Executive Summary

End-Of-Life De-Orbit Strategies

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Foreword

The objective of the performed study on End-of-Life (EoL-) de-orbiting Strategies was to provide an overview and assessment of propulsion-related methods to de-orbit spacecraft in LEO, or spacecraft which pass through LEO, to assess their applicability to different spacecraft-mission combinations and to establish a know-how basis of End-of-Life de-orbit strategies.

The study was performed within the General Study Programme of ESA, ESA-contract 15316/01/NL/CK by a study team consisting of OHB-System AG, Hyperschall-Technologie Göttingen (HTG) and DLR-Space Launcher Systems Analysis (SART).

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Abbreviation List

	Abbreviations
AOCS	Attitude and Orbit Control System
BC	Board Computer
CER	Cost Estimation Relationship
CG	Cold Gas
COMSTAC	Commercial Space Transportation Advisory Committee
COTS	Commercial off the shelf
EM	Engineering Model
EP	Electric Propulsion
GEO	Geo-stationary Orbit
GNC	Guidance, Navigation and Control
GPS	Global Positioning System
GSO	GeoSynchrones Orbit
GTO	Geo Transfer Orbit
H/W	Hardware
HEO	Highly Elliptical Orbit
IADC	Inter-Agency Space Debris Co-ordination Committee
LEO	Low Earth Orbit; Orbit Range below 2000 km Altitude
LEOP	Launch and Early Orbital Phase
Microsatellite	Satellite Mass < 100 kg (TBC)
MMH	Monomethylhydrazin
Nanosatellite	Satellite Mass < 10 kg (TBC)
NGSO	Non-Geosynchronous Orbits
OTS	Off-the-Shelf
PFM	Protoflight Model
QA	Quality Assurance
S/C	Spacecraft
SCF	System Cost Figure
S/W	Software
TDRSS	Tracking and Data Relay Satellite System
TM/TC	Telemetry / Tele-command
WP	Work Package

Applicable Documents

AD	Document Title
AD 1	End-of-life de-orbit Strategies, Statement of Work, TOS-MPC/2125/ML/ml, Rev. 5, 8th January 2001
AD 2	ESA Space Debris Mitigation Handbook, Release 1.0, April 7, 1999

Reference Documents

Reference	Document Title
RD 1	De-orbiting Concepts and Related AOCS Design Requirements (WP 2000), EOL-OHB-TN-001, Iss. 1, 28.02.2002
RD 2	De-orbit Methods and Classification (WP 3000), EOL-OHB-TN-002, Iss. 1, 29.04.2002
RD 3	End-Of-Life De-Orbit Strategies (WP 4000), EOL-OHB-TN-003, Iss. 1, 03.07.2002
RD 4	End-Of-Life De-Orbit Strategies – Final Report, EOL-OHB-FR-001, Iss. 1, 03.07.2002

1. INTRODUCTION

The historical practice of abandoning spacecraft and upper stages at the end of mission life has allowed roughly 2 million kg of debris to accumulate in orbit. If this practice continues, collisions between these objects will, within the next 50 years, become a major source of small debris, posing a threat to space operations that is virtually impossible to control.

Recent studies show an increasing probability of collisions between intact spacecraft and debris. If no countermeasures are taken, the number of debris particles will grow with a growth rate in the order of up to 5% per year. Due to the very high relative velocities in the order of 10 km/s, even very small particles in the milimeter size range can destroy spacecraft subsystems and thus eventually lead to the loss of the complete spacecraft.

The uncontrolled growth of the space debris population has to be avoided in order to enable safe operations in space for the future. Space system operators need to take measures now and in the future to conserve a space debris environment with tolerable risk levels in the future, particularly in LEO and GEO altitude regions.

Simulations have shown that a real reduction of the debris population can only be achieved with very far-reaching measures. The only effective way to limit the growth of the orbiting debris is to remove satellites and rocket upper stages at the end of their mission from the near Earth space. This can be done either by de-orbiting or re-orbiting of the spacecraft. In the first case a deceleration maneuver is performed, resulting either in an immediate atmospheric re-entry or in an orbit with limited residual lifetime. In case of re-orbiting, the spacecraft orbit is raised to an altitude having no more interferences with the orbits of operational spacecraft.

