SELECTION OF A PROPULSION SYSTEM FOR JASON-CS IN ORDER TO FULFIL SPACE DEBRIS MITIGATION REQUIREMENTS FOR ESA PROJECTS

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ABSTRACT

For two decades, the mission Topex-Poseidon and its successor mission Jason/Ocean Surface Topography Mission provide satellite data for the analysis of sea topography, wave heights and wind speeds. For the continuation of service mission Jason-CS, ESA's choice to rely on the CryoSat-2 platform design permits re-use of a well established product and proven processes. An industrial consortium led by Astrium GmbH has built the satellite CryoSat-2 which for over three years successfully provides altimeter measurements of the polar ice cap thickness evolutions. This platform is perfectly suited for accommodation of the Jason-CS instruments. Unlike CryoSat-2, Jason-CS is required to perform a post-mission disposal according to the Requirements for Space Debris Mitigation for ESA Projects. This paper discusses different technologies in terms of efficiency, feasibility and accommodation, aiming at minimizing necessary spacecraft design modifications.

1 JASON CONTINUTITY OF SERVICE

Jason-CS (Jason Continuity of Service) is the continuation of the famous Jason/Ocean Surface Topography mission, itself going back to the Topex-Poseidon mission. The mission objective remains to measure the ocean topography, which is obtained by the measuring the distance in between the satellite and the sea surface. Thanks to the precise orbit determination, this is known very accurately. The Jason-CS mission is embedded in the Global Monitoring for Environmental and Security Initiative (GMES). Its predecessor missions were bilateral missions in between France and the United States, carried out by both the French Space Agency Centre National d'Etudes Spatiales (CNES) and the National Aeronautic and Space Agency (NASA). Jason-CS will be carried out as a transatlantic partnership with the National Oceanic and Atmospheric Administration (NOAA), Jet Propulsion Laboratory (JPL), the European Organisation for the Exploitation of Meteorological Satellites (EUMETSAT), CNES, the European Commission (EC) and the European Space Agency (ESA). ESA is responsible for the procurement of the space segment and has identified the possibility to benefit of the CryoSat-2 platform heritage. This platform was built by Astrium GmbH.

The payload suite will include the Poseidon-4 altimeter manufacturer by Thales Alenia Space (TAS), the Advanced Microwave Radiometer provide by JPL, a Doppler Orbitography by Radiopositioning Integrated on Satellite (DORIS), Receiver, Global Navigation Satellite System Receiver (GNSSR) and a Radio Occultation Receiver and a Laser Retro-Reflector (LRR).

The mission orbit around 1336 kilometres (km) altitude with an inclination of 66 degrees (deg) ensures global coverage of most of Earth's non-frozen oceans. This orbit is non-sun-synchronous.

2 CRYOSAT-2 HERITAGE PLATFORM

Since its launch in April 2010, CryoSat-2 provides altimetry data from on drifting, near-polar orbit at an altitude of around 720 km. In order to cope with this drifting orbit, the CryoSat-2 overall platform design has the solar arrays mounted on a roof-shaped structure on top of a prolonged rectangular main body. Thanks to this, the power demand of the spacecraft is provided for all local hours of the ascending node. A drawing of the CryoSat-2 satellite is given in Fig. 1.

The front part of the satellite (top right in Fig.1) hosts the two antennas of the radar altimeter Synthetic

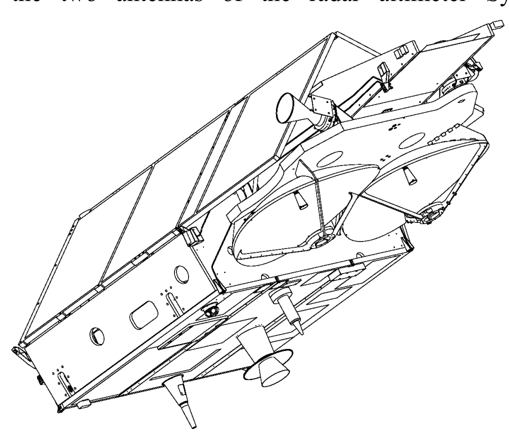

Figure 1.CryoSat-2 Spacecraft

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Aperture Radar (SAR)/Radar Altimeter (SIRAL, produced by TAS). Additionally, CryoSat-2 hosts also a LRR and a DORIS payload for precise orbit determination.

As can be seen, there are many resemblances in between Jason-CS and CryoSat-2, notably a similarity in payload and a drifting orbit for both missions.

