POLITECNICO DI MILANO
DEPARTMENT OF AEROSPACE ENGINEERING

Space Systems Engineering and Operations

AA 2023-2024

Professor: Michèle Roberta Lavagna

OSTM/Jason-2 Mission

Assignment 2: Mission Analysis and Propulsion Subsystem

Group 25

Person Code	Surname	Name
10726898	Juvara	Matteo Giovanni
10717230	Beretta	Beatrice
10705062	Nicolucci	Edoardo
10973691	Separovic	Marko Tomislav
10735389	Orsenigo	Samuele
10728962	Modelli	Simone

March 29, 2024

Contents

Acronyms

CCAFS Cape Canaveral Air Force Station. 5

ESA European Space Agency. 5, 6

iLRO Interleaved Long Repeat Orbit. 4, 6

LEO Low Earth Orbit. 6

LEOP Launch and Early Orbit Phase. 6

LRO Long Repeat Orbit. 4, 6

OCM Orbit Control Manoeuvre. 5, 6, 10

OSTM Ocean Surface Topography Mission. 4

POSEIDON Positioning,Ocean,Solid Earth, Ice Dynamics, Orbital Navigator. 4

PROTEUS Plateforme Reconfigurable pour l'Observation, les Télécommunications Et les Usages Scientifiques. 4–7

PSLV Polar Satellite Launch Vehicle. 4, 5

SRP Solar Radiation Pressure. 5, 6

TOPEX Topography Experiment. 4

TRL Technological Readiness Level. 7, 9

VAFB Vandenberg Air Force Base. 4, 5

WISE Wide-field Infrared Survey Explorer. 5

1 Mission Analysis

The OSTM/Jason-2 satellite lies on the same orbit as Jason-1 and the original TOPEX/POSEIDON mission. This creates a 10-day repeating ground track, in which 127 revolutions are completed during each cycle.

The mean classical orbit elements and other auxiliary data are given in the tables below [8].

Orbit element	Value
Semi-major axis	7741.43 km
Eccentricity	0.0000095
Inclination	66.04 deg
RAAN	116.56 deg
Argument of periapsis	90.0 deg

Table 1: Mean classical orbit elements

Auxiliary Data	Value
Nodal Period	112,4285 min
Repeat Period	9.9156 days
Number of revolutions within a cycle	127
Equatorial cross-track separation	315 km
Orbital speed	7.2 km/s
Ground track speed	5.8 km/s

Table 2: Auxiliary Data

1.1 Orbit Choice

In order to guarantee consistency with objectives and preserve programme continuity, it was decided that the mission had to continue on the same orbit as Jason-1. The most crucial aspect of the orbit is its inclination of 66°, chosen in order to cover most of the globe's unfrozen oceans, as depicted in Figure 1. The orbit is nearly circular to optimize resolution and observability. Its high altitude of 1336 km reduces interactions with the Earth's atmosphere and gravity field to a minimum, thereby facilitating more precise orbit determination and easier station keeping. Additionally, the orbit is designed to pass over two dedicated ground calibration sites: Cap Senetosa in Corsica and the Harvest oil rig platform in California, USA. [8]

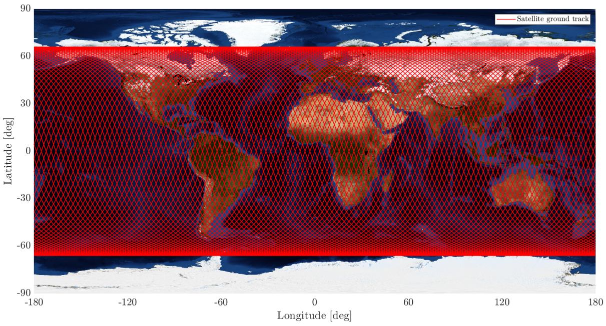

Fig. 1: 2D Ground Track for a 10 day cycle

1.2 ΔV Estimation

In the forthcoming analysis, we aim to quantify the ΔV necessary for the mission's entirety. We operate under the assumption that the mission primarily focuses on orbit maintenance, thereby omitting considerations for end-of-life phases. We do not include the manoeuvres to transition between orbits (LRO and iLRO) in our analysis as they are not pre-programmed but rather regarded as possibilities, akin to the situation observed in the Jason-1 mission.

