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OSTM/Jason-2 Mission

Assignment 4: Attitude and Orbital Control Subsystem

Group 25

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Acronyms

ACS Attitude Control Subsystem. 11

AOCS Attitude and Orbit Control System. 4, 6, 7

BBQ Barbecue Phase. 6

CNES Centre National d'Études Spatiales. 11

CSS Coarse Sun Sensor. 5–7

DTG Dynamically Tuned Gyroscope. 5

GPS Global Positioning System. 5, 7

GYR Gyro. 5–7

LUM Launch Mode. 6, 7

MAG Magnetometer. 5–7

MLI Multi Layer Insulation. 5

MTB Magnetic Torquer Bars. 6, 7

NASA The National Aeronautics and Space Administration. 11

NOM Nominal Mode. 6, 7, 10

OCM Orbit Control Mode. 6, 7

OSTM Ocean Surface Topography Mission. 5

POD Precise Orbit Determination. 4, 11

PRESTO PRoteus Engineering Simulator for Tests and Operations. 4

PROTEUS Plateforme Reconfigurable pour l'Observation, les Télécommunications Et les Usages Scientifiques. 4–6, 11

RDP Rate Damping Phase. 6

RWS Reaction Wheels System. 7

SADM Solar Array Drive Mechanism. 4

SHM Safe Hold Mode. 6, 7, 10

SPP Sun Pointing Phase. 6

SRP Solar Radiation Pressure. 8–10

SSM Secondary Surface Mirrors. 5

STA Star Tracker Array. 5

STAM Star Acquisition Mode. 5–7

STR Star Tracker. 5–7

THU Thrusters. 7

1 Architecture

Jason-2 is a Nadir-pointing, three-axis stabilized satellite. The spacecraft's attitude is held in place by reaction wheels and magnetic torque rods, while a hydrazine propellant system is used for precise orbital maintenance and station keeping. The system is mounted on the PROTEUS platform, which also has the capability to control the Attitude and Orbit Control System (AOCS) system. Figure 1 shows the positioning of the various components inside the PROTEUS platform.