The present document summarises the results of the study on End of Life De-Orbit Strategies, which investigates the **active disposal** of satellites and other S/Cs in LEOs. Major objective was the selection of the best-suited de-orbit strategy for the different satellite classes for uncontrolled de-orbit as well as for controlled de-orbit, if necessary. This study contained the following major tasks:

- Set-up and analyses of a reference satellites and missions data base
- Definition of De-orbit requirements for controlled and uncontrolled de-orbiting
- Definition of de-orbiting concepts including their impacts on satellite design
- Establishment of a de-orbit reliability model
- Establishment of a de-orbit cost model
- Determination of System Cost Figures
- Ranking and Classification of de-orbit methods
- Selection of the best-suited de-orbit strategy for the different satellite classes
- Review of existing technologies suitable for de-orbiting
- Identification of technology gaps

All results have been compiled and described in three technical notes [RD 1, RD 2, RD 3], a final report [RD 4] and various presentation hand-outs. This executive summary gives an overview on the performed tasks and approaches and summarises the major results and findings.

2. Summary

2.1 Reference Spacecraft

In the frame of the study, the different active End-of-Life De-Orbit strategies and concepts were investigated. Subject to this study were only actively stabilised satellites with Earth orbits below or passing through 2000 km. After analyses of the predicted future evolutions in the LEO-satellite market, a number of reference spacecraft were selected and different classes were defined, in order to cover the complete range of spacecraft and to find optimum solutions for each spacecraft class. For the study, the following spacecraft were considered as reference to investigate the de-orbiting manoeuvres:

Satellite Mass Category	Reference Mission
Nano, < 5 kg	Pathfinder
Nano, 5 - 20 kg	Munin
Micro, 20 - 100 kg	Safir 2
Mini, 100 - 500 kg	Abrixas
Medium, 500 - 1500 kg	IRS-1C
Large, > 1500 kg	2420 kg satellite (see candidate list)

Table 2-1: Final Reference Missions and Satellites

Following, a brief description of each reference satellite is shown.

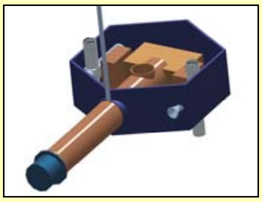
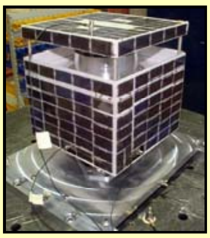
Reference Mission	Mission	Orbit	Power/ Mass/ Volume	Remarks
Nano Satellite Class, < 5 kg				
Pathfinder 	Investigation of Earth's magnetosphere Mission Life 1 year	Various, Apogee 5-25 RE (32000 km – 159000 km)	1.6 kg, Hexagon, diameter 24 cm, Height 9 cm Peak power 5 W, Avrg. 1 W	Demo-mission for a 100-200 nano satellite mission, Spin-stabilised, Cold-Gas for spinning
Nano Satellite Class, 5 - 20 kg				
Munin 	Auroral research, Mission Life 1 year	698 km x 1800 km	~ 4 W, 6 kg, Cubus, 21 x 21 x 21 cm^3	Passive magnetic stabilised

Figure 2-1: Reference spacecraft of Nano-Class

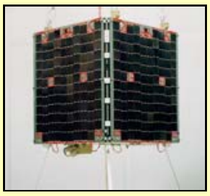
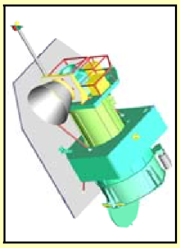
Reference Mission	Mission	Orbit	Power/ Mass/ Volume	Remarks
Micro Satellite Class, 20 - 100 kg				
SAFIR-2 (OHB) 	Communication, Mission Life 5 years	830km x 850km, SSO	Peak power 80 W, Avrg. 25 W 65 kg, Cubus, 50 x 50 x 50 cm³	Gravity-gradient stabilised + magnetic torquer
Mini Satellite Class, 100 - 500 kg				
Abrixas (OHB) 	Astronomic Sciences, Mission Life 3 years	580 km, 48°	200 W avrg. power, ca. 470 kg, 2.5 m x 1.8 m x 1.15 m	3 axes stabilised with momentum wheels

Figure 2-2: Reference spacecraft of Micro- and Mini-Class

Reference Mission	Mission	Orbit	Power/ Mass/ Volume	Remarks
Medium Satellite Class, 500 - 1500 kg				
IRS-1C 	Earth Observation, Mission Life 3 years	817 km, 98,1° (SSO)	810 W power generation, ~ 1250 kg (1160 kg dry) 1.7 m x 10.3 m x 2.1 m with solar panels unfolded	3-axes stabilised with reaction wheels and magnetic torquers Hydrazine propulsion system,
Large Satellite Class, > 1500 kg				
2420 kg - Satellite 	Astronomic Sciences, Mission Life 9 years	≈520 km, 53°	ca. 2420 kg ca. 4,7 m x 3,5 m x 2,4 m	3-axes stabilised, Approx. 700 kg Zerodur on-board

