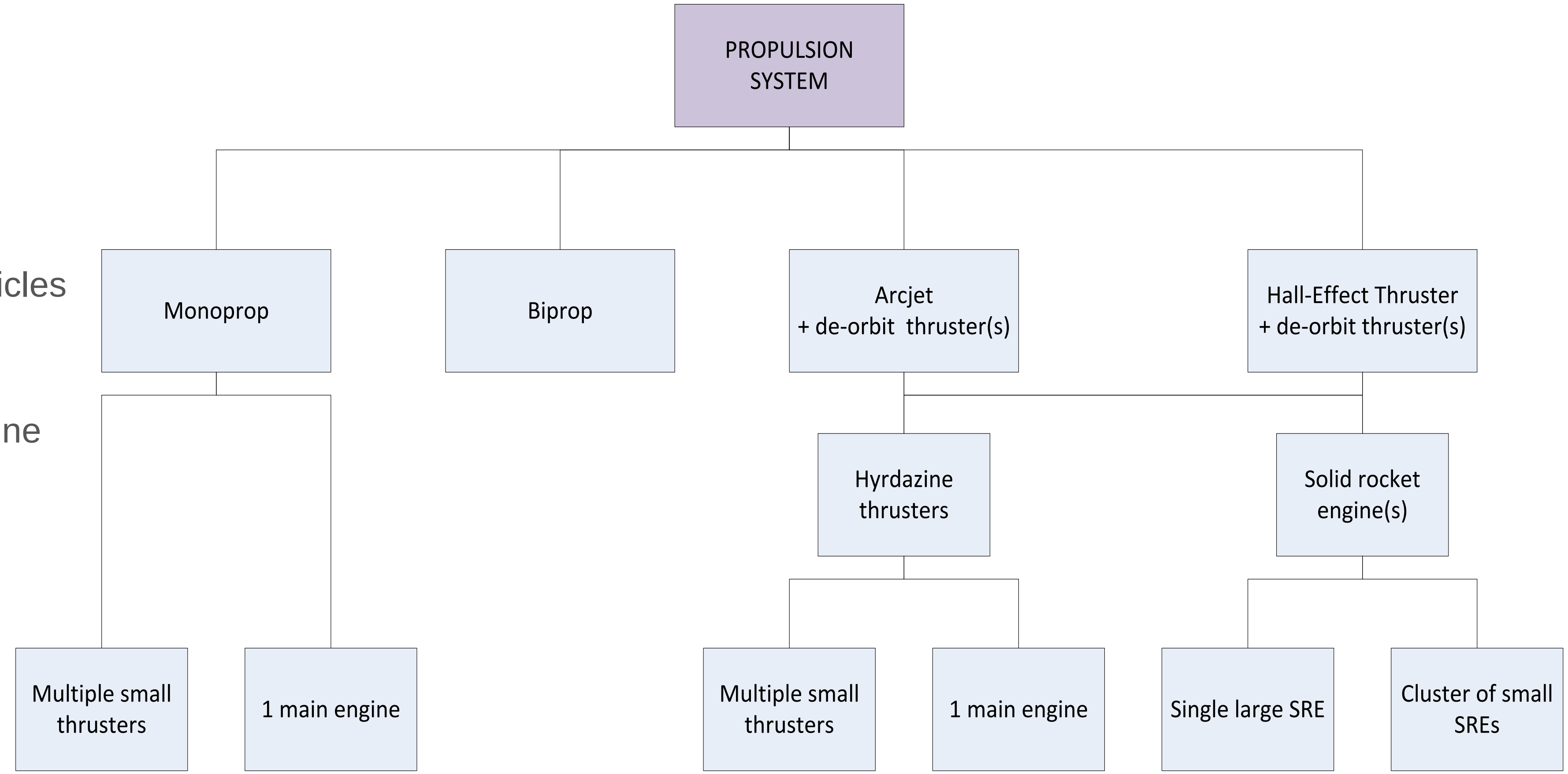
- Should be compatible with the mission architecture.
- Can be a limiting factor (rarely but TBC)

Acceleration

- A maximum of 0.04 g is often quoted for appendages, in particular for the links between a solar array drive mechanism and the spoke.
- For monopropellant systems, this is not an issue
  – the maximum thrust level under consideration is 400 N
  – the satellite mass starting at 1000 kg.
- This can be more of a constraint for solid rocket propulsion
  – Very high thrust over a short time

Propulsion Systems
Candidate Systems

- Monopropellant hydrazine
  - Widely used on LEO satellites
- Biprop
  - 50% improvement on Isp
  - But heavier and complex
  - Only interesting for very large vehicles and/or very large Delta-V budgets
- Arcjets
  - Low thrust, higher Isp with hydrazine
- Plasma thrusters
  - Now being implemented for LEO constellations
- Solid Rocket Motors
  - One-off, high-thrust motors



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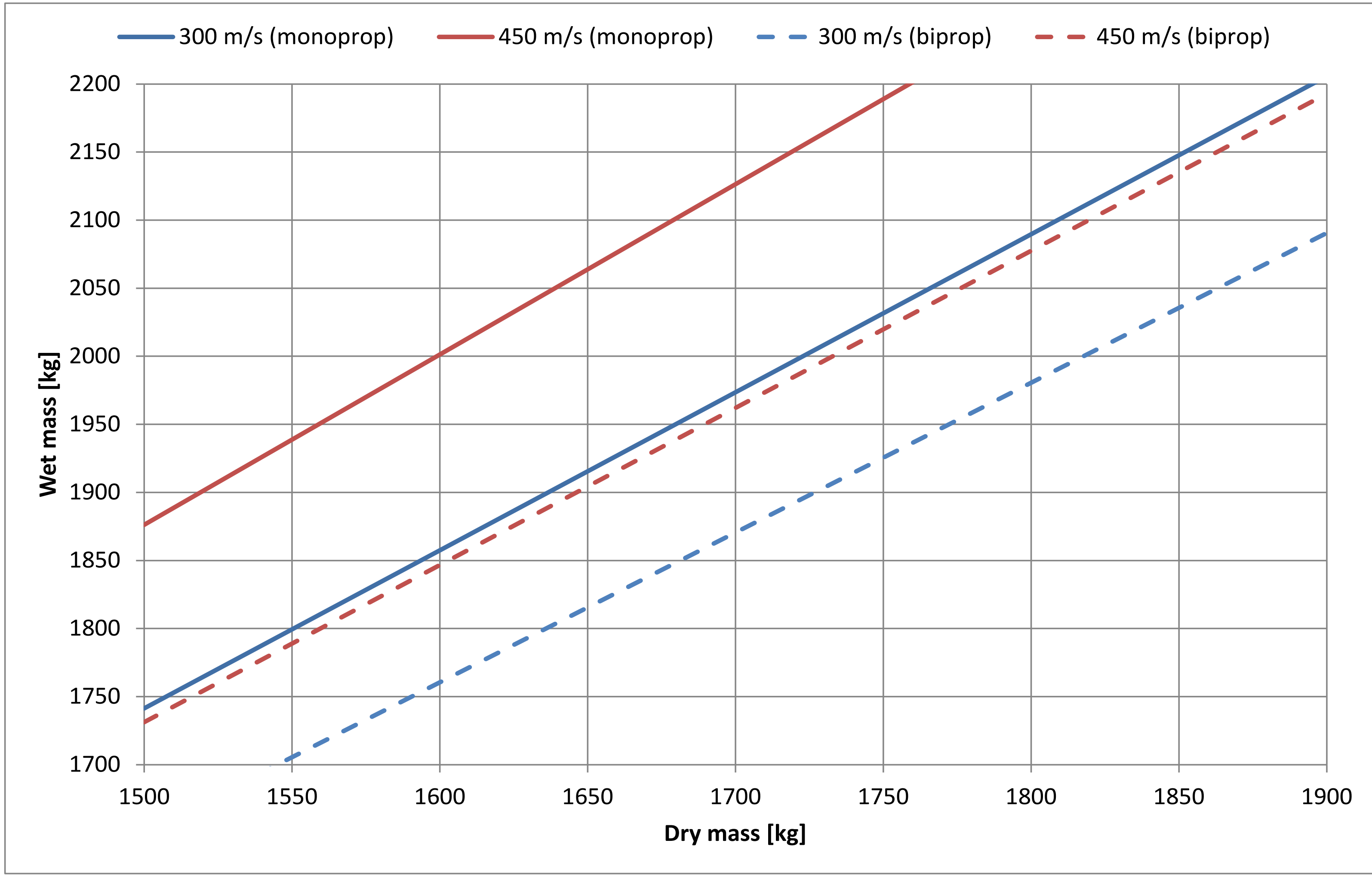
Propulsion Systems
Reason for not considering bi-prop

- Inherently more complex than monoprop
- For LEO mission, typical Delta-V budgets make the propellant mass saving small
- While bi-prop could enable missions at the limit of a launcher capability, it is a small niche
  - Factoring in the extra H/W mass, the saving can disappear quickly.