Concerning propulsion, CryoSat-2 is equipped with a nitrogen-based cold gas reaction control system, providing both means for orbit and attitude control.

CryoSat-2, however, is not designed for a post-mission disposal.

3 SPACE DEBRIS MITIGATION

As an ESA project, Jason-CS is required to ensure Space Debris Mitigation according the European Code of Conduct and an applicable internal policy. Moreover, this implies to ensure that the Jason-CS spacecraft will not remain for longer than 25 years in the protected Low Earth Orbit (LEO) zone.

In order to assess the required Δv, available tools of the ESA and the NASA have been used. The tools Debris Risk Assessment and Mitigation Analysis (DRAMA) by ESA and Debris Assessment Software (DAS) by NASA provide the required Δv for initial orbits. Furthermore, these tools permitted a low fidelity analysis on the casualty risk which yielded compliance with respect to the required threshold.

Different approaches have been investigated, namely:

- Controlled, direct re-entry
- Delayed re-entry via elliptic orbit
- Delayed re-entry via circular orbit
- Re-orbiting outside of protected LEO zone

A controlled and direct re-entry is executed as a single manoeuvre or a series of manoeuvres, lowering the orbit's perigee to an altitude that ensures atmospheric capture at a steep angle, yielding a defined ground footprint.

For an uncontrolled re-entry, the orbit is lowered in such a way that atmospheric capture occurs within a given post-mission lifetime, which for Jason-CS is specified to 25 years at most. Therefore, the orbital lifetime of orbits fulfilling this characterisation has been investigated, using the aforementioned tools DAS and DRAMA. Depending on the propulsion system, the disposal orbit is reached due to continuous thrusting over the orbit or via impulse burns in the apogee region (either as a single manoeuvre or a series thereof). When a continuous thrust is applied over the entire orbit, a circular orbit will remain circular. Applying impulse manoeuvres over the apogee region of the orbit will result in an elliptic orbit.

Re-orbiting outside the protected LEO zone is discarded as it is not permitted within the ESA Internal Policy applicable for Jason-CS. However, a re-orbiting to an altitude above the protected LEO zone would not be desirable in terms of Δv.

As there will be no attitude control (neither active nor passive), a randomly tumbling spacecraft is assumed. This implies that the cross sectional area is obtained by averaging the Concerning the spacecraft mass, the CryoSat-2 mass budget, updated with the new instrumentation yields a first estimate of the actual Jason-CS dry mass.

While in the course of the spacecraft development a more refined mass budget will be obtained, this initial configuration serves a basis to compare different principles and to select a suited propulsion system technology for Jason-CS in order to fulfil the space debris mitigation requirements.

A controlled re-entry requires more propellant for manoeuvres than a delayed re-entry, not to mention additional challenges such as providing attitude control at very low altitudes. Hence, the latter has been assessed. This analysis has yielded that when lowering to a circular orbit, an orbit height of approximately 610 km will provide an orbit with a compliant dwell time and when applying impulse thrusts, a perigee of roughly 380 km is required. Respectively, the propulsion system will need to provide the capability for 370 metres per second (m/s) for continuous thrust or 240 m/s for impulse manoeuvres to lower orbit's perigee.

4 TECHNICAL IMPLEMENTATION

Next to ensuring the compliance to the Space Debris Mitigation requirements, the Jason-CS propulsion system has to ensure that further constraints are fulfilled. As aforementioned, Jason-CS will re-use the CryoSat-2 platform. Therefore, it is a key element to minimize modifications for cost efficiency. Supporting the continuation of an operational mission, the platform is designed to rely on simple technologies with significant heritage. Hence, the following technologies have been considered:

- Cold Gas Propulsion System
- Mono-Propellant Propulsion System
- Bipropellant Propulsion System
- Electric Propulsion System
- Solid Propulsion Engine

Generally, a key parameter for the sizing is the required amount of propellant, which can be computed according to the well-known formula (repeated in Eq. 1 for convenience).

$$m_p = m_0 \cdot \left(e^{\frac{\Delta v}{g \cdot I_{sp}}} - 1 \right) \tag{1}$$

In Eq. 1, m_P denotes the required propellant mass, m_0 the spacecraft dry mass, g the acceleration and I_{sp} the specific impulse. It can be seen that the specific impulse has a significant impact on the propellant mass. Constant values for the specific impulses are assumed for the comparison of different technical means to perform the post-mission disposal. These typical values, which have been used in the assessments, are summarized in Tab. 1.