Figure 2-3: Reference spacecraft of Medium- and Large-Class

2.2 DE-ORBIT REQUIREMENTS

In a next step, de-orbiting requirements applicable to the spacecraft were defined. This set of requirements included design requirements applicable to specific subsystems or components of a spacecraft (e.g. propulsion, ACS), debris mitigation requirements, mission requirements and operational requirements. Special attention was given to the definition of controlled de-orbit requirements. In the frame of the definition of de-orbit requirements, the following cases were considered:

- Orbital Lifetime of spacecraft
- Effect of De-Orbit Function Reliability
- Controlled De-Orbit
- Uncontrolled De-Orbit
- Disposal to orbit with limited lifetime (e.g. 15, 25, 40 years).

The main driver for the orbital lifetime is initial orbital altitude. In the envisaged range of spacecraft, it varies a few month to weeks below 400 km up to more than 1000 years above 900 km initial altitude. For elliptic orbits with an Apogee altitude >30000 km, the inclusion of third-body perturbations in the determination of the orbital lifetime is necessary. It is considerably shorter than determined when neglecting them. Below 30000 km, these disturbances are negligible. Spacecraft in circular orbits below ≈600 km have an orbital lifetime below 15 years in the practical range of area-to-mass ratios. These spacecraft do not require a manoeuvre for an uncontrolled de-orbit or to an orbit with limited lifetime. With regard to the effect of debris mitigation, a de-orbit function reliability R=0,9 was found to be sufficient, assuming that from 2015 onwards 50% of the missions and from 2030 onwards 100% of the missions perform a de-orbit manoeuvre. Most demanding with regard to the required ΔV is the direct controlled de-orbit manoeuvre, followed by an uncontrolled but direct re-entry. Some 15% in ΔV can be saved, if a transfer to an orbit with limited lifetime is permissible. The difference in ΔV with regard to a disposal to a 15-year orbit or to a 40-year orbit is only marginal (<5%).

The aerodynamic torque as the strongest external disturbing perturbation becomes relevant below an orbital altitude of 300 km. Thus, in order to avoid excessive stabilisation means, the active phase of the de-orbit manoeuvre shall be performed above this altitude. Due to thrust misalignments and CoG-variations, even without disturbing external torques a stabilisation of the spacecraft during the propulsive phases is required. This can be provided either by a 3-axes-stabilisation system or a spinning around 1 axis.

In the case where the atmospheric destruction process is expected to be incomplete or the residual risk for ground population is too high, a controlled re-entry with prescribed re-entry location has to be carried out. This is the case, if the predicted risk of human casualties exceeds a specified limit, typically 0.01% per re-entry event, or the spacecraft contains hazardous objects with large masses and/or radioactive or poisonous materials. Thus, it was determined if and how much a spacecraft will be destroyed during its atmospheric re-entry. Very small satellites (m<≈ 20 kg) burn up during re-entry for all initial conditions. Therefore controlled de-orbit is not required for this class of spacecraft, except if they have hazardous objects on board. Very Heavy satellites (m>≈ 500 kg) survive the re-entry at least partially in any case. Therefore in this case the re-entry should be controlled and steep enough (perigee < 60 km) to ensure a well-defined impact area and location. For medium-sized satellites it depends on the initial conditions whether they reach ground in larger pieces or not. For a steep re-entry after a controlled de-orbit manoeuvre it is more likely that larger pieces reach ground, but in this case it is well known where they will impact. For a shallow re-entry, as it is the case after a disposal to a limited lifetime orbit, it is more likely that the satellite breaks up into small pieces, and the fragment dispersion can become large, but the fragments are likely to burn up during their way to ground or loose at least enough mass to represent no severe hazard when impacting.