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Propulsion Systems
Monopropellant Hydrazine Systems

European thrusters available:

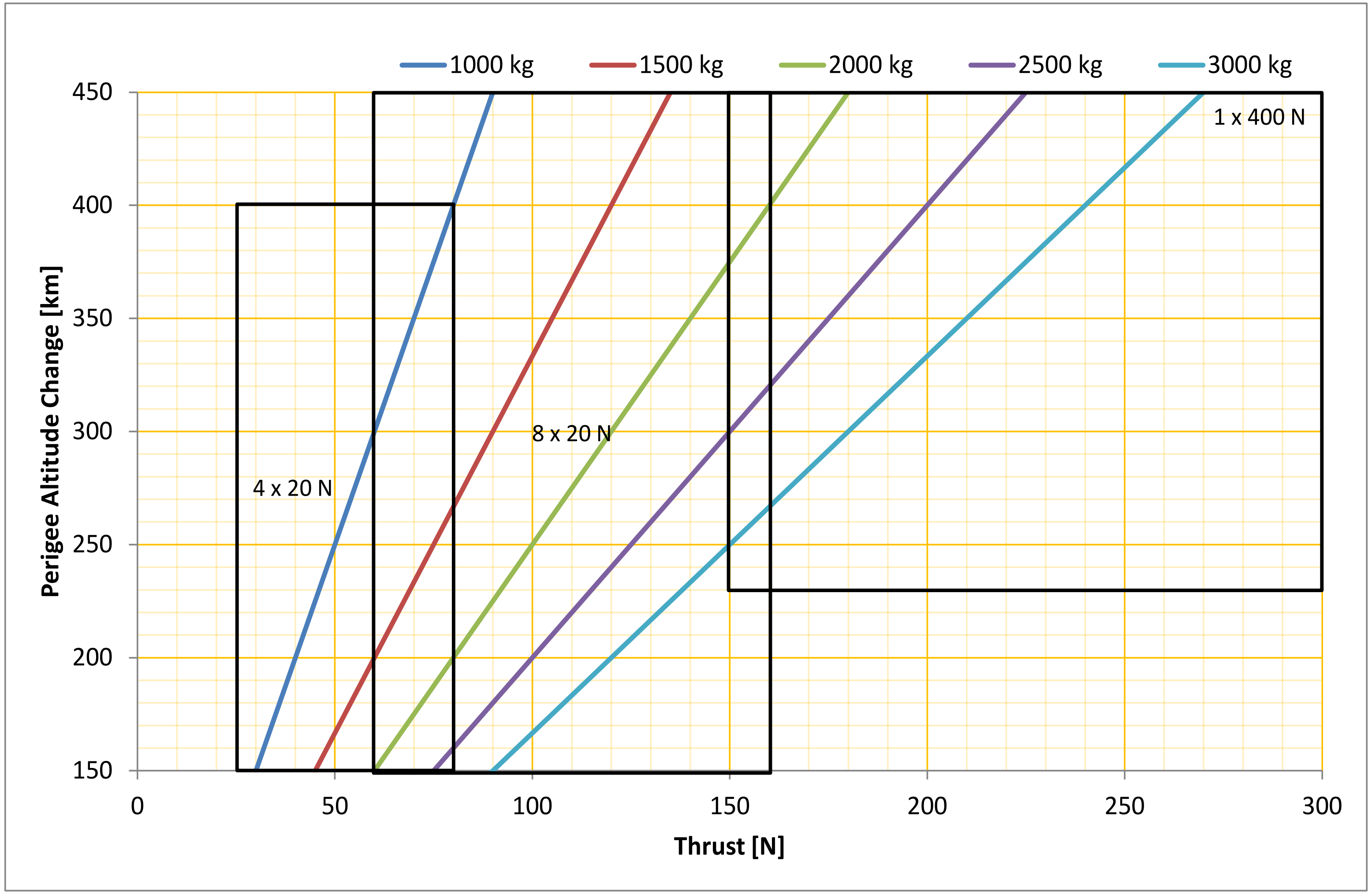
- 1 N and 5 N (not for re-entry Delta-V)
- 20 N, 400 N

Note: US thruster at 200 N

Candidate architectures

Three “zones” to be covered:

- Lower end: 4 x 20-N
- Upper end: 1 x 400-N
- Middle:
  - – 8 x 20-N (European)
  - – 1 x 200-N (non-European)
  - – Can still be covered by 1 x 400-N



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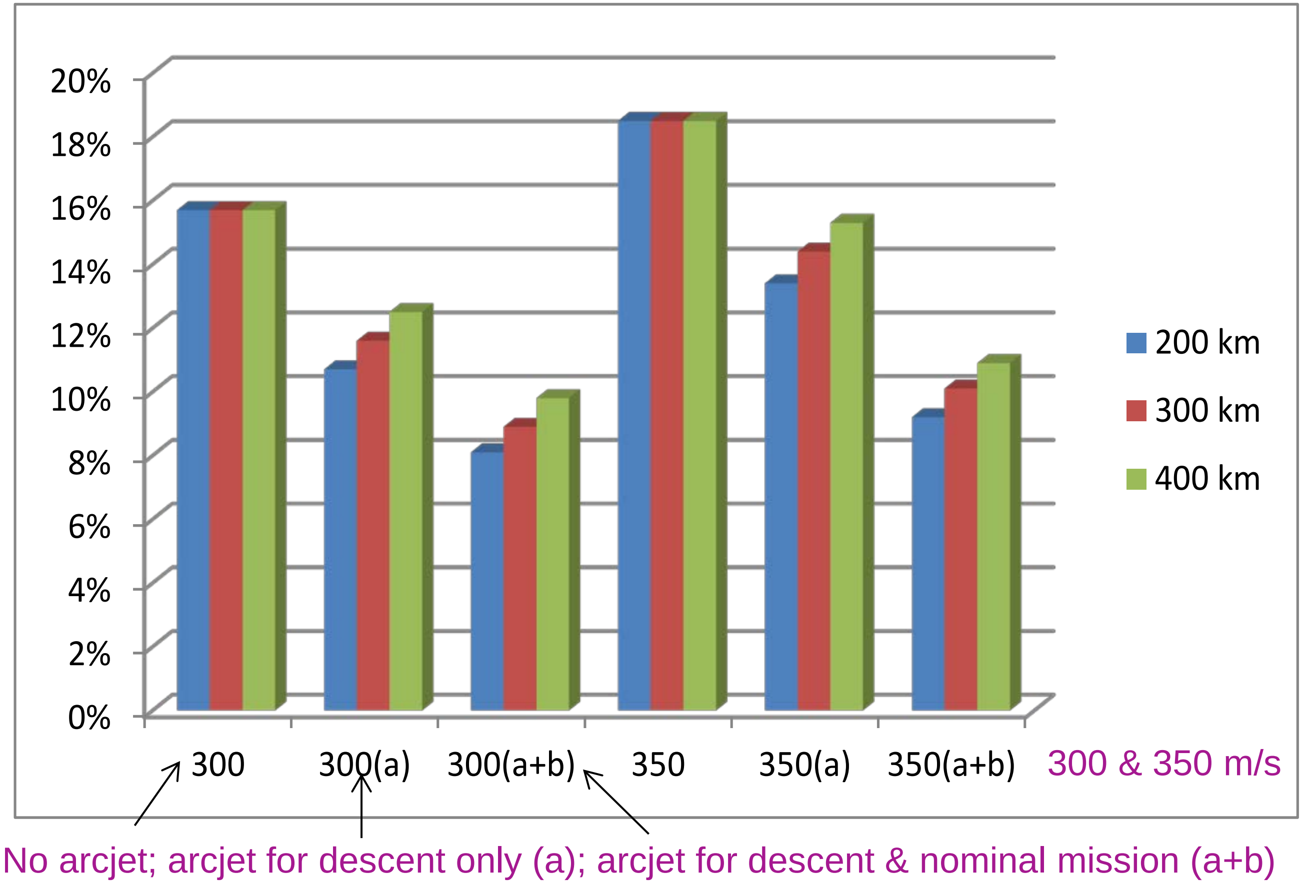
Propulsion Systems
Preliminary evaluation of arcjet-based systems

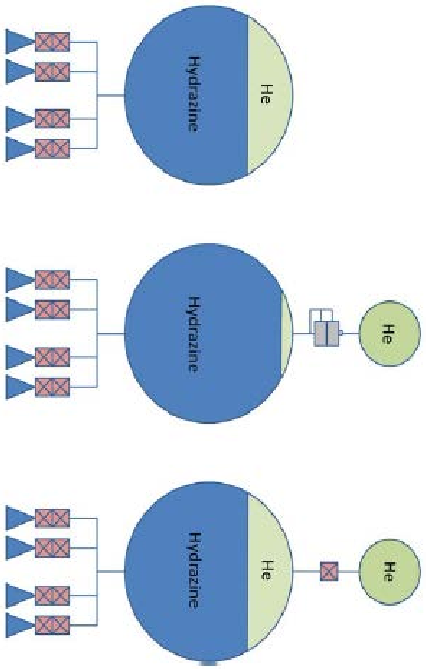
Arcjets are interesting because of the higher Isp.