To reach the desired orbit for the mission and calculate the total ΔV needed to achieve the agreed upon goal, the mission is divided into distinct phases, each of which has it's own criticalities and solutions.

1.3 Launcher Selection

According to the PROTEUS documentation [5], the mounting system for Jason-2 is compatible with various launch vehicles, including Ariane V, Delta II, PSLV, Rockot, Soyuz, Taurus, and many other. To determine the most suitable option for our specific orbit, several factors need to be considered. The primary constraint for launcher selection is the 66° inclination, which varies depending on the launch site. As depicted in Figure 2, our options are limited to the Vandenberg Air Force Base (VAFB), Kourou (Guiana Space Centre), and Russian or Indian launch sites.

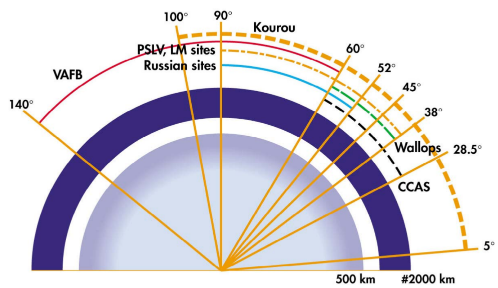

Fig. 2: Achievable orbits VS launch sites and vehicles

Next, the fairing volume compatibility is evaluated, setting the minimum diameter at 1.910 m and a height of at least 2.218 m, excluding the nose cone. This is constrained by the PROTEUS platform dimensions of 1.910 x 0.954 x 1 m and the Jason-2 payload dimensions of 0.954 x 0.954 x 1.218 m.
Finally, the ΔV that each launcher can provide is controlled to make sure it respects the missions minimum requirements. A rough estimate is performed starting from the ideal velocity equation $\Delta V_{\text{ideal}} = \sqrt{\frac{\mu}{R_0}}\sqrt{2 - \frac{R_0}{R_0+h}}$ [14] and the Earth's rotation contribution $\Delta V_{\text{rot}} = V_E \cos(i)$ [14]. Losses are then incorporated in the equation, this is done considering worst-case scenarios for gravity (1.3 km/s) and drag (0.15 km/s) as explained in the Wertz, Everett, Puschell book [21] and implementing the ESA margin guideline of 2% [9]. From this estimate a required ΔV of approximately 10 km/s is obtained.

Launchers	**Success**	**Launch site**	**Usable volumediameter [mm]**	**ΔV compatible**	**$\approx$ Cost [$ milion]**
Ariane 5	100 % (23/23)	Europe/Kourou	4570 or 4800	Oversized [13]	150
Delta 2 7320-10	98 % (163/165)	CCAFS or VAFB	2743	v7320-10: Compatible [19] v7920 and more: Oversized	51-137
PSLV	94 % (17/18)	India/Shriarikota	2900	Not found	30
Rockot	94 % (16/17)	Russia	2520	Compatible [12]	41.8
Soyuz	94 % (1654/1753)	Russia	3395	Oversized [18]	80
Taurus	75 % (6/8)	CCAFS or VAFB	2055	v2210: Little Margin v2110: Compatible [17]	40-50

Table 3: Launcher Data until 2011

For this mission, the selected launcher was the Delta 2-7320-10C, featuring a dual-stage configuration and a 10-foot fairing height (3 meters), enough to accommodate the Jason-2 satellite. While our analysis also considered Rockot and Taurus as viable, cost-effective alternatives; the high success rate and extensive track record of the Delta 2, on top of the continuity with the Jason-1 mission (which was also launched utilizing Delta 2) probably led to its selection by Jason-2's team.

1.4 Orbit Acquisition

Taking into account the selection of the Delta 2 launcher, its manual documents[19] declare an injection precision of ± 9.3 km on perigee altitude and ± 0.05 degrees on inclination (with a 3σ dispersion). In a similar mission named WISE, the maximum error in mean anomaly is estimated to be within ± 7.5 minutes [20]. Considering the worst-case scenario, the ΔV can be estimated using a plane change followed by a shape change and a phasing maneuver of the duration of more or less 70h to rendezvous behind Jason-1.