2.3 INVESTIGATED PROPULSION SYSTEM OPTIONS

The complete range of propulsion system options, including storable chemical propulsion systems as well as electric propulsion systems was reviewed and their requirements with regard to the de-orbit function were determined. The main principal characteristics of these options are summarised following:

	Advantages	Disadvantages
Cold Gas	■ Simple ■ Low system cost ■ Reliable ■ Safe	■ Extremely low Isp ■ Moderate impulse capability ■ Low density ■ High pressure
Mono Propellant	■ Wide thrust range ■ Modulable ■ Proven	■ Low Isp ■ (mostly) toxic fuels
Bi-Propellant (storable)	■ Wide thrust range ■ Modulable ■ Proven	■ Complex ■ Costly ■ Heavy ■ Toxic
Solid Propulsion	■ Simple ■ Reliable ■ Low cost ■ High density ■ Low structural index	■ One thruster per burn ■ Total Impulse fix ■ Currently not qualified for long-term space application
Hybrid Propulsion	■ Simple ■ Modulable ■ Low cost ■ Reliable	■ Not qualified ■ Lack of suitable oxidiser for long-term mission
Electrical Propulsion	■ Very high Isp	■ Low thrust ■ Complex ■ Large manoeuvre time ■ Power consumption

Table 2-2: Principal characteristics of spacecraft propulsion systems

In a next step, detailed de-orbit manoeuvres and strategies were investigated and traded against each other. The usage of in-plane multi-burn manoeuvres was found to be the most appropriate de-orbit manoeuvre strategy. This is valid for all engine thrust-levels, for high-thrust engines down to low-thrust electric propulsion engines. For the latter, decreased average power requirements can be achieved by applying a multi-burn strategy.

2.4 CLASSIFICATION AND RANKING

Mass and volume models have been established to determine the impact of the various de-orbiting concept options on the design of the spacecraft. A set of criteria to evaluate different de-orbiting strategies, concepts and propulsion systems have been defined. A System Cost Figure (SCF), allowing a standardised weighting of different influence parameters, was established, in order to perform a quantitative evaluation and ranking of the different options. The results of this ranking are summarised in Table 2-3.

Satellite	Propulsion System					
	Cold Gas	Solid Prop.	Mono Prop.	Bi-Prop.	Arcjet	Ion
Pathfinder	3	1	2	4	N/A	N/A
Munin	3	1	2	4	N/A	N/A
Safir	4	1	2	3	N/A	N/A
Abrixas	6	1	2	3	4	5
IRS-1c	6	1	1	3	4	5
2420kg-Sat	6	1	2	2	4	5

Table 2-3: Ranking of de-orbiting options for all spacecraft

As a general result, cold gas propulsion systems were found to be limited in their applicability as de-orbit propulsion system to relatively small satellites ($m_{Sat} \leq\approx$ 500 kg), due to the severe mass impact. Electric propulsion systems are most useful in big spacecraft, ($m_{Sat} \geq\approx$ 1500 kg), having either electric propulsion for other reasons (e.g. station keeping) or having installed substantial electric power under the assumption of uncontrolled de-orbit. The solid propellant system is for almost all spacecraft the best solution to perform an end-of life de-orbit manoeuvre, followed by the mono-propellant system. This ranking is relatively insensitive to the size of the manoeuvre. Electric propulsion can be used as de-orbit option only if the spacecraft exceeds a minimum size. For smaller spacecraft, the additional electric power to be installed makes the use of the electric propulsion not reasonable. For larger spacecraft and braking manoeuvres, the I_{sp} of the propulsion system is getting more and more important. Thus, electric propulsion become more and more competitive. The reason for the worse ranking of E.P. is, that in any case considered here it was needed to install additional electric power to keep in case of electric propulsion the manoeuvre time below the specified one year. This disadvantage with regard to mass, cost, volume, complexity could not be compensate, resulting in a 4th rank for electric propulsion at the best. If it is permissible to extend the overall manoeuvre duration, the electric propulsion becomes more and more attractive, especially for larger manoeuvres as required for an uncontrolled de-orbit from very high initial altitudes. A controlled de-orbit can not be performed exclusively with electric propulsion, since at least the final burn requires to generate a significant ΔV outside the atmosphere, requiring a sufficient thrust level not available with electric propulsion. In case of a controlled de-orbit, a combination of a high-thrust system with electric propulsion or a pure chemical propulsion system is needed.

Although the absolute effort to install a EOL-de-orbit function into a spacecraft increases with the size of the spacecraft, the relative impact is most for the small vehicles. Whereas a spacecraft of the Pathfinder-class has a SCF in the order of $SCF_{Pathfinder,\ best}$=500, it decreases by a factor of 10 for the >2000kg class. Thus it is more easy to implement the EoL-de-orbit function into the larger spacecraft, where the overall budgets are increased only by a few percent.