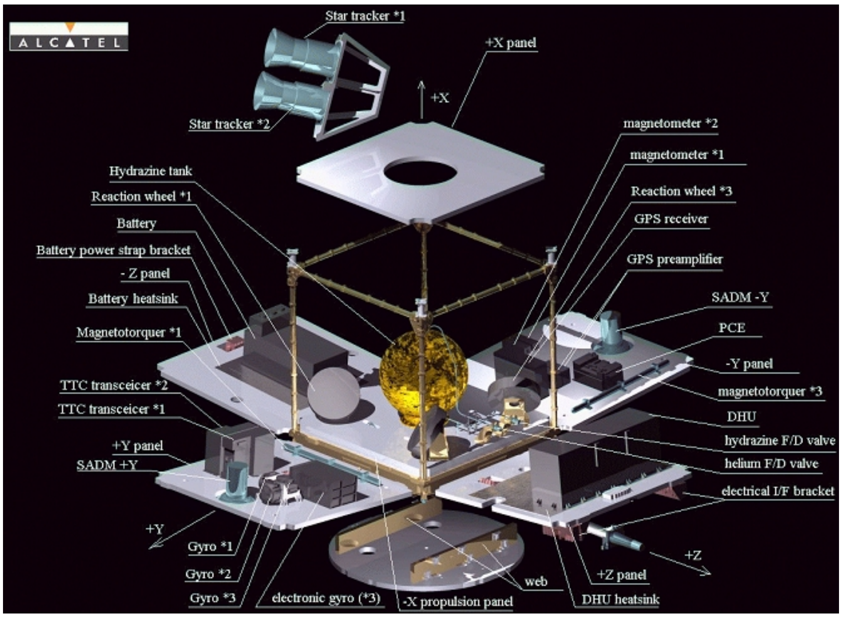

Fig. 1: Proteus Platform Exploded Assembly Drawing

In order to achieve a data acquisition accuracy of 33 mm [3], the Jason-2 satellite must be equipped with a very precise Attitude and Orbit Control System to ensure the high pointing accuracy required. The PROTEUS platform guarantees a stability of 0.0007 deg/s at low frequencies ($< 1\,\text{Hz}$) and high pointing accuracy of $0.05°$, with the addition of a further $0.05°$ as a pointing bias for all axis[13]. Furthermore, Jason-2 is able to determine its orbit position through an integrated GPS system mounted on board, achieving a precision of around 120 meters [6]. Additional systems, such as the Precise Orbit Determination (POD) payloads, are used to improve the satellite's orbit determination capability even more, reaching an impressive sub-meter precision [3]. This accuracy allows Jason-2 to fly during nominal mode in a Nadir pointing position, aligning all its instruments towards Earth.

Due to its particular orbit, the position of the sun with respect to the satellite continuously varies during a full revolution of the spacecraft, requiring careful management of the satellite's power systems. To ensure a continuous power supply, the solar array is equipped with an independent driver known as the Solar Array Drive Mechanism (SADM). This mechanism is coupled with a yaw steering movement, which is used to orient the solar array towards the sun, as depicted in Figure 3, thus minimizing thermal and power losses. Throughout the entire charging operation, the +Z satellite axis remains pointed towards the Nadir axis.

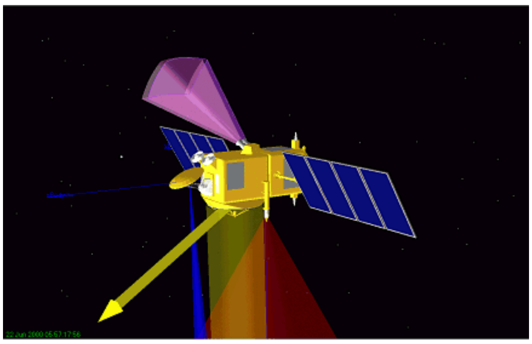

Fig. 2: 3D illustration of Jason2's attitude determination by PRESTO

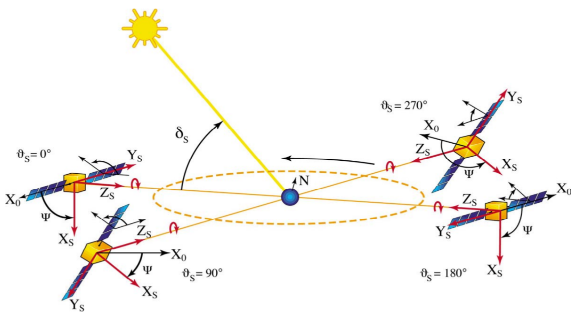

Fig. 3: Yaw steering correction for solar panels

For missions such as the OSTM Jason-2, which are designed to operate almost continuously for at least 3 to 5 years, reliability is absolutely crucial. Therefore, most of the sensors and actuators on board of the satellite have one or more copies implemented for redundancy.

1.1 Sensors

The sensor set of the Jason-2 is composed of two 3-axis star trackers (STR), three 2-axis gyrometers (GYR), two 3-axis magnetometers (MAG), eight coarse sun sensors (CSS), and a GPS for orbit positioning. These sensors include all the copies implemented on the spacecraft for redundancy.

The Star Tracker Array (STA) is composed of two CALTRAC® Star Trackers produced by EMS Techologies Inc., one for nominal use and one for cold redundancy, and the structure containing the star trackers, which is coated in Multi Layer Insulation (MLI) and Secondary Surface Mirrors (SSM) for thermal and radiological protection [6]. The star trackers have a maximum power usage of 14 W and a mass of 3.4 kg each [11], while the whole STA has a total mass of 11.4 kg [6].