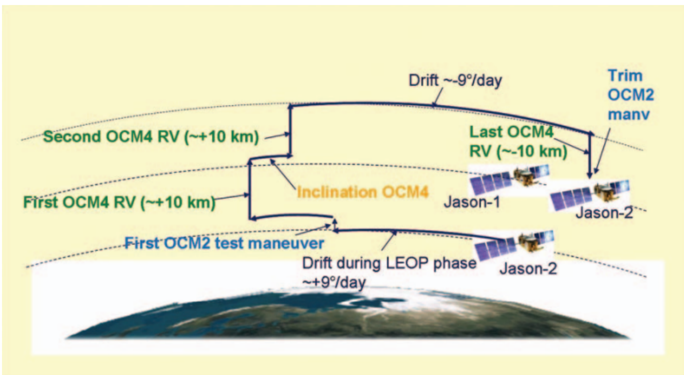

Fig. 3: Orbital acquisition strategy

In this way, a total $\Delta V = 13.4$m/s is obtained, to which a 5% margin has to be added according to the ESA margin guidelines [9]. So the final requirement for this phase is $\Delta V = 14.07$m/s, which corresponds to $\Delta m = 3.28$kg of hydrazine used.

This value is a gross estimate and may not accurately reflect reality. During the Jason-2's orbit acquisition, outlined in Figure 3, orbit corrections and testing persisted for 20 days, which resulted in the consumption of 3.5 kg of hydrazine through three main maneuvers utilizing four thrusters each (OCM4), with a magnitude of approximately 3 m/s for each maneuver, alongside other maneuvers of lesser significance [11].

1.5 Station Keeping

A satellite orbit slowly decays over time due to air drag, Earth's non-homogeneous gravity field, Solar Radiation Pressure (SRP), and smaller perturbations. Periodic maneuvers are required to keep the satellite

in its nominal orbit. Each orbit maintenance maneuver is performed using only two thrusters (OCM2) to minimize the amount of fuel consumed.
In the 2009 mission overview [10] it is outlined how the station keeping maneuvers are performed approximately once every 2 months in order to keep the satellite inside a ± 1 km range from it's nominal altitude. By considering a simple Hohmann transfer between the nominal orbit and the extremities of this range, and applying this manuever once every 2 months; the total yearly ΔV is 2.78 m/s ($\Delta m = 0.625 kg$). To this value a twofold margin has to be applied, as per ESA Guidelines [9], which brings the total yearly ΔV to 5.56 m/s, or about 1.29 kg of fuel consumed every year.

1.6 Margins and Comparison

	ΔV [m/s]				Mass [Kg]		
	Reverse Engineering			True Data			
	Calculated	Margin	Total		Reverse Engineering	True Data	Remain mass
Launcher Selected	Delta II, Tauros and Rocktot			Delta II			**28**
Orbit Acquisition	13.4	5 %	14.7	15.01	3.28	3.5	**24.5**
Station Keeping	2.78 (for year)	100 %	61.18 (for 11 years)	23.80	14.20	5.5 (estimeted $\approx$ 0.5 kg/year)	**19**
Interleaved	not considered		0	11.91	not considered	2.73 (estimeted - not found)	**16.27**
LRO			0				
i-LRO			0	2.41		0.55	**15.72**
Fuel Depletion			0	50.16		11.32	**4.4**
END	4 kg (for emergency)		18.02	19.82	4.00	4.4	**0**
		Total	**93.27 m/s**	**123.11 m/s**	**21.48 Kg**	**25.27 Kg**	

Table 4: Mass and Velocity Comparison

As highlighted in the table above, the actual station keeping costs were much lower then what was expected at the beginning of the mission, this left room for other maneuvers once the initial mission of the satellite was ended, such as the movement on the LRO and successively on the iLRO. Another aspect which wasn't initially considered is the remainance of fuel once the satellite was set to be decomissioned, this was solved by depleting most of the fuel left, keeping a small quantity for emergencies. This small remainance is kept for any emergency maneuver needed, such as avoiding any collision with active satellites, and amounted to approximately 4 kg of hydrazine. All of the real data of the satellite was taken from the 2018 Mission Status Report from the CNES [7].