The following changes in the overall system budgets (ROM) have to be expected for the reference-S/Cs for two investigated single/multi-burn de-orbit manoeuvres creating a total valid of ΔV=100 m/s & 200 m/s.

Spacecraft-Class	Δ-Cost [M€]	Δ-Mass [kg]	Δ-Volume [ltrs]
Nano, m < 5 kg	+0,7÷0,8 M€ (+75%)	+0,7÷0,9 kg (+60%)	+2.5÷3 ltrs (+50%)
Nano, 5 kg < m < 20 kg	+0,9÷1 M€ (+50%)	+1,4÷1,9 kg (+60%)	+3÷6 ltrs (+45%)
Micro, 20 kg < m < 100 kg	+0,9÷1 M€ (+15%)	+6÷12 kg (+20%)	+6÷9 ltrs (+10%)
Mini, 100 kg < m < 500 kg	+1,2÷1,4 M€ (+4%)	+25÷50 kg (+10%)	+15÷30 ltrs (+1%)
Medium, 500 kg < m < 1500 kg	+1,9÷2,46 M€ (+3%)	+70÷140 kg (+10%)	+40÷80 ltrs (+1%)
Large, m > 1500 kg	+3÷4 M€ (+3%)	+130÷230 kg (+10%)	+70÷130 ltrs (<1%)

Table 2-4: System Budgets for EoL-manoeuvre, ΔV=100 m/s & 200 m/s (using solid propellants)

The values given are those determined for the investigated reference spacecraft for the two defined manoeuvres (ΔV=100 m/s & 200 m/s). With these manoeuvres, the following de-orbiting tasks could be performed (controlled de-orbit - Table 2.4-5, uncontrolled de-orbit - Table 2.4-6):

ΔV	Circular orbit	Ell. Orbit, Perigee 300 km	Ell. Orbit, Perigee 500 km	Ell. Orbit, Perigee 1000 km	Ell. Orbit, Perigee 1500 km	Ell. Orbit, Perigee 2000 km
100 m/s	400 km	Max. 71 m/s for all Apogees	Apogee ≥ 6600 km	Apogee ≥ 33500 km	Apogee ≥ 58000 km	Apogee ≥ 83000 km
200 m/s	780 km	Max. 71 m/s for all Apogees	Max. 127 m/s for all Apogees	Apogee ≥ 7400 km	Apogee ≥ 20500 km	Apogee ≥ 32500 km

Table 2.4-5: Maximum initial orbital altitudes for controlled de-orbit

Spacecraft	∆V= 100 m/s	∆V= 200 m/s
Pathfinder	■ from 940 km to 15 years orbit ■ from 980 km to 25 years orbit ■ from 1020 km to 40 years orbit	■ from 1330 km to 15 years orbit ■ from 1370 km to 25 years orbit ■ from 1400 km to 40 years orbit
Munin	■ from 875 km to 15 years orbit ■ from 910 km to 25 years orbit ■ from 935 km to 40 years orbit	■ from 1250 km to 15 years orbit ■ from 1290 km to 25 years orbit ■ from 1320 km to 40 years orbit
Safir-2	■ from 830 km to 15 years orbit ■ from 870 km to 25 years orbit ■ from 900 km to 40 years orbit	■ from 1200 km to 15 years orbit ■ from 1240 km to 25 years orbit ■ from 1280 km to 40 years orbit
Abrixas	■ from 850 km to 15 years orbit ■ from 910 km to 25 years orbit ■ from 915 km to 40 years orbit	■ from 1210 km to 15 years orbit ■ from 1260 km to 25 years orbit ■ from 1300 km to 40 years orbit
IRS-1C	■ from 880 km to 15 years orbit ■ from 910 km to 25 years orbit ■ from 950 km to 40 years orbit	■ from 1260 km to 15 years orbit ■ from 1300 km to 25 years orbit ■ from 1340 km to 40 years orbit
2420 kg-spacecraft	■ from 830 km to 15 years orbit ■ from 870 km to 25 years orbit ■ from 900 km to 40 years orbit	■ from 1210 km to 15 years orbit ■ from 1250 km to 25 years orbit ■ from 1280 km to 40 years orbit

Table 2.4-6: Maximum initial orbital altitudes for disposal to orbit with limited lifetime

For a broad range of typical spacecraft having initial circular orbital altitudes below about 590 km, no specific EoL-manoeuvre is required, because their remaining lifetime is below 15 years, the lowest value presently discussed with regard to debris mitigation.