To make the most of it, the proportion of the mission using arcjets need to be maximised
- Nominal mission
- Descent down to 200 km

A propellant mass saving equivalent to 10% of the dry mass is possible



Propulsion Systems
Pressurisation Concepts



	Pros	Cons	Comments
Blow-down	Simplicity / reliability	EOL loss of performance (thrust, Isp)	Best for simplicity and small Delta-V budgets
Regulated	Thruster performance	Complexity Lifetime / reliability	Good for short missions (monoprop). Necessary for electric propulsion systems such as arcjets.
Re-pressurisation	Alternative to Regulated systems (monoprop only).	Complexity / reliability	Consider lifetime of pyrovalves.

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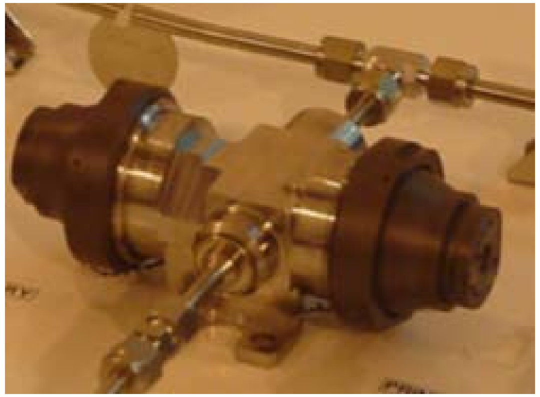
Propulsion Systems
Re-pressurization vs. Blow Down: General Recommendations (1/2)

Pressure Control leads to complex design

- Additional tank(s) for pressurant
- Additional Fill & Vent Valves and Test Ports
- PCA (Pressure Control Assy) relying on mech. regulator
  - SOA but still the propulsion component with the lowest reliability figures
  - Only one USA company exists with stable production (Standford-μ)
  - Requires bespoken leakage control (isolation if not used for long time)

Electronic Regulator looks promising but

- Design, process & procedure not consolidated
- Mass (and complexity) increased due to additional electronics



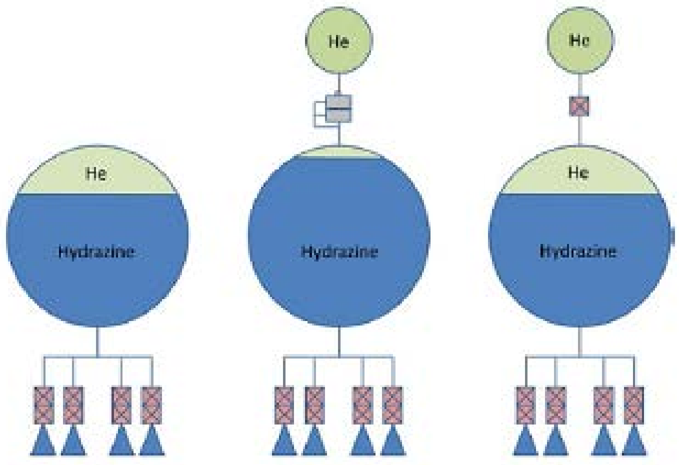
Propulsion Systems
Re-pressurization vs. Blow Down: General Recommendations (2/2)

Re-pressurization systems present same challenges as previously but

- Limited service time → leakage control less severe?
- Used after several years in flight → functionality at risk after long non-ops!
- Re-pressurization allows to operate the RCT at low inlet pressure during nominal
  - Reduced MIB for optimum AOCS

Unless large gain in mass, a simpler system is to be preferred

- Usually, gain of 20% w.r.t. S/C mass is required to switch from blow-down (mono) to regulated (bi)



AOCS Requirements

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AOCS Requirements for the Disposal Phase

During the de-orbit operations, the AOCS has to:

- Perform yaw slews before and after orbit control manoeuvres
- Stabilise the satellite during orbit control manoeuvres
- Stabilise the satellite in a specific attitude between orbit control manoeuvres
- Provide a safe mode for as long as possible

The yaw slews ahead of the orbit control manoeuvres can be performed by reaction wheels or thrusters depending on the mission and satellite design

Changes and potential challenges introduced by the disposal phase:

- With a 400 N thruster, requiring on-modulation of the attitude thrusters for balancing the disturbance torques due to thrust misalignment and CoM offset.
  - The high disturbance torques encountered drive the torque capacity needed and therefore the thruster configuration.
- At lower altitudes, aerodynamic disturbance torques may exceed the torque and momentum capacity of the wheels
  - Need for attitude thrusters in on-modulation for stabilisation between Delta-V manoeuvres.
- The use of the nominal safe mode may be prohibited by:
  - High aerodynamic disturbance torques
  - Potential limitations on the use of attitude sensor (low altitude).

    ➔ Inhibit the safe mode after a low perigee altitude has been crossed.

RCS configuration and disturbance torques

Main Engine plus attitude thrusters

The sizing of the RCS depends on:

- main engine's operative force level
- satellite geometry
- performance requirements of possible additional orbital manoeuvers:
  - Minimum resolution to fulfil the minimum ΔV
  - Minimum efficiency to fulfil the numbers of manoeuvers in the mass budget

On MetOp-SG, a branch of 4 thrusters canted at 15deg is employed for:

- controlled re-entry,
- orbital manoeuvers,
- safe-mode.

Four 20-N thrusters

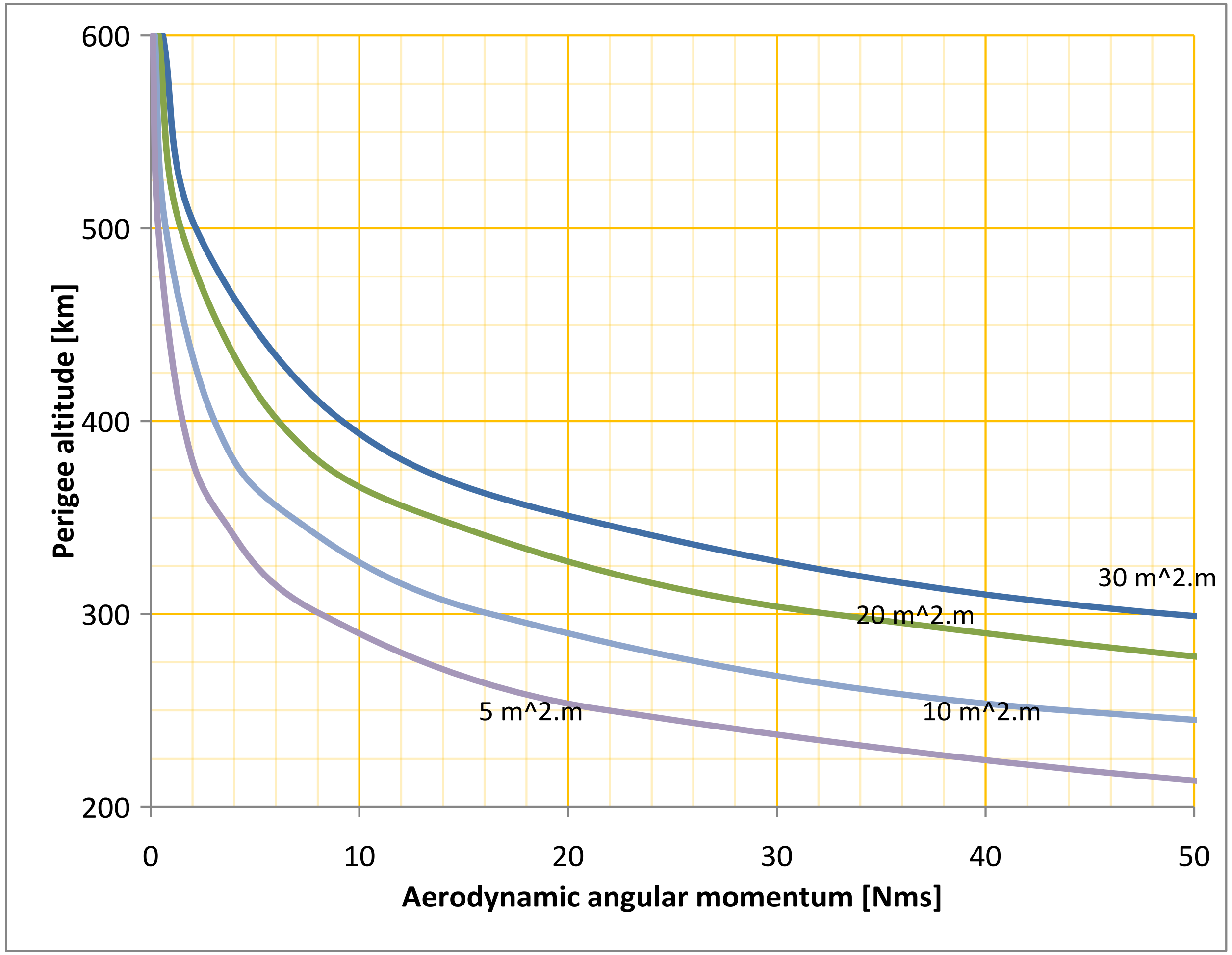
If the authority of an RCS is sufficient to provide the controlled re-entry burn demand, a small cant angle in both directions is sufficient to counteract the disturbance torques, without penalising the propellant budget.