2 Propulsion System

2.1 Propulsion System Overview

The Jason-2 propulsion system is part of the Plateforme Reconfigurable pour l'Observation, les Télécommunications Et les Usages Scientifiques (PROTEUS). PROTEUS is a 3-axis stabilized platform designed for satellite missions in Low Earth Orbit (LEO) and has a mass of $265 kg$ [15]; it includes the solar panels, attitude control systems, propulsion systems, data storage, and telecommunication systems for the satellite it services. The PROTEUS platform was used for all Jason-1, Jason-2 and Jason-3 missions.
The propulsive system of PROTEUS is a monopropellant system, which means that the energy source is chemical and stored in a single compound, hydrazine in this case . Specifically PROTEUS's propulsion system is a pressure-fed blowdown system, in which the pressurising gas, Nitrogen N2, is stored in the same tank as the propellant from which is divided through a diaphragm. The system is composed of four 1N thrusters, which decompose the hydrazine trough a catalyst. The chemical energy in the hydrazine bonds is transformed in enthalpy, trough the decomposition reaction with the catalyst, and then in kinetic energy through the nozzle.

The PROTEUS propulsion system has to provide a ΔV for three main functions:

- **Orbit Injection Correction**: the launcher injects Jason-2 in a Launch and Early Orbit Phase (LEOP) 10 km below the nominal orbit with all other orbital parameters as close to nominal as possible. The propulsion system shall provide the ΔV needed to correct any deviated orbital parameters.
- **Station Keeping**: due to SRP, air drag, and J2 perturbances the orbital parameters deviate in time. The propulsion system shall provide the ΔV needed to keep the spacecraft on the nominal orbit.
- **End of Life Maneuvers**: the propulsion system shall provide the ΔV needed to perform any extra maneuvers at the end of its mission life, such as making room on the orbit for its successor or deorbiting the satellite.

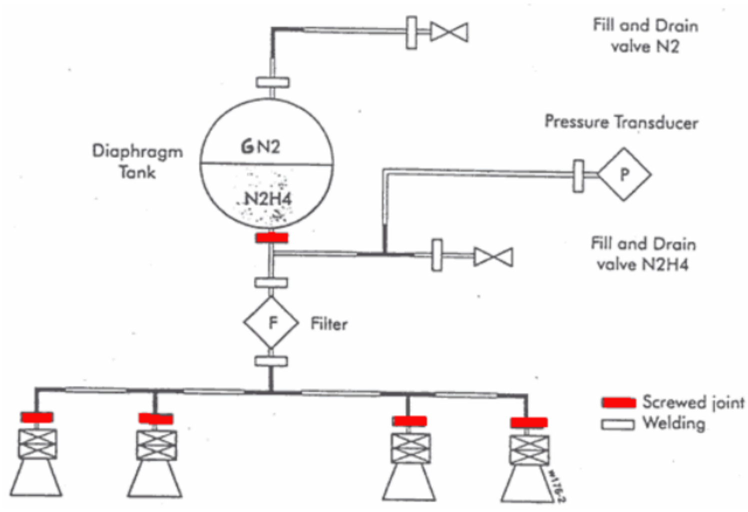

Fig. 4: Schematic of PROTEUS's Propulsion System

Due to the nature of these these requirements the propulsion system in this case can be categorized as a ***secondary*** propulsion system, even if it is the only one present on the satellite, due to the low ΔV it has to provide. The ***primary*** propulsion system is technically the launcher itself, which provides all the ΔV needed for injection (roughly three orders of magnitude larger).

3 Propulsion System Sizing and Analysis

Due to its very high Technological Readiness Level (TRL) and its successful implementation in the Jason-1 mission, a hydrazine monopropellant blowdown propulsion system was selected for the Jason-2 mission. This system configuration has been used extensively in the past and has proven to be economical, versatile, and reliable.