Table 1 Assumed Specific Impulse for Propellants

	Specific Impulse [s]
Cold Gas (nitrogen)	70
Monopropellant (hydrazine)	210
Bipropellant (mixed oxides of nitrogen)	280
Solid propulsion engine	300
Electric Propulsion	500

Using typical values (denoted in Tab.1) in Eq.1 permits an initial comparison of the propellant mass for the different propulsion system technologies. As within this study, the focus lays on the comparison of different technologies and not on the actual design, loss factors, such as gravity loss, are neglected. The results are visualized in Fig. 2.

While the propulsion mass provides first estimate on propellant system sizing, it lacks additional points of interest such as overall system mass and compatibility to the existing satellite design. This is assessed for each propulsion system technology individually.

4.1 Cold Gas Propulsion System

With respect to the CryoSat-2 heritage platform, the usage of a cold gas propulsion system for Jason-CS bears the significant advantage that the sole change required is to host the additional amount of propellant. On CryoSat-2, a cold gas propulsion system with nitrogen propellant provides the means for the orbit control manoeuvres. For Jason-CS, however, the required Δv is significantly higher and thus, nitrogen with its low specific impulse implies at least 300 kilograms (kg) assuming a density of 276 kilograms per cubic metre (m³) of propellant and a 740 kg spacecraft dry mass. Given the low specific mass, a volume of

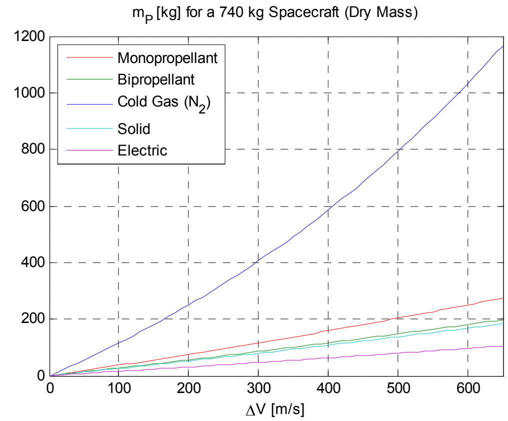

Figure 2. Fuel mass over Δv for different technologies

about one thousand litres would be required for the nitrogen mass. Alternatively, Freon-14 or Xenon could be used with a better specific density impulse, requiring a smaller propellant volume but a higher mass. Assuming a Δv of 240 m/s and 740 kg SPACECRAFT dry mass for a series of impulse manoeuvres to lower the satellite's perigee yields the results in the same table. As the required propellant volume and mass values are evidently exceeding the accommodation possibilities by far, the cold gas propulsion system technology can be considered not suitable for the Jason-CS post-mission disposal, even when assuming improvements of the specific impulse when using resistor thrusters.

Table 2 Cold Gas: Characteristics for Δv = 240 m/s

	Specific Impulse [s]	Density [kg/m³]	Mass [kg]	Volume [m³]
Nitrogen	70	276	310	1.1
Freon-14	50	1031	467	0.5
Xenon	30	2170	933	0.4

4.2 Solid Propulsion Engine

Solid propulsion engines bear the advantage that they store the propellant necessary for the post-mission disposal within a relatively small volume. Depending on the detailed designed, it could even be possible to attach the solid propulsion engine within the launcher interface ring of the spacecraft and hence limiting required design changes on the CryoSat-2 heritage platform.

A solid propulsion engine provides thrust in the range of several kilonewtons (kN). Once it is ignited, the process cannot be actively controlled anymore. When not perfectly in line with the satellite's centre of mass, the thrust vector will induce significant disturbance

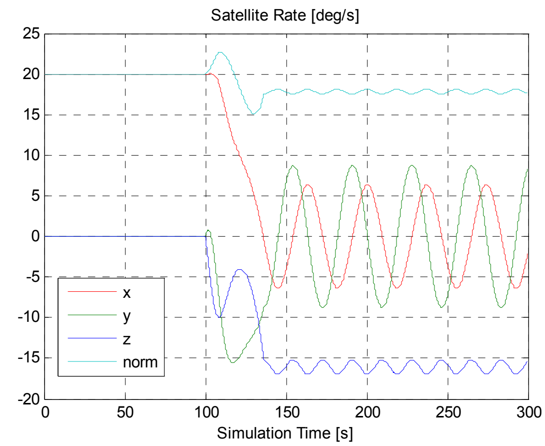

Figure 5. Spacecraft Rate, Initial Rate: 20 deg/s

torques. As this cannot be fully avoided, these disturbances have to be either compensated by the Attitude and Orbit Control System (AOCS). The expected disturbances, even when assuming a favourable thrust vector alignment, are too high to be compensated by a standard three axis stabilisation AOCS. A mounting uncertainty in the millimetre range in positioning of the solid propulsion engine thruster outlet and a misalignment of a tenth of a degree (deg) has been assumed. Hence, the possibility of spin stabilisation has been investigated as well.