With a minimum canting of 3°, the rejection of the disturbances is fulfilled.

Effect of the penultimate perigee altitude

Typical reaction wheels on LEO satellites can deliver angular momentum control in the range 20 to 70 Nms,

- Assuming:
  - 20 Nms
  - about half of that is dedicated to absorbing the aerodynamic drag torque
- the penultimate perigee for a large A x (CoG-CoP) should be above 400 km.
  - This avoids the need for extensive additional verifications of the AOCS during the satellite development.
- For smaller satellites, the satellite could cope with the aerodynamic drag environment around 300 to 350 km.
- As with the semi-controlled re-entry, one could design the descent with an even lower perigee by choosing a so-called neutral attitude, to avoid saturating the wheels



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Overview of the 3 Study Cases

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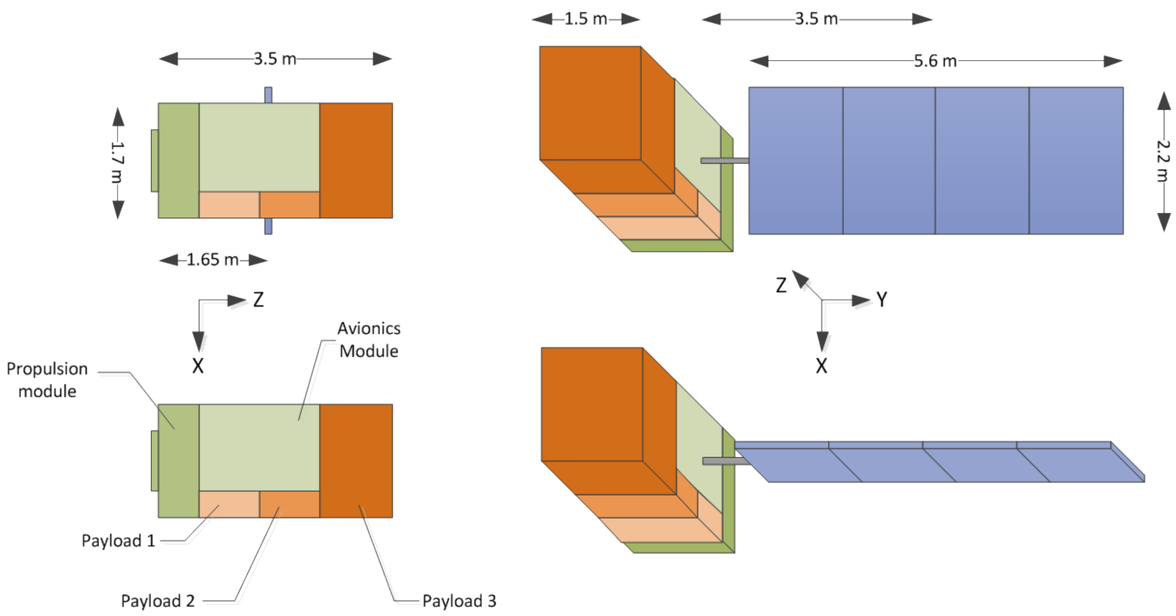
Case 1: 1000-kg satellite

Parameter	Value
Mass	1000 kg
Envelope (w/o solar arrays)	1.8 x 1.6 x 3.6 m
Inertia	[1100, 1100, 600] kg.m^2
Centre of Mass	[0, 0, 1.2] m
Centre of Pressure	[0, 0, 1.7] m

Case 2: 1500-kg satellite
Case 3: 2500-kg satellite

Parameter	Value
Mass	1500 kg
Envelope (w/o solar arrays)	1.7 x 1.5 x 3.5 m
Inertia	[2200, 1400, 1450] kg.m^2
Centre of Mass	[0, 0.1, 1.7] m
CoM-CoP along Y	0.1 – 3.6 m (S.A. dependent)

Parameter	Value
Mass	2500 kg
Envelope (w/o solar arrays)	1.7 x 1.5 x 4 m
Inertia	[5200, 2800, 3500] kg.m^2
Centre of Mass	[0, 0.2, 2.1] m
CoM-CoP along Y	0.2 – 4.9 m (S.A. dependent)



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Mission Delta-V budget

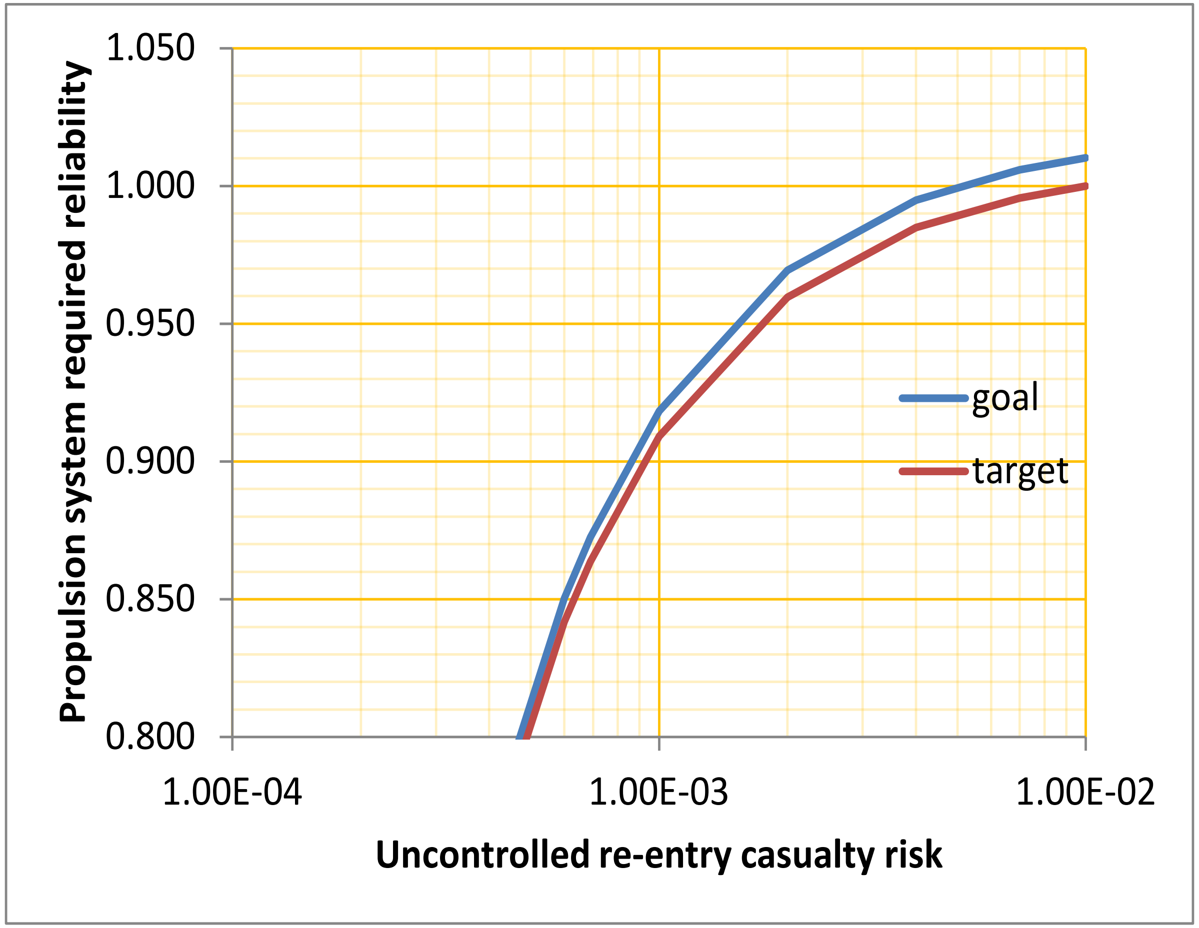
All cases assume a Delta-V budget of 300 m/s. The Delta-V budget is driven by:

- The disposal Delta-V;
- Inclination control (out-of-plane manoeuvres) in the order of 15 m/s every 2-3 years;
- Launcher injection correction ~20 m/s

Contribution	Delta-V (m/s)	Comment
Launcher injection corrections	20	
Orbit maintenance		
Inclination & MLST	30	Mission of 7-10 years
Semi-major axis & eccentricity	12	Mission of 7-10 years
Collision avoidance	8	Mission of 7-10 years
Disposal	230	800 km to 30 km
Mission Total	300	

Reliability Assessment

Subsystem	Typical figures (pessimistic)	Worst-case allocation
Data Handling	0.995	0.990
Power	0.999	0.995
AOCS	0.998	0.995
PDHT (switched off)	1	1
Propulsion	0.970	0.950
SATELLITE	0.962	0.931
Permissible casualty risk of uncontrolled re-entry	2.65×10^{-3}	1.45×10^{-3}
Equivalent casualty risk area (based on a limit of 7.2 m^2)	~190 m^2	~104 m^2



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Monopropellant Hydrazine Systems

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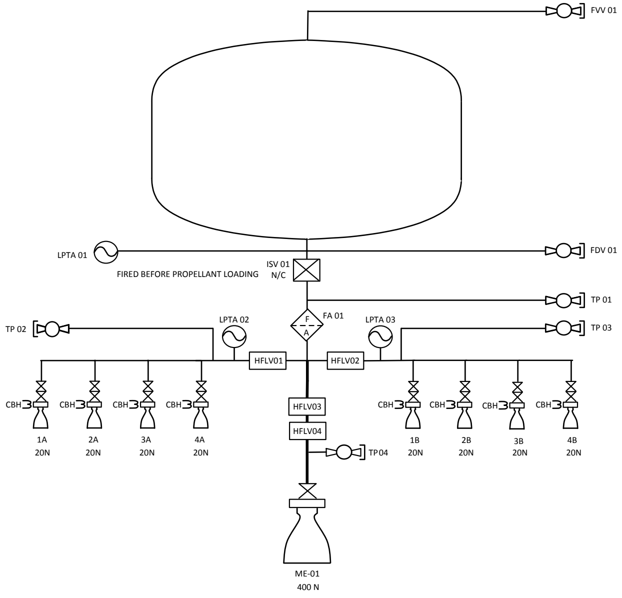
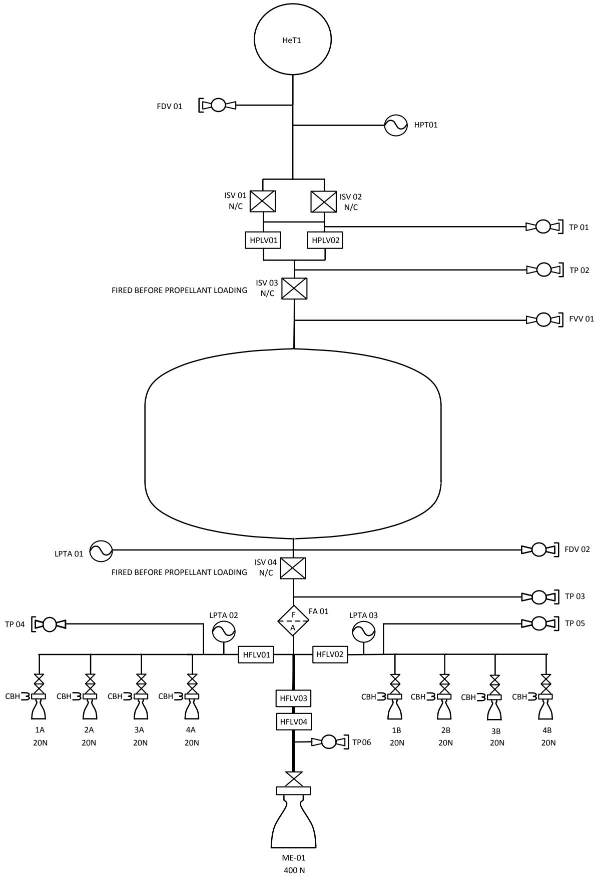
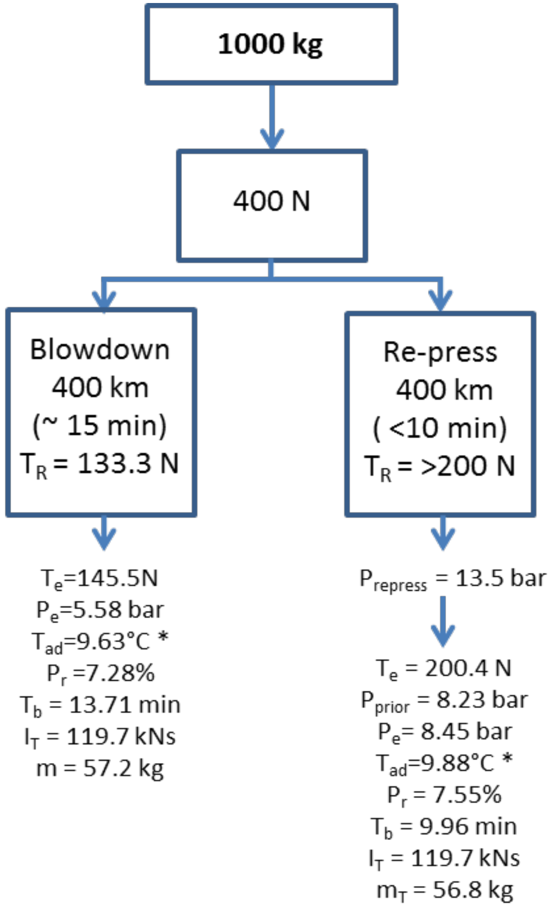
Architecture: 1 x 400-N + 4 x 20-N (x2)





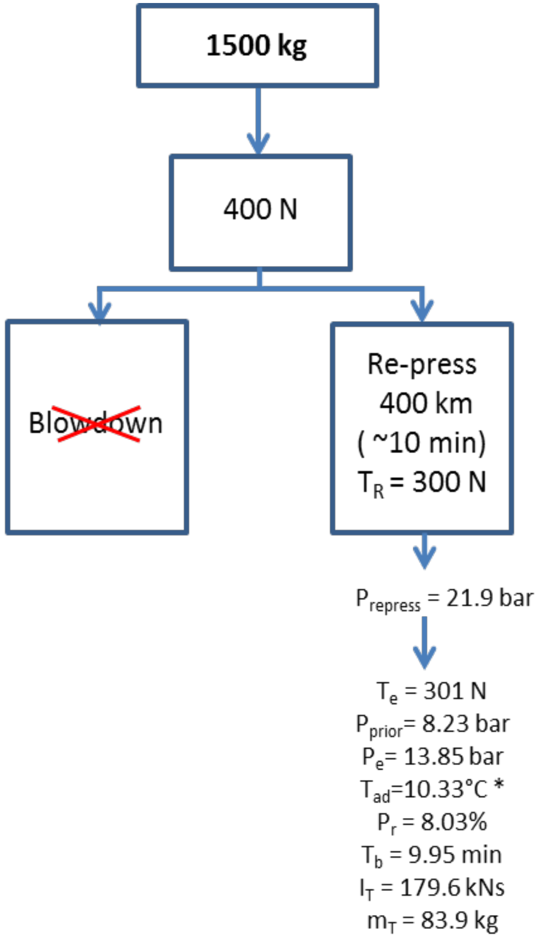
Architecture: 1 x 400-N + 4 x 20-N (x2)
Case 1