2.5 De-Orbit Strategy and Concept

The following described strategies for a controlled and uncontrolled EoL-manoeuvre were found to be best-suited for the different spacecraft classes:

- single or multiple apogee burn(s) to lower perigee until final limited lifetime orbit
- single burn time < 20% of orbit period, active braking only above 300 km (disturbing atmospheric torque)
- Spin stabilisation along the de-orbit thrust vector axis, which is aligned parallel to the velocity vector in the apogee (orbit tangent)

- In case of a controlled de-orbit, the final burn shall result in a perigee altitude equal or lower 60 km altitude, in order to have a safe atmospheric capture and a confined and pre-defined ground impact area
- The EOL function is assumed to be activated by ground command or by the EOL function itself if sign of life signal from S/C is interrupted
- The EOL function is assumed to be autonomous controlled by appropriate electronic & SW if the add. effort would not exceed ground controlled operations
  - This is in general true for long manoeuvre durations (E.P.)
  - Supervision and correction from ground shall be possible
- In case of controlled de-orbit, the manoeuvre is assumed to be ground supervised in order to cope with unexpected events
- In case of uncontrolled de-orbit full autonomous control is assumed
- Sun sensor or horizon sensors, magnetometer, and magnetic torquers as sensors recommended.
- Solid propellant or MEMS propulsion system

Assuming that the de-orbit function is implemented in future satellite designs, two baseline concepts are recommended:

1) Integrated de-orbit function: The de-orbit function is integral part of in the Attitude & Orbit Control subsystem (see Figure 2-4).

2) Additional De-orbit Control Subsystem: An extra subsystem is dedicated to the de-orbit function (see Figure 2-5).

An add-on de-orbit module including own control and power is recommended in case of existing satellite design without de-orbit function.