The baseline gyro array (GYR) consists of three 2-axis gyros and their respective electronics. Due to the extensive utilization throughout all the Jason-2 control modes, a fully redundant configuration was selected, in which only two gyros out of three operate in nominal mode. The gyros are mounted in an orthogonal configuration on the spacecraft, thus providing redundant measurements on all spacecraft axis. These sensors are the baseline error sensors for all modes except survival mode [13]. The REGYS 3S Dynamically Tuned Gyroscope (DTG) were the sensors used in this mission and were manufactured by Sagem, now known as Safran.

The Coarse Sun Sensor (CSS) set consists of 8 analog solar cells mounted on the spacecraft to provide a 4-pi steradian coverage, allowing 360 x 180° visibility. The analog sun sensors provide a current output proportional to the cosine of the sun angle with respect to the normal of the solar cell. Along with the magnetometers, the CSS are used as the control sensors during survival mode and are used to provide initial attitude determination during the stellar acquisition process [13]. The loss of only one out of eight solar cells is permitted to maintain nominal functioning. The analog solar cells are manufactured by Adcole Maryland Aerospace and are grouped into two arrays of 4 cells each, one for the Pitch axis and one for the Yaw axis, weighing 0.13 kg each [7].

The PROTEUS baseline design contemplated a non-redundant configuration for the magnetometer, but for longer missions, such as Jason-2, it was decided to implement a second magnetometer for cold redundancy and to improve reliability. This sensor provides measurement of the local Earth magnetic field, which is used during survival mode and initial attitude determination. Additionally, it is used during Star Acquisition Mode to provide momentum management control signals. To ensure a proper reading of the Earth's magnetic field, all magnetic torquer bars will not be commanded during measurement; this is done to prevent the formation of a spacecraft-developed magnetic field [13]. The magnetometer used on the Jason-2 satellite was the IM-103 produced by Ithaco Space Systems, which has a power usage of 0.95 W and a mass of 0.231 kg [10].

The baseline sensor set did not include the use of GPS, however it was implemented on Jason-2 to validate its use as an attitude determination system. This was done by comparing the results obtained with the STR with the GPS data during all operational modes [13]. The GPS mounted on the PROTEUS platform was developed and manufactured by Laben, which is now part of Thales Alenia Space.

In Table 1 the main data from each sensor is presented.

	Number	Redundancy	Mass [kg]	Power [W]	FOV	Accuracy	Range
STR	2	1 + 1	3.4	14	22° x 18°	±0.013° for Pitch/Yaw ±0.035° for Roll	/
GYR	3	2 + 1	/	/	/	/	/
CSS	8	1 Failure Allowed	0.13	0	360° x 180° (4-Pi Steridian)	±1° at null ±5° in linear range	500 - 1300 μA
MAG	2	1 + 1	0.231	0.95	/	±4 mG	±600 mG

Table 1: Sensor Data

The total mass of the sensor set is 8.3 kg while the total power usage is 29.9 W.

1.2 Actuators

The attitude control of Jason-2 is done with a set of 4 reaction wheels which are desaturated with 3 magnetic torque bars.

The reaction wheels used in the Proteus Bus are the RSI 8-120/351 manufactured by Teldix (now Collins Aerospace) [9]. They have a torque of 0.075 Nm, a momentum at nominal speed of 8 Nms and operational speed range of ±3500 rpm, they are usable as both momentum and reaction wheels [13] [4]. Four reaction wheels are used in a pyramidal configuration [13] as it allows the loss of a single reaction wheel without affecting the mission, furthermore the satellite can go into its Safe Hold Mode with only one operational reaction wheel [9].

The magnetic torque bars used in the Proteus Bus are the TR60CFR manufactured by Ithaco Space Systems (now Collins Aerospace) [10] [13] [5]. They have a linear dipole of 60 Am^2, a residual dipole of 0.7 Am^2 and a weight of $1.7\,kg$. They have high redundancy as each magnetic torque bar has a nominal and a redundant coil [9].