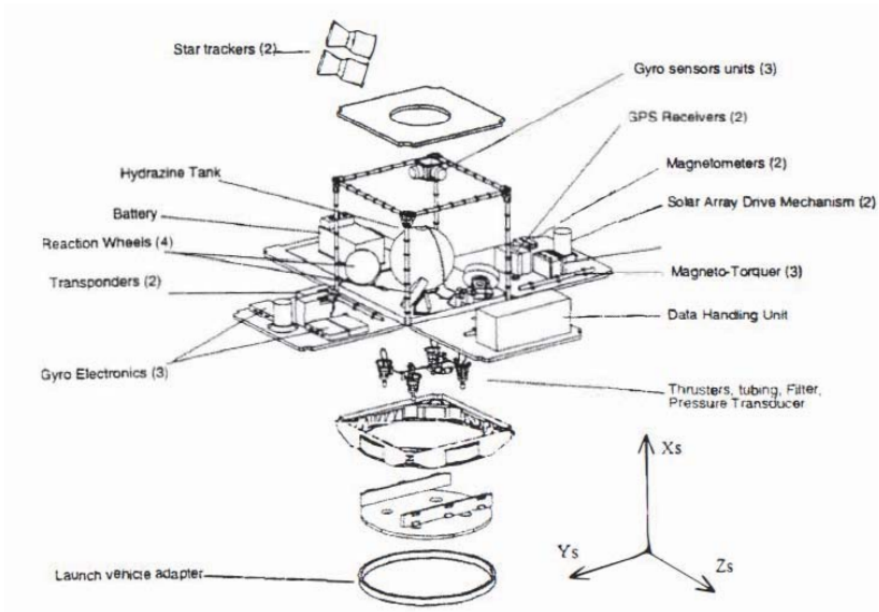

Fig. 5: PROTEUS internal view [2]

The sizing present in the following sections is based on the thrusters data [1] used on the PROTEUS platform. The costants used in the following sections are: $g0 = 9,80665 m/s^2$, $R_{N2} = 296,8 J/kgK$.

3.1 Propellant Selection and Masses

Hydrazine was specifically selected because of its stability, high performance in the form of specific impulse and clean exhaust.
The dry mass $m_{dry} = 477kg$ is obtained from the subtraction of the launch mass of $505kg$[15] and the true propellant mass of $28kg$[15]. Considering the obtained m_{dry}, the ΔV requirement of $123m/s$ (calculated from launch and propellant masses [15] through the Tsiolkovsky rocket equation) and considering that the thrusters have a $I_{SP} = 220s$, the required fuel load m_f can be calculated using the Tsiolkovsky rocket equation:

$$\Delta V = I_{SP} \cdot g_0 \cdot ln \left(\frac{m_{dry} + m_{prop}}{m_{dry}} \right) \tag{1}$$

$$m_{prop} = 1.03 \cdot 1.02 \cdot 1.005 \cdot m_{dry} \cdot \left(e^{\Delta V / I_{SP} \cdot g_0} - 1 \right) \tag{2}$$

An ullage of 3% is considered along with residuals of 2% (MAR-MAS-080) and an extra 0.5% for loading uncertainties. The required fuel load is then $m_{prop} = 29,55kg$.

3.2 Tank Sizing and Masses

A blowdown titanium spherical tank is implemented for the mission. The tank is equipped with a diaphragm made from EPN-40 (EPDM type silica-free rubber [3]), as it's compatible with hydrazine's corrosiveness and has a high specific strength. Nitrogen is used as pressurant due to the fact that it's less prone to leakage with respect to Helium and because it's relatively cheap.

To calculate the required tank volume and pressurant mass, the ideal gas model is assumed and the propellant is assumed to be kept at a constant temperature of $T_{tank} = 298K$ [6]. At the given temperature, hydrazine has a density of $\rho_{prop} = 1003,6kg/m^3$ [16] and as such the total volume of hydrazine is $V_{prop} = 29,44L$.

The thruster has an inlet pressure range of $22bar - 5.5bar$, considering feed loses of $\Delta P_{feed} = 0.5bar$, the tank pressure range should be between $P_i = 22.5bar$ and $P_f = 6bar$. This produces a blow down ratio of $B = 3.75$. The volume of gas can be calculated with the following formula assuming isothermal flow and taking into account an extra 10% margin of unused propellant volume (MAS-CP-010).

$$V_{gas,i} = V_{prop} \left(\frac{1}{B-1} + 0.1 \right)$$

The obtained value is $V_{gas,i} = 13,65L$.
The mass of pressurant can be calculated, including a 20% margin, using the following formula:

$$m_{gas} = 1.2 \cdot \frac{P_i \cdot V_{gas,i}}{R_{N2} \cdot T_{tank}}$$

An $m_{gas} = 0,417kg$ is obtained.

The required tank volume can be calculated, taking into account an extra 1% extra volume taken by the balder, with: $V_{tank} = 1.01 \left(V_{gas,i} + V_{prop} \right)$. And $V_{tank} = 43,52L$ is obtained. The radius of the tank is $r_{tank} = 218,2mm$.