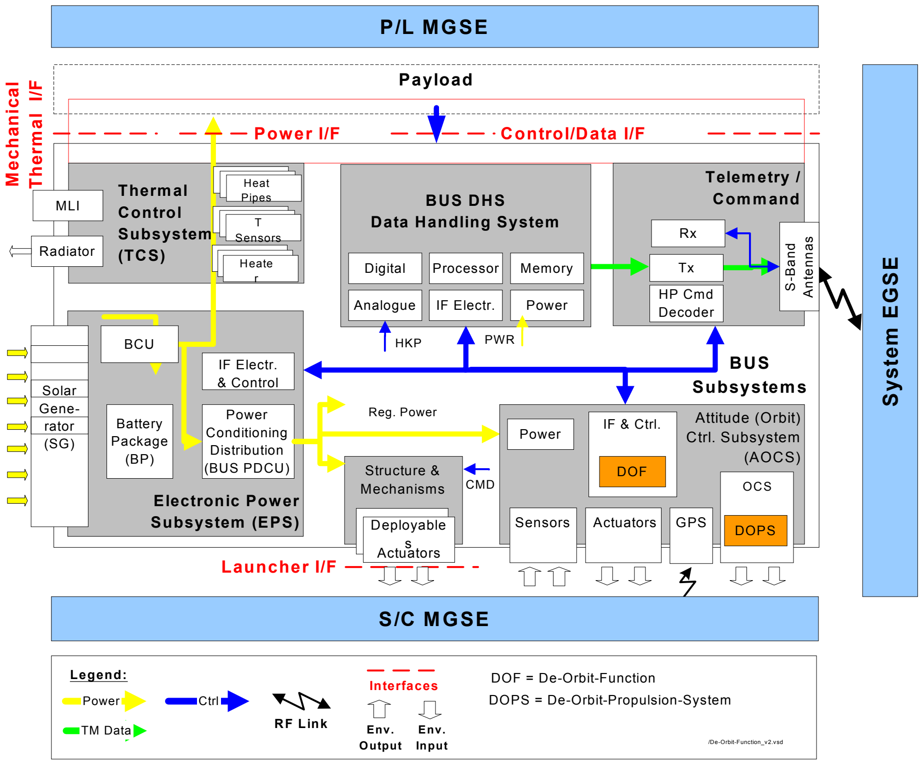


Figure 2-4: Integrated De-Orbit Function Design

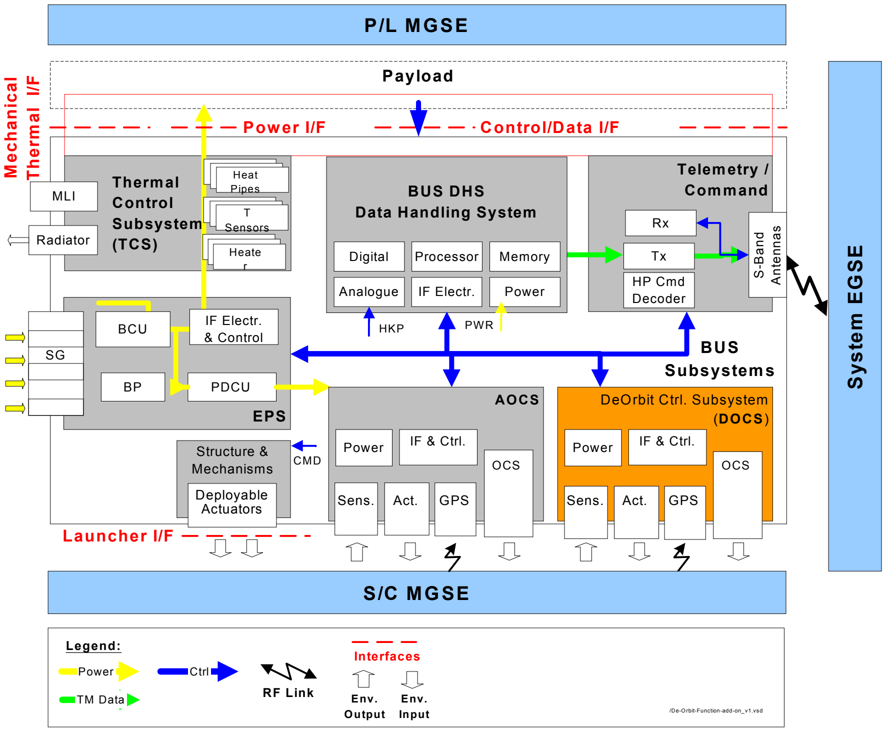


Figure 2-5: Additional De-Orbit Control Subsystem Design

Concerning operations, the major distinction can be made with respect to the necessity of a controlled or an un-controlled de-orbit. Therefore some generic operational guidelines can be distinguished here as well, which are outlined in the following.

Controlled de-orbit

The necessity of a multi-burn manoeuvre exists, if the required ΔV for the de-orbit cannot be achieved during a single-burn duration, which is required to be below 20% of the orbit period. Otherwise a single-burn manoeuvre can be performed.

Furthermore, any single-burn de-orbit manoeuvre, can be divided into any fraction of multiple burns, if the OCS supports this. An intentionally split of the de-orbit manoeuvre into several burns, each of which have to be supervised from ground, can improve the reliability and accuracy of the complete de-orbit manoeuvre, especially at high initial altitudes and an intensive burn duration for a single-burn manoeuvre. Because the reliability and accuracy of the de-orbit is especially important for the controlled re-entry, a multi-burn strategy should always be considered.

For multi-burn manoeuvres this leads to the requirement to minimise the time frame between the planning of the controlled de-orbit and the final burn. With respect to the type of propulsion system being used, the following guidelines can be stated:

- A total of 3-5 burns are assumed to be a meaningful maximum number for chemical OCS.
- The usage of solid propellant favours a single-burn manoeuvre, which minimises the planning and operational effort, but may decrease the manoeuvre accuracy. The investigation of the availability of solid propellant OCS for a multi-burn manoeuvre is recommended.
- If an E.P. system is used for lowering the perigee, the detailed planning of the final, high-thrust re-entry burn(s) have to be performed after the finalisation of the E.P. manoeuvre phase.

In general one can say, that for a controlled de-orbit precise orbit propagators and manoeuvre planning tools have to be available. In order to get the inputs for the planning of the de-orbit manoeuvres, and in order to command the manoeuvres and to check-out their success, the ground based operational effort is furthermore intensified by the following aspects:

- precise determination of the actual satellite orbit including
  - ground based tracking and ranging effort, e.g. use of tracking radar ground stations such as FGAN,
  - satellite based orbit determination data, based on GPS measurements,
  - use of the very latest NORAD two-line-elements,
- use of additional ground stations for direct commanding of the de-orbit manoeuvres; otherwise time-tagged commands have to be used,
- use of additional ground stations for the supervision of the de-orbit manoeuvres and for checking-out the success of the final burn.