T_R = Total thrust required in sp/c axis during CRE Phase 3
T_e = Total thrust in sp/c axis during CRE Phase 3
P_{repres} = Re-pressurisation shot
P_{prior} = Tank pressure prior to re-pressurisation
P_e = Tank pressure @ end CRE Phase 3
T_{ad} = Min pressurant adiabatic temperature CRE Phase 3
P_r = End propellant residual
T_b = Duration of last burn
I_T = Max total impulse (per thruster in thruster axis)
m_T = Max total throughput (per thruster)

* Uses Nitrogen pressurant to achieve +ve adiabatic temperatures (gamma = 1.4)

Architecture: 1 x 400-N + 4 x 20-N (x2)
Case 2



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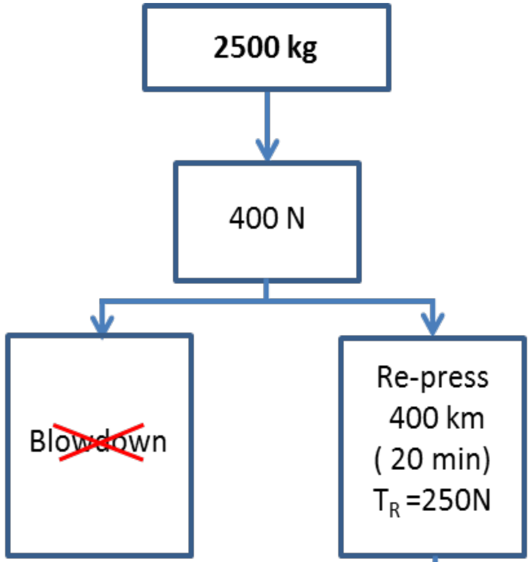
Architecture: 1 x 400-N + 4 x 20-N (x2)
Case 3



$P_{repress}$ = 17.7 bar

T_e = 250.6 N
P_{prior}= 8.23 bar
P_e= 11.16 bar
T_{ad}=10.11°C *
P_r = 7.79%
T_b = 19.9 min
I_T = 299.3 kNs
m_T = 140.9 kg

T_R = Total thrust required in sp/c axis during CRE Phase 3
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I_T = Max total impulse (per thruster in thruster axis)
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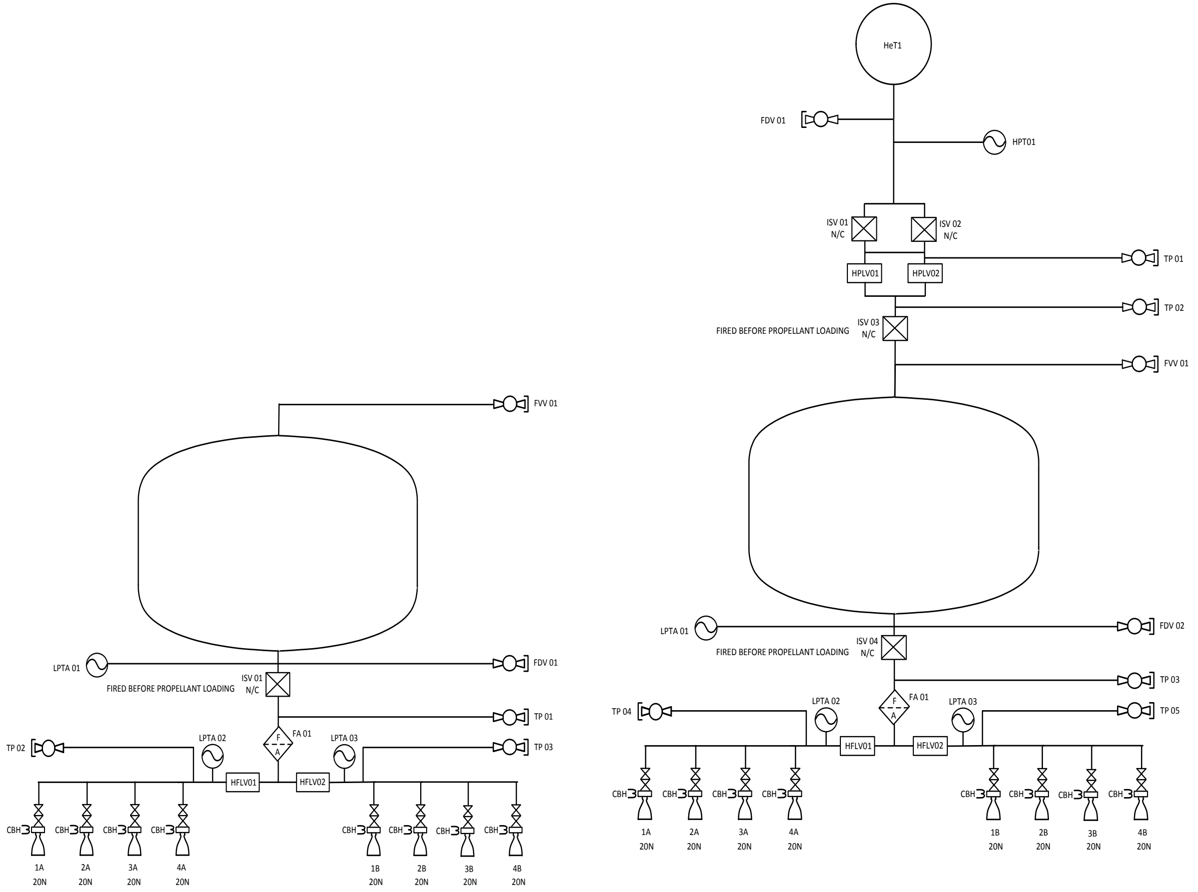
* Uses Nitrogen pressurant to achieve +ve adiabatic temperatures (gamma = 1.4)

Architectures
4 x 20-N (x2)



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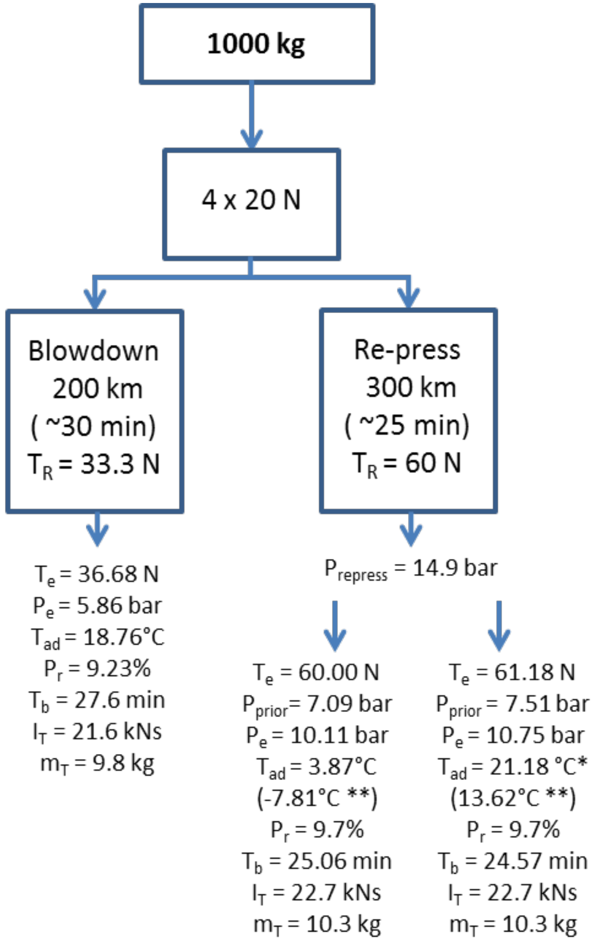
Architecture: 1 x 400-N + 4 x 20-N (x2)
Case 1



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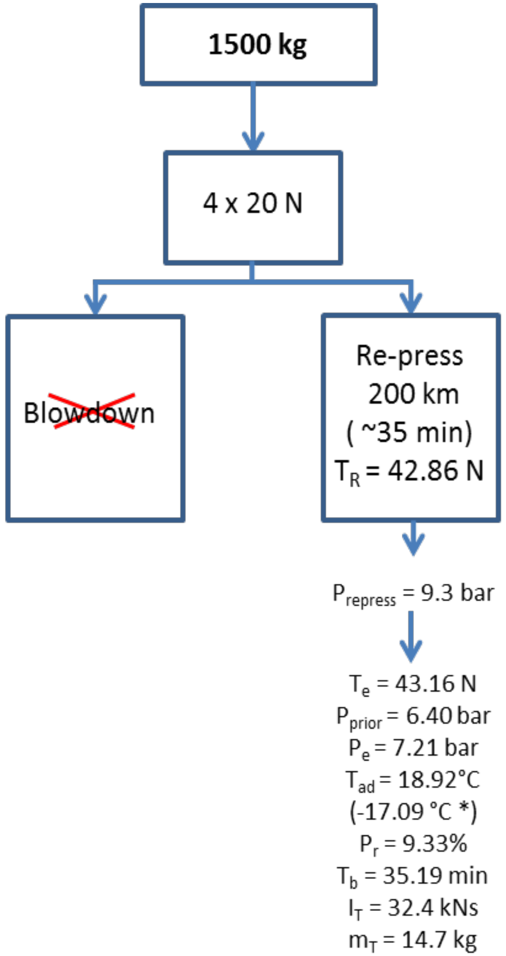
* Uses Nitrogen pressurant to achieve +ve adiabatic temperatures (gamma = 1.4)
** Adiabatic temperature during CRE phase 1