The orbit control of Jason-2 is done with a set of 4 1N hydrazine thrusters. The 1N thrusters were manufactured by Astrium (now Airbus Defence and Space), with a thrust of $0.32 - 1.1\,N$, specific impulse $200 - 223\,s$ and minimum impulse bit $0.01 - 0.043\,Ns$ [1]. All four thrusters are pointed in the same direction, this provides redundancy as OCM2 is guaranteed to work with the loss of any single thruster and could still work if two thrusters are lost as long as they are diagonal to each other.

	Number	Redundancy	Mass [kg]
Reaction Wheels	4	3 + 1	—
Magnetic Torque Bars	3	1 Redundant Coil on each Bar	1.7
Thrusters	4	1 Failure Allowed (loss of OCM4)	0.29

Table 2: Actuator Data

The total mass of the actuator set is 6.26 kg.

2 Control modes

Jason 2 employs a range of control modes to manage its operations effectively, ensuring mission objectives are achieved while maintaining safety and functionality.

Start-up (LUM)
The Attitude and Orbit Control System is completely empowered during launch. This mode is reffered as Launch Mode (LUM).

Safe Hold Mode (SHM)
This mode is vital for ensuring the satellite's safety and stability in various scenarios and is often reffered to as Survival Mode. SHM is activated in case of detection of anomalous condition or hardware failure. Moreover this mode is used to acquire the initial attitude after separation with the launcher. SHM consists of three main phases: Rate Damping Phase (RDP), Sun Pointing Phase (SPP), and Barbecue Phase (BBQ). The primary goal of SHM is to autonomously achieve a safe attitude configuration, detumbling from launch rotation, and then orienting with the -X satellite axis directed towards the Sun at a mean roll angular rate of -0.25°/s. During the BBQ phase, the satellite slowly rotates in space to achieve an even temperature distribution under solar radiation. In this mode, PROTEUS provides the minimum satellite management needed to support vital functions for diagnosis or anomaly handling. These functions include ground-to-satellite communication, thermal control, battery management, failure management, and reduced payload power 30W. Coarse Sun Sensors and Magnetometers are used for attitude measurement, while Magnetic Torquer Bars generate torque. Additionally, two of the four reaction wheels are utilized to provide gyroscopic stiffness.

Star Acquisition Mode (STAM)
This mode is instrumental in reacquiring fine attitude, position, and time information during the transition from Safe Hold Mode to Nominal Mode. It begins when the satellite is considered safe and is engaged upon receiving a ground command while in SHM. During STAM, payload operations are restricted to verifying instrument behavior, with priority given to housekeeping operations. Typically, the nominal payload is turned off, but if necessary, two of the sixteen power lines can be maintained on to ensure payload power reduced to 30W.

Nominal Mode (NOM)
Nominal Mode facilitates essential satellite management functions and provides generic or specific services required by the payload, where the scientific missions are performed. This mode includes power distribution, commanding, status monitoring via the Mil-STD-1553B bus, precise data acquisition, and fine pointing. Nominal Mode can be engaged in various ways, including transitioning from Star Acquisition Mode upon ground request, automatically when leaving the OCM, or through a NOM reset process initiated by ground request for equipment configuration changes. In this phase, the spacecraft is inertially stabilized using the reaction wheels, while attitude is maintained with a precision in the order of arcseconds through Kalman filtering of Star Tracker and GYR data. During NOM, any excess momentum accumulated in the reaction wheels due to external disturbance torque is dissipated magnetically using Magnetic Torquer Bars (MTB).

Satelllite OCM Modes
Orbit Control Mode (OCM)s are essential for executing orbit adjustments effectively. While their performance and services resemble Nominal Mode, attitude pointing may be affected, and payload functioning may be restricted during significant maneuvers.

- OCM2 is used for moderate orbit adjustments with only two thrusters, ensuring control without major disruptions.
- OCM4 handles significant maneuvers using all four thrusters, as the maneuvers performed require more precision.