Uncontrolled de-orbit

In the first place, the same operation guidelines and strategies apply for the un-controlled de-orbit as for the controlled one. But for the un-controlled de-orbit the effort is more relaxed, because there are no strict requirements with respect to the location of the argument of perigee and the additional disturbances of the de-orbit ellipse.

The important strategies and guidelines, which apply for the planning and the operation of an un-controlled de-orbit are summarised in the following:

- The perigee has to be lowered with a single-burn or a multi-burn strategy.
- The duration of one single burn has to be below 20% of the orbit period; for a longer burn duration a multi-burn strategy becomes mandatory.
- Any single-burn de-orbit manoeuvre, can be divided into any fraction of multiple burns, if the OCS supports this.
- The single/first de-orbit manoeuvre can be performed in the range of the nominal TM/TC ground station; any further manoeuvres can be performed by direct commands or by time-tagged ones, uploaded by the G/S (if not the complete de-orbit is performed autonomously by the satellite due to the loss of the TM/TC contact).
- When planning and commanding the further de-orbit burns of a multi-burn strategy, the duration of the burns and the disturbance of the line of apsides have to be taken into account, in order to perform the manoeuvre(s) in the vicinity of the apogee and to receive the desired perigee altitude and the required EoL effect.
- Each de-orbit manoeuvre is autonomously controlled by the spacecraft itself and checked by the nominal TM/TC ground station at the next contact period. The use of additional ground stations is assumed not to be necessary.

2.6 EoL-DE-ORBITING: TECHNOLOGY STATUS

In order to define the need for technology development, the following definition was used to determine the today available **T**echnology **R**eadiness **L**evel (TRL):

Qualification Level		Characterisation of Technology Status
System test, launch and operations	**TRL 9**	Actual system „flight proven" through successful mission operations
System/subsystem development	**TRL 8**	Actual system completed and „flight qualified" through test and demonstration (ground or flight)
Technology demonstration	**TRL 7**	System prototype demonstration in a space environment
	TRL 6	System/subsystem model or prototype demonstration in a relevant environment (ground or space)
Technology development	**TRL 5**	Component and/or breadboard validation in relevant environment
	TRL 4	Component and/or breadboard validation in laboratory environment
Research to provide feasibility	**TRL 3**	Analytical & experimental critical function and/or characteristic proof-of-concept
	TRL 2	Technology concept and/or application formulated
Basic technology research	**TRL 1**	Basic principles observed and reported

Table 2-7: **US Technology Readiness Level Definition**[1]

This definition was used in the following for the characterisation of the various technologies applicable for the EoL-function, to identify the need for further developments and proposals to close identified technology gaps.

[1] Source: FESTIP Design Standards and Technology Assumptions, FSS-DRI-SC-3110-001

Following, the assessed technologies and their TRL is given.

Technology	TRL (in Europe)
■ System level deorbit function	2
■ Scalable de orbit Sensor-/Actuator Package (system level)	2
■ Scalable de orbit Sensor-/Actuator Package (component level)	9
■ Arcjets	7
■ Resistojets/EHT	5
■ SPT/HCT	7
■ Ion Thrusters	8
■ PPT	1
■ Small Solid Motors	7
■ Solid Propellant Micro Propulsion	5
■ Solid State Gas Generators (nitrogen and oxygen)	5
■ Solid State Gas Generators (hydrogen)	3/4
■ Micro Cold Gas	4/5
■ Micro-pumps	3 (?)
■ Micro-Bipropellant	1
■ Micro-Monopropellant	1
■ Alternative Propellants	3-6
■ Hybrid propulsion	4
■ Self-consuming structures	1

Table 2-8: Summary of technology status in Europe

The summary in Table 2-8 shows clearly, that a number of component technologies, having regular applications in spacecraft but for other functions than EoL-de-orbiting, are well advanced and not far from a “flight-ready” status. A big gap exists on system level due to the fact, that until today a EoL-manoeuvre was not demonstrated in Europe. Thus, here measures to close this gap, e.g. by the definition and execution of ground- and flight experiments/projects to verify the EoL-function is recommended.

Taking the results of the overall study, the solid-propellant propulsion systems have shown a great potential for use in a de-orbit function. Thus, in a near-term approach it is recommended to qualify the required smaller solid propellant motors until they reach the TRL9. Other technologies in the field of propulsion are very attractive and might have the potential to be very competitive to the favoured near-term-solution solid propellant, but they are far away from a “flight-ready” status. In this field, a long-term program with the objectives to demonstrate feasibility and to develop and demonstrate these technologies is necessary. Here also other applications might have a benefit from these improvements.