The properties of Ti-6AI-4V are: $\rho_{tank} = 4430kg/m^3$ and $\sigma = 1100MPa$ [4]. Assuming a thin wall approximation, the required thickness of the tank walls can be calculated with the following formula:

$$t_{tank} = \frac{f_s \cdot P_i \cdot r_{tank}}{2\sigma}$$

Considering a safety factor of $f_s = 2$, a value of $t_{tank} = 0,446mm$ is obtained. The final external diameter of the tank is $d_{tank} = 437,3mm$. Finally the mass of the tank can be obtained through the formula:

$$m_{tank} = \rho_{tank} \cdot \frac{4}{3}\pi \cdot \left(\left(r_{tank} + t_{tank} \right)^3 - r_{tank}^3 \right)$$

Which gives a mass of $m_{tank} = 1.18kg$.
The values obtained can be compared with the actual values for the mission [3]:

	Calculated	Actual
m_{prop}	$29,55kg$	$28kg$
V_{tank}	$43,52L$	$37,5L$
d_{tank}	$437,3mm$	$420mm$
m_{tank}	$1,185kg$	$3,9kg$

3.3 Feeding Lines

As visible in Figure 5, the feeding lines of the system are in an "H" configuration connected in the middle to the tank and at the vertices to the four thrusters. This allows for a compact, economical and not so complex configuration, considering the absence of valves and redundancies in the lines. This absence can be justified by the fact that the system is classified as a secondary propulsion system and that the system's TRL is very high.
Along the feeding lines a pressure transducer is placed, this is useful for knowing the amount of propellant at any given time in the tanks. Indeed knowing the tank volume $V_{tank} = 37,5L$, the temperature $T_{tank} = 298K$ [6], the density of the propellant $\rho_{prop} = 1003.6kg/m^3$ [16] and the measured pressure P_m:

$$V_{gas} = \frac{m_{N2} \cdot R_{N2} \cdot T_{tank}}{P_m}$$

Having the current N2 volume, the propellant volume can be calculated:

$$V_{prop} = V_{tank} - V_{gas}$$

From which the mass of propellant is obtained:

$$m_{prop} = \rho_{prop} \cdot V_{prop}$$

At last, knowing the thruster's specific impulse $I_{sp} = 220s$ and dry mass $m_{dry} = 477kg$, the available ΔV can be obtained through the Tsiolkovsky equation (1).

3.4 Thrusters

This propulsion system makes use of four 1N thrusters placed in a square configuration angled parallelly to the X axis of the satellite, as shown in Figure 5. This configuration is convenient as attitude control is not required from the system and firing two thrusters at opposite ends or four thrusters at a time does not change the attitude of the satellite. Since the thrusters are not gimbaled, the satellite has to be rotated by the attitude control system to provide trust in a specific direction . This configuration has the advantage of having a low complexity and cost, but has as a downside the fact that, while the propulsion system is providing thrust, the satellite can't be in its nominal data-acquisition mode.

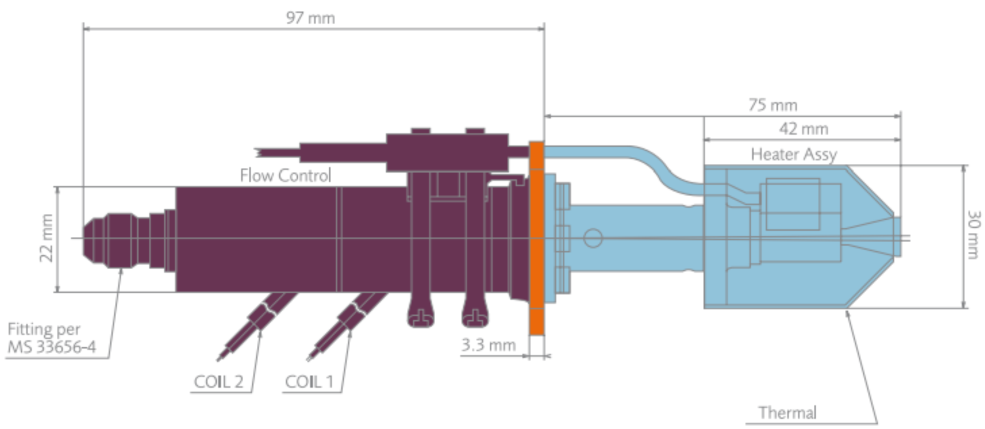

Fig. 6: 1N Thrusters schematic [1]

As said before, the thrusters are usable in configurations of two or four at a time respectively as OCM2 and OCM4 modes, this allows for two different thrust options. These thrusters are also not throttled, due to a lack of a valve for flow control in the feeding line. The thrust diminishes in direct correlation to the mass flow of the propellant. The less propellant is in the tank, the lower the pressure is in the system, the lower is the mass flow, due to the blowdown structure of the system.