The choice of which OCM to utilize depends on mission objectives and the magnitude of the required orbit adjustments. These modes are engaged upon ground request, always transitioning from the Nominal Mode. They automatically transition back to Nominal Mode when the orbit maneuver is completed or transition to Start-up Mode in case of any alarm.

The use of the components of the AOCS in each mode is summarized in Table 3.

	LUM	**SHM**	**STAM**	**NOM**	**OCM2**	**OCM4**
Gyro	OFF	ON (not used)	ON	ON	ON	ON
Star Tracker	OFF	OFF	ON	ON	ON	ON
Coarse Sun Sensor	OFF	ON	ON	ON (not used)	ON (not used)	ON (not used)
Magnetometer	OFF	ON	ON	ON	ON	ON
Reaction Wheels System	OFF	ON	ON	ON	ON	ON
Magnetic Torquer Bars	OFF	ON	ON	ON	ON	ON
Thrusters	OFF	OFF	OFF	OFF	ON (only 2)	ON (all 4)
GPS	OFF	ON	ON	ON	ON	ON

Table 3: Hardware utilization in each mode [13].

2.1 Pointing Budget

In operational mode, the pointing budget is determined by the accuracy requirements of the scientific instruments on board, as discussed in section 1. The precision of pointing must meet a minimum of 0.15° (3σ), and the knowledge of pointing should be within 0.05° of the pointing axis, with a bias of less than 0.05° (3σ) [13]. These requirements are met through the use of star trackers in conjunction with reaction wheels.

In the other two control modes, the pointing accuracy is tailored to align the antennas with Earth and position the Sun shade and solar panels towards the Sun.

The typical values, without perturbations from the payload, are indicated in the table. However, for a more comprehensive evaluation, further simulations and advanced modeling, considering the full Jason-2 system, including both the platform and payload, are necessary.

Frequency Band	**Pointing stability (3σ)**
0.1 to 1 Hz	0.0007 °/s
1 to 5 Hz	0.0003 °/s
5 to 20 Hz	0.01 °/s
20 to 80 Hz	0.0002 °/s
$>$ 80 Hz	0.03 °/s

Table 4: Typical satellite pointing stability using Proteus platform

The final annual report [8] details the pointing precision of the satellite in its final phase. The system achieved an average precision below 0.005 degrees from nadir pointing, compared to 0.07 degrees in the first year [12]. Notably, the requirement was 0.2 degrees, indicating that the AOCS overperformed and improved throughout the mission.

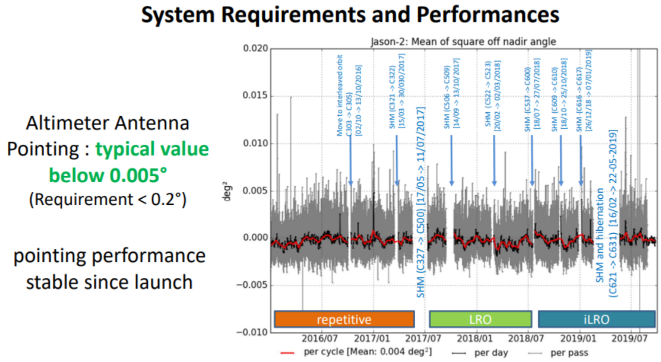

Fig. 4: Pointing Performances in the extended phase

3 Disturbances

A satellite in an Earth-centered orbiting mission is subject, during its operational life, to several environmental disturbance torques. Counter acting the the attitude disturbances coming from the environment is crucial in order to pursue the scientific objectives of the mission and to perform the attitude control's operations.

In the particular case of Jason-2 the disturbances which have a bigger impact on the spacecraft are: Atmospheric Drag, Gravity Gradient, Solar Radiation Pressure (SRP), and Magnetic Moment. The disturbance are calculated in the worst case scenario for each disturbance and then a 100% margin is applied for safety.

Atmospheric Drag

The Atmospheric Drag is a disturbance caused by the collision of the spacecraft with air particles coming from the atmosphere and it is constant for Earth-pointing satellites, such as Jason-2.