For this, a simple offline simulation environment has been established, simulating simultaneously attitude and orbit dynamics of the Jason-CS spacecraft. A perfect initial attitude with respect to the velocity vector is simulated. The satellite x-axis runs parallel to the velocity vector. At 100 seconds after the start of the simulation, a thrust of about 6.5 kN is applied towards the minus x-axis direction for a duration of 36.4 s, ideally providing the capability to lower the perigee height to about 200 km. Several scenarios have been

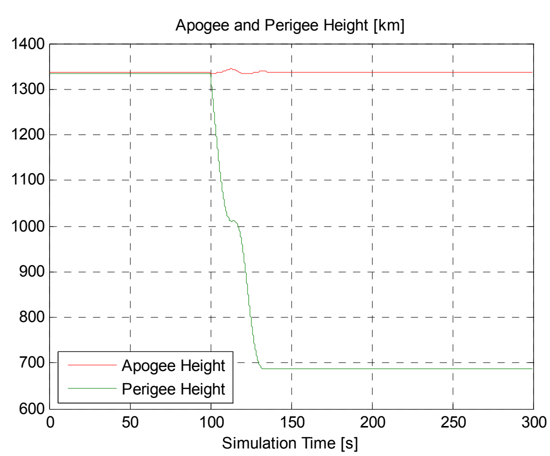

Figure 4. Apogee and Perigee, Initial Rate: 20 deg/s

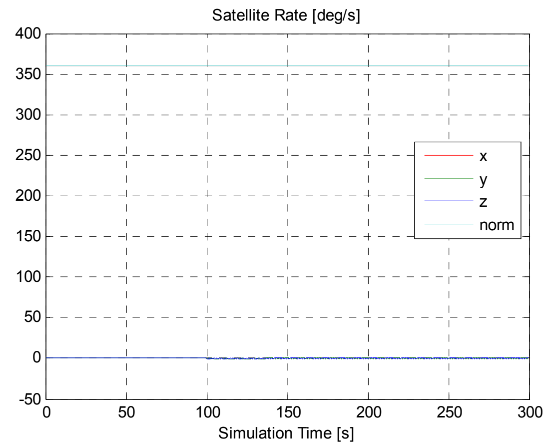

Figure 3. Spacecraft Rate, Initial Rate: 360 deg/s

simulated. The deviation moments of moments of inertia matrix are assumed to be zero. The initial conditions are varied with respect to thrust direction and initial rate.

As expected, once the thrust once force vector is aligned perfectly with the centroidal axis, the orbit manoeuvre is executed as desired, even without any passive stabilisation. However, once there is a small misalignment, even with an initial rate of 20 degrees per second (deg/s) a passive stability with respect to the thrust vector direction is not acquired. Once the solid propulsion engine is ignited (at 100 s), the disturbance torque causes a spin around the z-axis as visualised in Fig. 3.

Due to the uncontrolled rotation of the satellite axis, the thrust vector is no longer directed against the flight direction. Hence, the orbit change is no longer carried out as desired. This can also be seen in Fig. 4. While the orbit's perigee is lowered initially, about half way through the firing it stays constant. Due to this, the required perigee altitude is not acquired, not to mention

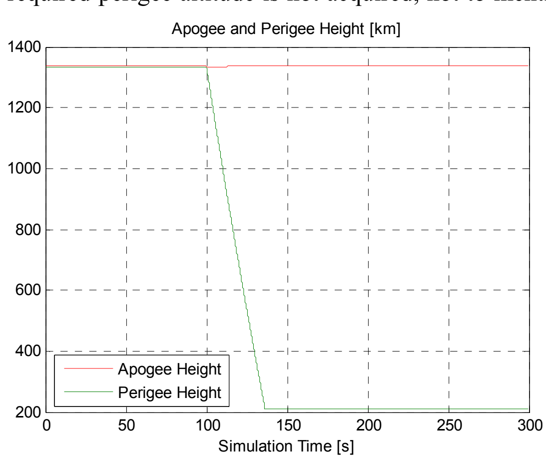

Figure 6. Apogee and Perigee, Initial Rate: 360 deg/s

that the final orbit is not predictable.