Architecture: 1 x 400-N + 4 x 20-N (x2)
Case 2



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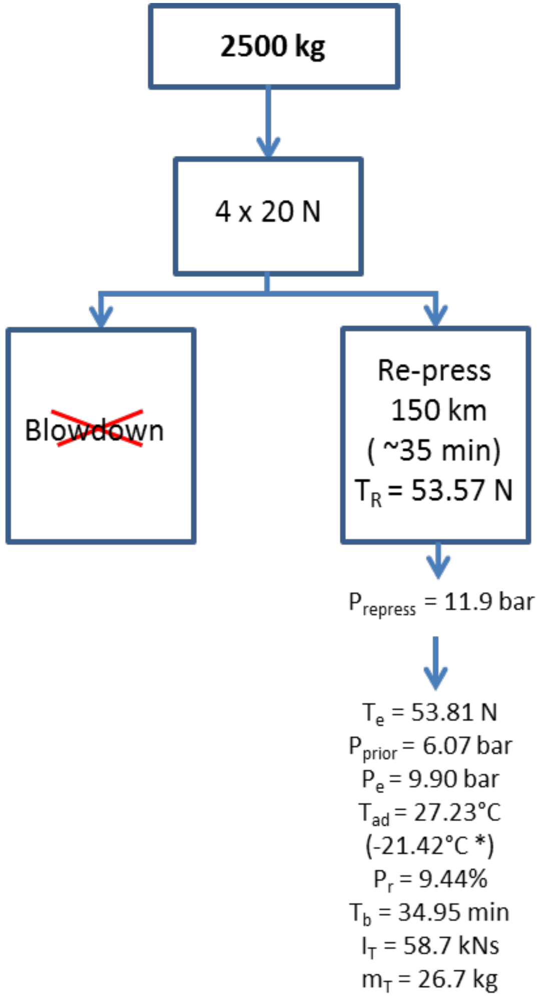
* Adiabatic temperature during CRE phase 1

Architecture: 1 x 400-N + 4 x 20-N (x2)
Case 3



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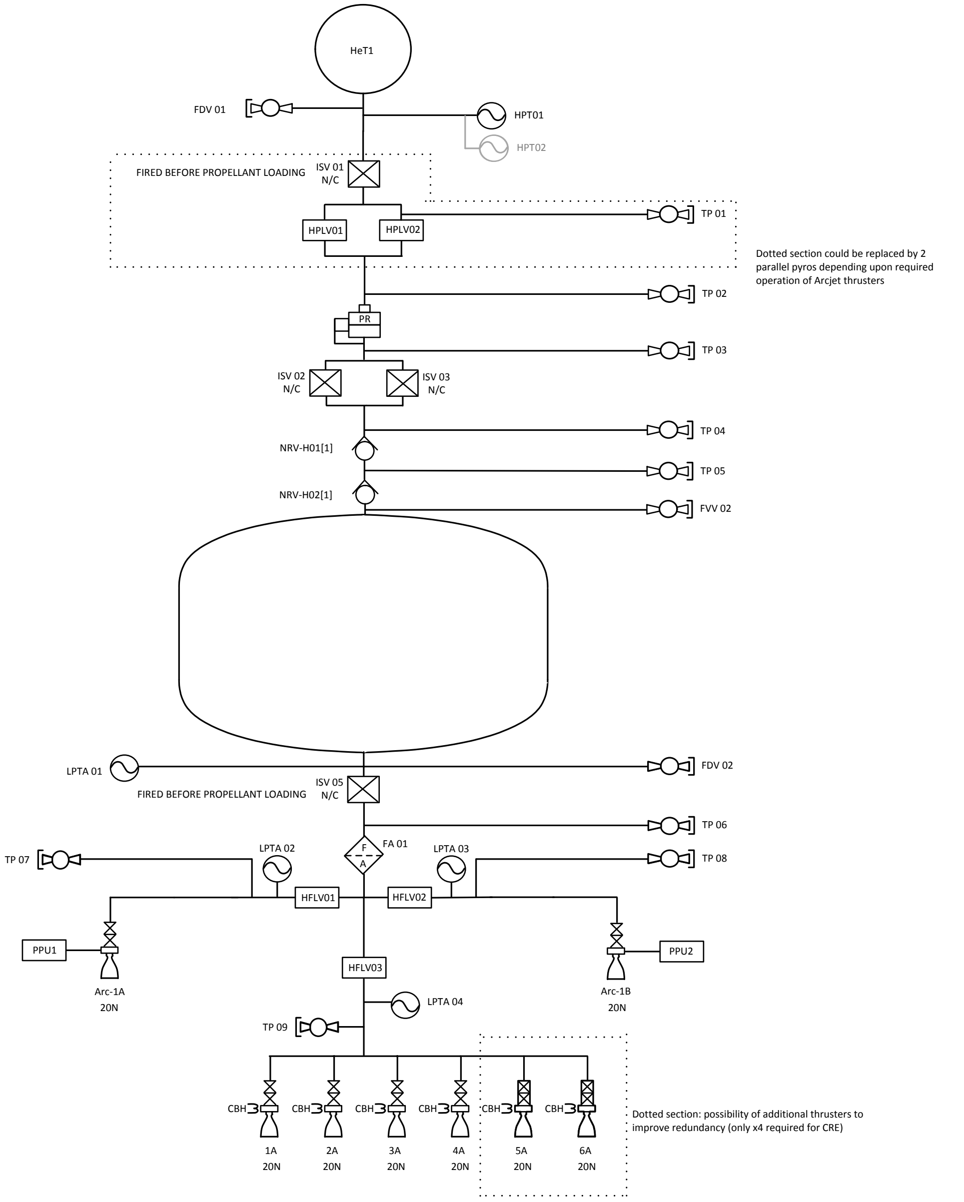
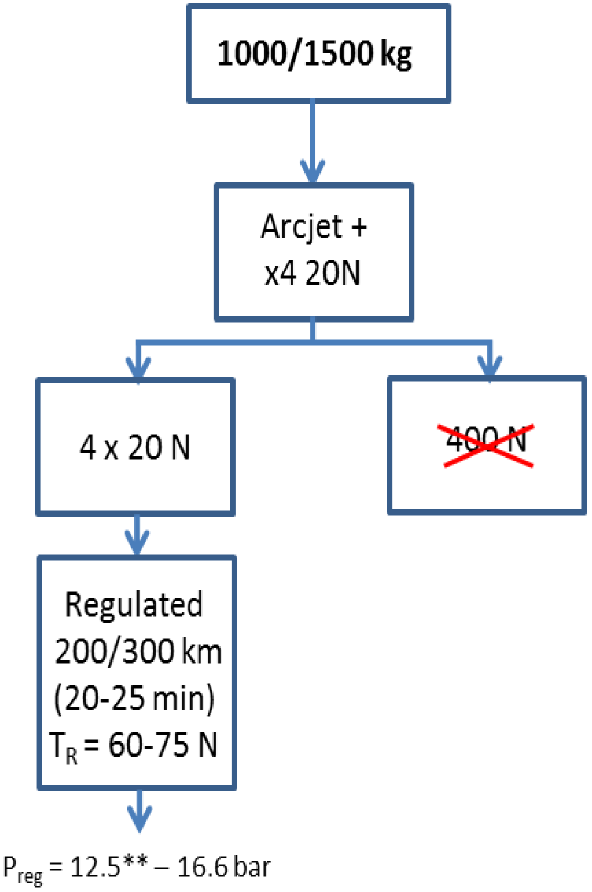
* Adiabatic temperature during CRE phase 1

Architecture: Arcjet (x2) + 4 x 20-N



Architecture: Arcjet (x2) + 4 x 20-N
Cases 1&2



T_R = Total thrust required in sp/c axis during CRE Phase 3
P_{reg} = Required pressure level setting of mechanical regulator