To calculate the torque created by the Atmospheric Drag, the following formula is used:

$$T_{AD} = \frac{1}{2}\rho C_D A v^2 \, (c_p - c_g) \tag{1}$$

ρ is the air density at the desired altitude;

C_D is the drag coefficient of the the spacecraft, which is 2.75 at Jason's 2 altitude [6];

v is the velocity of the spacecraft, which is computed knowing the orbital parameters and is equal to 7188.3 m/s;

A is the spacecraft's relative area with respect to the velocity vector, which, knowing the spacecraft's mass-to-area ratio and the satellite's mass [6], can be computed as $\sim 5\, m^2$;

$(c_p - c_g)$ is the difference between the center of aero pressure and the center of mass, which was assumed to be 0.5 m.

Gravity Gradient

Gravity Gradient is a disturbance caused by the effect of gravity on different parts of the spacecraft and it is constant for Earth-pointing satellites.

To calculate the torque created by the Gravity Gradient, the following formula is used:

$$T_{GG} = \frac{3\mu}{2R^3} \, (I_{max} - I_{min}) \, sin(2\theta) \tag{2}$$

μ is the standard gravitational parameter of Earth, which is $3.986 \cdot 10^{14} \, m^3/s$;

R is the radius of the satellite's orbit, which is 7714 km;

$(I_{max} - I_{min})$ is the difference between the maximum and minimum moment of inertia, which is $705.1 - 321.2 = 383.9 \, m^2 kg$ [14];

θ is the max deviation of the Z axis with respect to the local vertical, which is assumed to be 45° as a worst case scenario.

Solar Radiation Pressure
The Solar Radiation Pressure is a disturbance caused by the interaction of solar radiation particles with the surface of the spacecraft and it is cyclic for Earth-pointing satellites.

To calculate the torque created by the Solar Radiation Pressure, the following formula is used:

$$T_{SRP} = \frac{F_s}{c}\, A_s\,(1+q)\,cos(I)\,(c_{sp} - c_g) \tag{3}$$

F_s is the solar constant at spacecraft distance, which is $1367\,W/m^2$;

c is the speed of light, which is $2.997 \cdot 10^8\ m/s$;

A_s is the relative area with respect to the direction of the Sun which is $3.105m^2$ for the main body of the spacecraft and $9.8m^2$ for the solar array [2];

q is the reflectance factor, which is 0.246 for the main body of the spacecraft and 0.098 for the solar array [2];

I is the incidence angle of the solar radiation particles, which for worst case scenario is assumed to be 0°;

$(c_{sp} - c_g)$ is the difference between the center of solar radiation pressure and the center of mass, which was assumed to be 1 m for the solar array and 0.5 m for the main body of the spacecraft.

Magnetic Moment
The Magnetic Moment is a disturbance caused by the interaction of Earth's magnetic field with the spacecraft and it is cyclic for Earth-pointing satellites.

To calculate the torque created by the Magnetic Moment, the following formulas are used:

$$T_{MAG} = D \cdot B \tag{4}$$

$$B = \frac{2M}{R^3} \tag{5}$$

D is the residual dipole of the spacecraft, which is assumed to be $5\,Am^2$ [10];

B is the value of Earth's magnetic field at a certain altitude;

M is Earth's magnetic moment, which is $7.96 \cdot 10^{15} Tm^3$;

R is the radius of the satellite's orbit, which is 7714 km;

Disturbance	Type	Value
Atmospheric Drag	Constant	$1.776 \cdot 10^{-7}\,Nm$
Gravity Gradient	Constant	$5.000 \cdot 10^{-4}\,Nm$
SRP	Cyclic	$5.789 \cdot 10^{-5}\,Nm$
Magnetic Moment	Cyclic	$7.283 \cdot 10^{-5}\,Nm$
	Total	$6.309 \cdot 10^{-4}\,Nm$
	With Margin	$0.0013\,Nm$

Table 5: Mean disturbance values

The total disturbance values are reported in Table 5.