The thrusters have two independent valves for redundancy, as well as a heater to ease the start up and is designed for both, long term steady state and pulse mode operations [1] .

References

[1] 1N Monopropellant Hydrazine Thruster.
https://www.space-propulsion.com/spacecraft-propulsion/1n-hydrazine-thruster.html.

[2] ICSO 1997 - A new European small platform: Proteus and prospected optical application missions .

[3] Rafael space propulsion catalogue.
https://www.rafael.co.il/wp-content/RAFAEL-SPACE-PROPULSION-2021-CATALOGUE-2.pdf.

[4] Ti 6al 4v (grade 5) titanium alloy data sheet.
https://kyocera-sgstool.co.uk/titanium-resources/ti-6al-4v-grade-5-titanium-alloy-data-sheet/.

[5] CNES & ALCATEL. Proteus user's manual, 2004.

[6] Pradeep Bhandari. Robust Thermal Control of Propulsion Lines for Space Missions. Technical report, Jet Propulsion Laboratory, California Institute of Technology, 2016.

[7] CNES C. Marechal. Jason-2 Project Status, 2018.
https://ostst.aviso.altimetry.fr/fileadmin/OPEN-03-Marechal.pdf.

[8] CLS, CNES, NASA & EUMETSAT. OSTM/Jason-2 Product Handbook, 2011.
https://www.ospo.noaa.gov/Products/documents/J2-handbook.pdf.

[9] ESA. Concurrent design facility studies standard margin philosophy description, 2017.

[10] CNES G. Zaouche. OSTM/Jason-2 Mission Overview, 2009.
https://www.aviso.altimetry.fr/fileadmin/documents/OSTST/2009/oral/Zaouche.pdf.

[11] J. Perbos G. Zaouche. OSTM/Jason-2: Assessment of the System Performances (Ocean Surface Topography Mission: OSTM). *Marine Geodesy*, 33(sup1):26–52, 2010.

[12] EUROCKOT Launch Services GmbH. Rockot User's Guide, 2011.
https://www.eurockot.com/wpcontent/uploads/2012/10/UsersGuideIss5Rev0web.pdf.

[13] Ariane Space Group. Ariane 5 User's Manual, 2011.
https://www.arianespace.com/wp-content/uploads/2011/07/Ariane5-Users-Manual-October2016.pdf.

[14] Arif Karabeyoglu. Launch Trajectories, Advanced Rocket Propulsion course of Stanford University , 2019. https://web.stanford.edu/cantwell/AA284A-Course-Material/Karabeyoglu.pdf.

[15] NASA. Ocean Surface Topography Mission/ Jason 2 Launch - Press Kit/JUNE 2008.
https://sealevel.jpl.nasa.gov/internal-resources/389/.

[16] O'Neil. *The Merck Index - An Encyclopedia of Chemicals, Drugs, and Biologicals.* Royal Society of Chemistry, 2013.

[17] Northrop Grumman Orbital. Taurus Launch System, 1999.
http://www.georing.biz/usefull/Taurus-User-Guide.pdf.

[18] Arian Space. Soyuz User's Manual, 2012.
https://www.arianespace.com/wp-content/uploads/2015/09/Soyuz-UsersManualMarch2012.pdf.

[19] ULA. Delta II, Payload Planners Guide, 2007.
https://www.ulalaunch.com/docs/default-source/rockets/deltaiipayloadplannersguide2007.pdf.

[20] ULA. Wide-field Infrared Survey Explorer, 2009.
https://www.ulalaunch.com/docs/default-source/news-items/dii-wise-mob.pdf.

[21] J.R. Wertz, D.F. Everett, and J.J. Puschell. *Space Mission Engineering: The New SMAD.* Space technology library. Microcosm Press, 2011.