** excess thrust would be generated as min pressure by Arcjet mechanical regulator : 13.8

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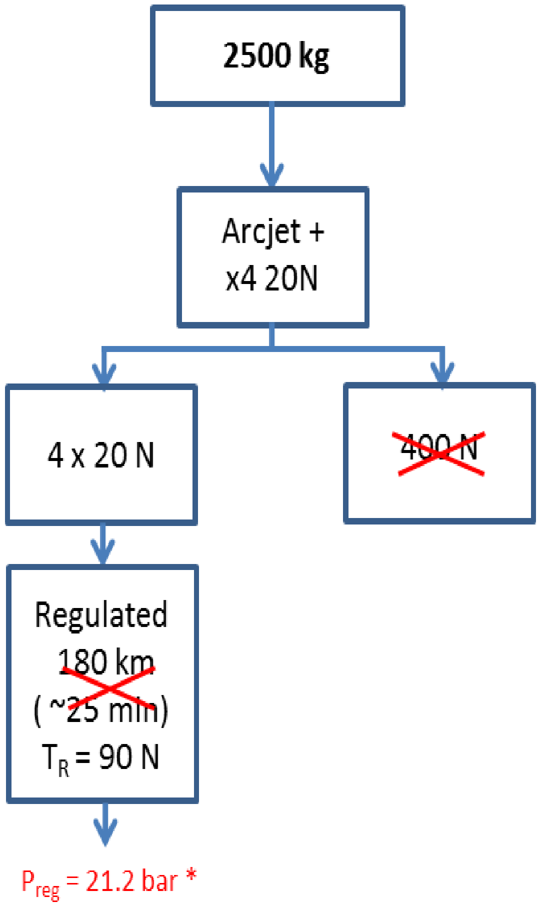
Architecture: Arcjet (x2) + 4 x 20-N
Case 3

Note:

- increasing the burn duration to 30-min should make the 4x20-N design possible
- Alternatively, increase to 8x20-N thrusters



T_R = Total thrust required in sp/c axis during CRE Phase 3
P_{reg} = Required pressure level setting of mechanical regulator

- Arcjet has maximum inlet pressure: 18.6 bar

Solid Rocket Motors

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SRM Classes

Two preliminary sizing scenarios:

- Scenario 1:
  - Penultimate perigee at 430 km altitude
  - Final Delta-V = 115 m/s (perigee reduction of 400 km)
- Scenario 2:
  - Penultimate perigee at 180 km altitude
  - Final Delta-V = 44 m/s (perigee reduction of 150 km)

Spacecraft Dry Mass (kg)	Scenario 1 (115 m/s)	Scenario 2 (44 m/s)	Maximum permitted thrust level
1000 kg	118 kN.s	45 kN.s	400 N
1500 kg	176 kN.s	67 kN.s	600 N
2500 kg	294 kN.s	111 kN.s	1000 N

Proposed sizes based on heritage from D-Orbit

SRM class ID		S10	S50	S200	S110
SRM total impulse	[kNs]	10	50	200	110
SRM thrust	[N]	130	250	500	500
SRM ext diameter	[mm]	160	215	295	295
SRM length	[mm]	220	550	1005	553
SRM wet mass	[kg]	5	23	90	52
Burn duration	[s]	77	200	400	220

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Case 1
Propulsion Synthesis

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Case 1
Summary

Recommendations:

- classical monopropellant hydrazine system in blow-down
  - heritage and simplicity
- This can be with or without a main engine
  - off-the-shelf 400-N thruster is more than capable.

- For ion thruster as the main propulsion system, a single rocket motor could be used for the final manoeuvre.
  - However the cumulated hardware mass of both systems nearly wipes out the propellant mass saving brought by the ion propulsion over monopropellant hydrazine.

- Arcjets are not beneficial for this size of satellite.

Propellant	Propulsion Configuration	Pressure regulation	Penultimate perigee	Average thrust	Burn duration
Hydrazine	1x 400N + 4x20N	Blow-down	430 km	145.5 N	13.7 min
Hydrazine	1x 400N + 4x20N	Re-pres	430 km	200.4 N	<10.0 min
Hydrazine	4x20N	Blow-down	230 km	36.7 N	27.6 min
Hydrazine	4x20N	Re-pres	330 km	61.2 N	24.6 min
Hydrazine	Arcjet + 4x20N	Regulated	330 km (worst case)	67.7 N	21.9 min
Solid propulsion	1xS110	N/A	410 km	500 N	3.7 min
Solid propulsion	1xS50	N/A	200 km	250 N	3.3 min

Case 2
Summary

Recommendations:

- Monopropellant hydrazine re-pressurised system with 400-N main engine
  - A versatile architecture, covering satellites as heavy as MetOp-SG.
  - Different size of the propellant tank and re-pressurisation system.
- For heavier satellites at the limit of a launcher capability AND larger mission Delta-V budgets, arcjets may become interesting to meet the mass constraint.
- The same applies to ion propulsion with a solid rocket motor.
  - 500-N, 110-kN.s impulse SRM appears to be versatile and cover a mass range up to 2.5-ton.
  - The challenge is related to thrust-induced perturbations. Spin stabilisation may be suitable but must be analysed on a case-by-case basis

Propellant	Propulsion Configuration	Pressure regulation	Penultimate perigee	Average thrust	Burn duration
Hydrazine	1x 400N + 4x20N	Re-pres	430 km	301.0 N	10.0 min
Hydrazine	4x20N	Re-pres	230 km	55.8 N	28 min
Hydrazine	Arcjet + 4x20N	Regulated	300 km (worst case)	67.7 N	29.6 min
Solid propulsion	1xS110	N/A	280 km	500 N	3.7 min

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Case 3
Summary

- Based on existing launchers, it is unlikely that mass is an issue for satellites around 2500 kg.
  - Hence, the mass savings offered by electric propulsion with solid rocket motors, or combined hydrazine + arcjet, are not sufficient to be of interest.
- A hydrazine system based on a 400 N main engine with multiple 20 N thrusters for attitude control and disposal back-up scenario is currently preferable
  - Simplicity and heritage

Propellant	Propulsion Configuration	Pressure regulation	Penultimate perigee	Average thrust	Burn duration
Hydrazine	1x 400N + 4x20N	Re-pres	430 km	250.6 N	19.9 min
Hydrazine	4x20N	Re-pres	180 km	53.6 N	35.0 min
Hydrazine	8x20N	Re-pres	330 km	107.2 N	31.0 min
Hydrazine	Arcjet + 4x20N	Regulated	200 km (worst case)	67.7 N	31.1 min
Solid propulsion	1xS200	N/A	300 km	500 N	6.7 min
Solid propulsion	1xS110	N/A	180 km	500 N	3.7 min

Summary

DEFENCE AND SPACE

AIRBUS

Key findings

Trajectory

- There are no definite, strong recommendations for the trajectory.
- A targeted perigee of 30 km is a good first guess for most satellites, and could be refined at a later time of a project.
- The penultimate perigee altitude will be a compromise between the AOCS capability (together with the aerodynamic properties of the satellite) and the capability of the propulsion system to perform the final large manoeuvre.
- Simple relationships and figures of merit are provided to guide the preliminary sizing of the propulsion system.

Propulsion

- For most satellites, a monopropellant hydrazine propulsion system is preferable
  - acceptable performance
  - minimum system impact,
  - Particularly well suited if the Delta-V budget does not exceed 350-400 m/s approximately.
- Where thrust levels above 100 N are required, this can be achieved with European-made, 400 N thrusters.
  - Thus, it is not recommended to invest in the development of a thruster of an intermediate thrust level.
- For some niches, other propulsion systems could be considered:
  - Hydrazine arcjet or bi-propellant, for missions with large Delta-V budgets (>400 m/s) and where the satellite mass may be critical for the capability of the desired launcher.
  - Solid rocket motor for satellites with ion propulsion.

Key Findings (2)

AOCS:

- A thruster-based, three-axis stabilisation attitude control is necessary during the de-orbit manoeuvres.
- Disabling of the Safe Mode will be necessary below a critical altitude.
- For solid rocket motors, spin stabilisation may be a simpler solution than 3-axis stabiisation
  - but analyses are necessary for the inertias of the specific satellite under study

Recommendations

The following technology development may be of interest:

- In the case of all-electric satellites
  - a 110-kN.s SRM appears to be versatile.
  - However, other limitations remain with respect to attitude control when using this type of propulsion.

- Monopropellant systems are highly advantageous.
  - It may be worth considering developing green propulsion thrusters, including a 400-N main engine, for when the usage of hydrazine becomes proscribed.

Thank you