4 Reverse Sizing

4.1 Reaction Wheels

The reaction wheels provide attitude control to the spacecraft by counteracting the external disturbances and performing slew maneuvers and are sized accordingly to perform and accomplish these tasks.

Station Keeping
As seen in section 3, the total disturbances acting on the satellite, considering a 100% margin, are $T_{tot} = 0.0013\,Nm$. The total angular momentum generated by the disturbances can be computed as:

$$h_{dis} = T_{tot} \cdot T_p = 0.0013 \cdot 6720 = 8.736\,Nms \tag{6}$$

Where T_p is the period of the orbit, which is 6720 s.

Knowing that the angular momentum of each wheel at nominal speed is 8 Nms, which means that the reaction wheels saturate in ~ 3 orbits.

Slew Maneuver

Assuming a maximum slew rate of 0.5 °/s (0.25 °/s in SHM) and a slew angle of 180°, as a worst case scenario, the torque needed can be easily computed with the following formula:

$$T_{RW} = N_{RW} \theta_{max} \frac{I_{max}}{t_{min}^2} \tag{7}$$

Where t_{min} is the minimum time to perform the maneuver, knowing the maximum slew rate, in this case $t_{min} = 360\,s$ (720 s in SHM).
The torque required by each reaction wheel to perform a 180° slew at maximum slew rate is $T_{RW} = 0.0684\,Nm$ (0.0171 Nm in SHM), which is lower then the maximum available torque of each reaction wheel (0.075 Nm).

4.2 Thrusters

Slew Maneuver

Assuming the same maximum slew rate of 0.5 °/s (0.25 °/s in SHM) and slew angle of 180° as earlier, the torque needed by the thrusters can be easily computed with the following formula:

$$T_{th} = \frac{I_{max}\,\theta_{max}}{n_{th}\,L\,t_{min}^2} \tag{8}$$

Where L is the distance between the thruster and the centre of gravity, which is 0.845 m [14].
Using 2 thrusters in Nominal Mode the thrust needed by each thruster is 0.0101 N, which is just over the minimum impulse of each thruster (0.01 N) [1]. For Safe Hold Mode and Nominal Mode using 4 thrusters the thrust required by each thruster is too low, thus making any slew maneuver using thrusters not possible.

Reaction Wheel Desaturation

Reaction Wheels desaturation can be performed either passively by changing the solar panel orientation, to exploit Solar Radiation Pressure, or actively by firing a set of thursters or using magnetic torquers. the last option is usually the preferred option, nut in this paragraph we will explore the possibility of using thrusters for desaturation.
This maneuver is usually performed with only 2 thrusters, as it's considered a secondary maneuver, but in this section the possibility of using 4 thrusters is explored as well.

Assuming a required desaturation time of 30 s, the required thrust from each thruster can be computed with the following formula:

$$T_{th} = \frac{h_{RW}}{n_{th}\,L\,t_{des}} \tag{9}$$

The required thrust using two thrusters is $T_{th} = 0.6312\,N$, while using four thrusters is $T_{th} = 0.3156\,N$. These values are acceptable, since both are lower than the maximum amount of thrust which can be produced [1].

Propellant Mass Budget

It is possible to compute the propellant mass by combining the duration of each momentum dump manoeuvre and the total number of dumps required according with the following equation:

$$m_p = n_{des} \frac{t_{des}\,T_{th}}{I_{sp}\,g_0} \tag{10}$$

Where I_{sp} is the specific impulse, which is 220 s [1]. Considering the total mission time of 11 years, the number of desaturations maneuvers to perform is 718, considering a 20% margin.
The total propellant mass which would be used to desaturate the reaction wheels is 0.1 kg using two thruters and 0.05 kg using 4 thrusters, which are acceptable values considering the 28 kg fuel tank.

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