Fig. 5 shows the spacecraft rate when the manoeuvre is carried out with an initial spacecraft rate of 360 deg/s. As can be seen, the disturbance torque has negligible effect and the manoeuvre is performed as desired, depicted in Fig. 6.

In order to ensure an initial rate of the required magnitude, however, the AOCS equipment will need to be augmented to ensure a proper rate build-up in such a way that the satellite spin axis is oriented with respect to the spacecraft velocity vector. Practically, this implies the installation of a quasi-parallel AOCS for the spin-stabilised satellite during post-mission disposal in addition to the three-axis controlled AOCS for the nominal mission. Therefore, the concept of using a solid propulsion engine for de-orbiting has been discarded.

4.3 Electric Propulsion

Electric propulsion systems are characterized by their high specific impulse. Nonetheless, they provide low thrust, which results in a time intensive post-mission disposal manoeuvre. They are generally used for continuous thrust manoeuvres when used for orbit control. An electrical propulsion system is generally more complex than chemical propulsion systems and its power consumption is significantly higher. As the propulsion system will be required to operate after the entire scientific mission phased, an electric propulsion system will not be in line with the supplied power of the CryoSat-2 heritage platform or become a driver for the modification.

4.4 Liquid Propulsion

Liquid propulsion systems are widely used on Earth observation satellites. Moreover, spaceflight-proven liquid propellant propulsion system components are widely available. For Jason-CS, mono-propellant propulsion systems as well as bi-propellant propulsion systems have been considered. As can be deduced from Tab. 1 and Fig. 2, a bi-propellant system will require a smaller propellant mass of about 25 per cent for Jason-CS. Depending on the individual thruster angle inclination towards the resulting thrust vector of all thrusters combined, the fuel mass for the post-mission disposal using a hydrazine-based propulsion system is around 100 kg when assuming a 740 kg spacecraft dry mass.

However, bi-propellant propulsion systems require an additional tank for the oxidizer and subsequently a more complex pipe-work. When relying on a mono-propellant propulsion system, the task can be simplified. A modular concept derived from mono-propellant systems flown on previous satellites can be used for Jason-CS. The module is integrated as a fully tested sub-system onto the spacecraft from the rear of the spacecraft (launcher interface). Four thrusters per branch, which are inclined, provide means for both attitude and orbit control, are used per branch. The overall setup is depicted in Fig. 7.

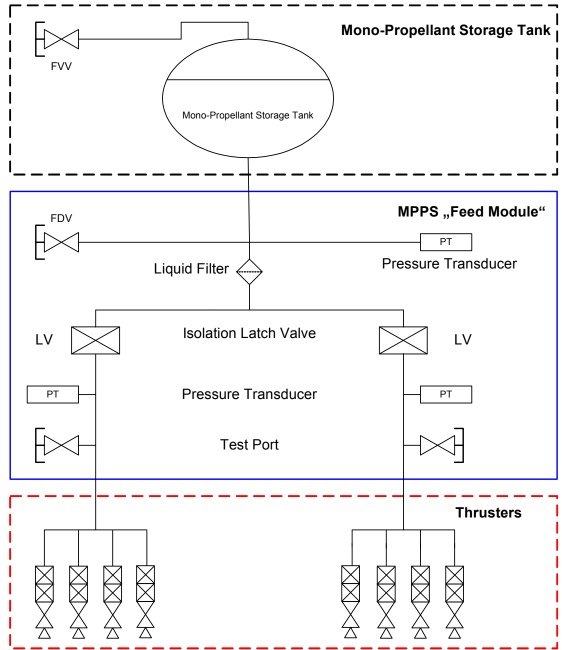

Figure 7. Jason-CS Propulsion Module Schematics

5 CONCLUSION

The trade-offs described in this article have permitted to select a propulsion system suited to support the Jason-CS space debris mitigation requirements. A mono-propellant propulsion system is considered the most suited option thanks to its low system complexity, possibility to accommodate within the Jason-CS spacecraft, heritage and acceptable specific impulse and mass of the propellant.

6 ACKNOWLEDGEMENTS

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