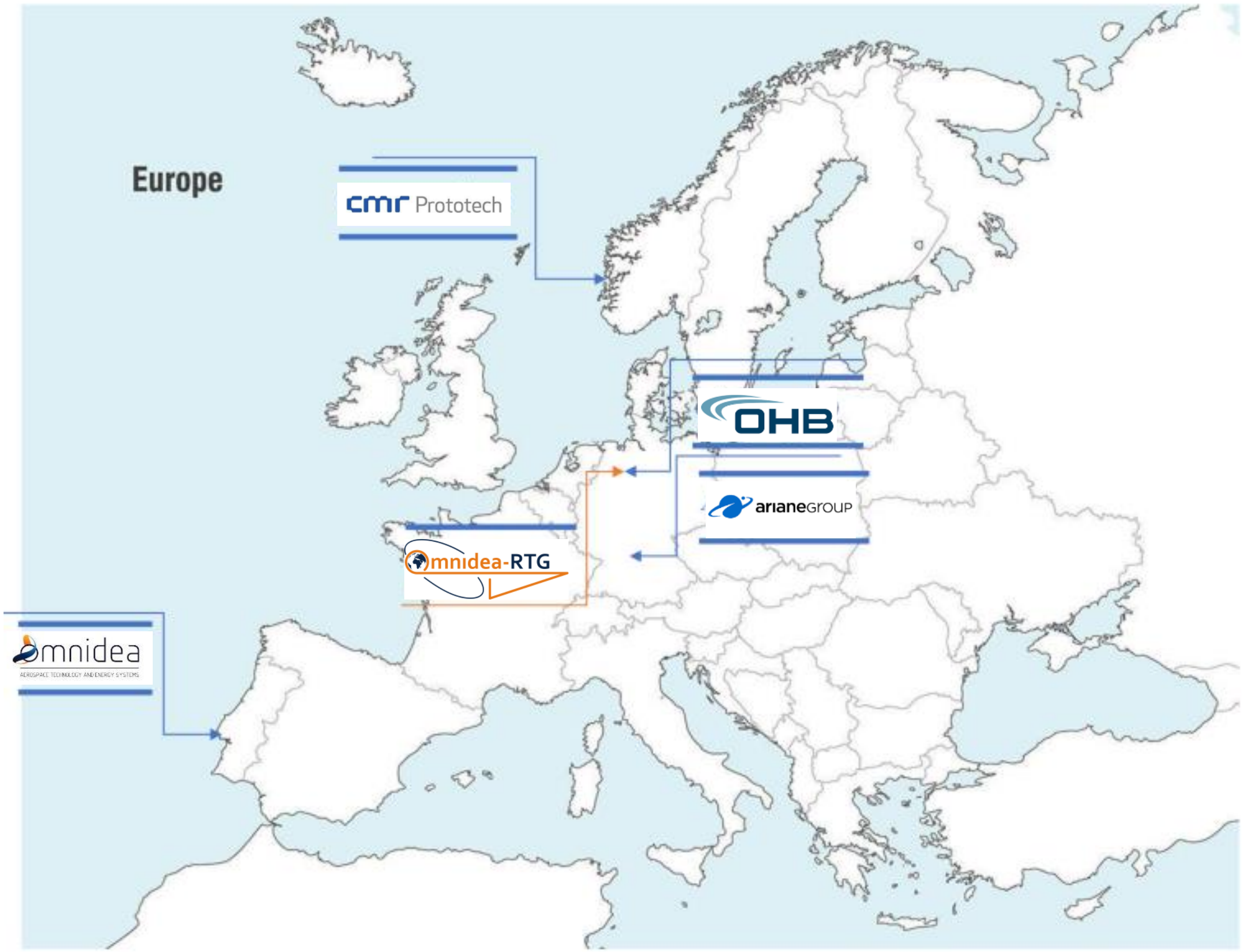
Electrolysis based water propulsion for a future 1-ton class LEO sat.

Final WPS project presentation – April 2021

Consortium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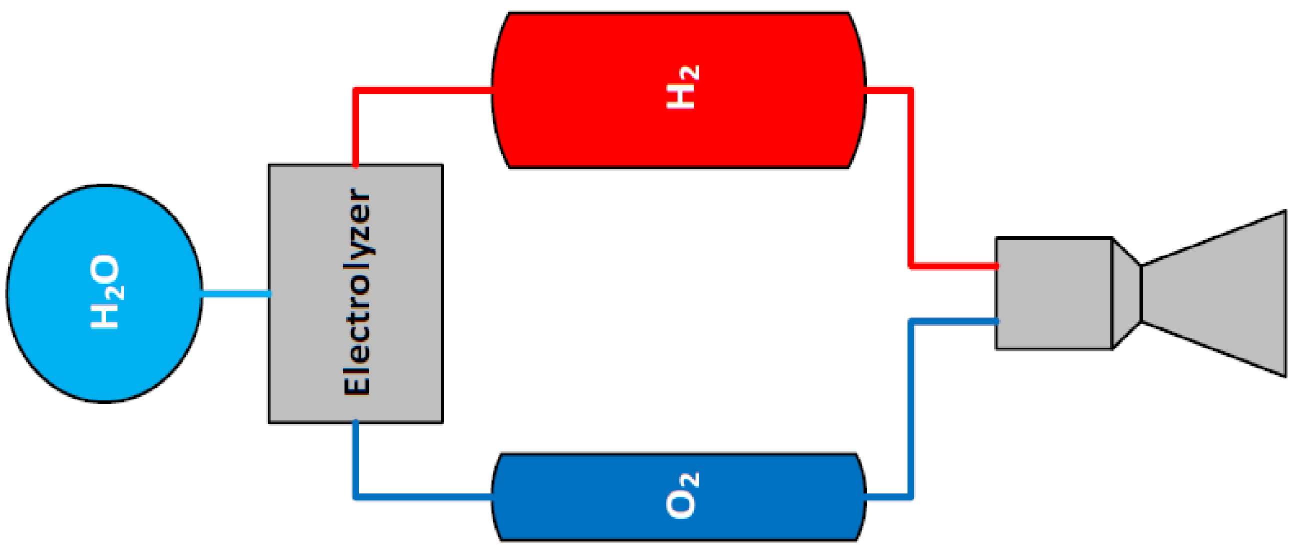
3. Mission Analysis and selection – what mission profile did we select from ESA's SoW and why?
4. Electrolyser studies and trade-off – the ELY is the "component" of this WPS; what technologies exist? Where has ESA focused its development until now? what are the advantages and disadvantages of each tech?
5. Thruster performance – the WPS tested a GH2-GO2 thruster; in view of theoretical Isp (vs. OF ratio, and Temp), what performance did we get in Continuous and Pulse firing? Did it support the decision to fire at stoichiometry?
6. Propulsion architecture and performance – how did we arrive at the mass and power budgets? What is included in terms of de-orbit? How does the architecture look and how does it compare to monoprop?
7. Roadmap and recommendations – context for a roadmap elaboration; assumptions for TRL estimations of individual components - how does the roadmap correlate to assumptions on LAF vs. CVF?

Introduction and definition

Electrolysis based Water Propulsion is defined as Space propulsion, with liquid water as propellant and electrolysis being performed for GO2 and GH2 production, with electrical power produced on-board.

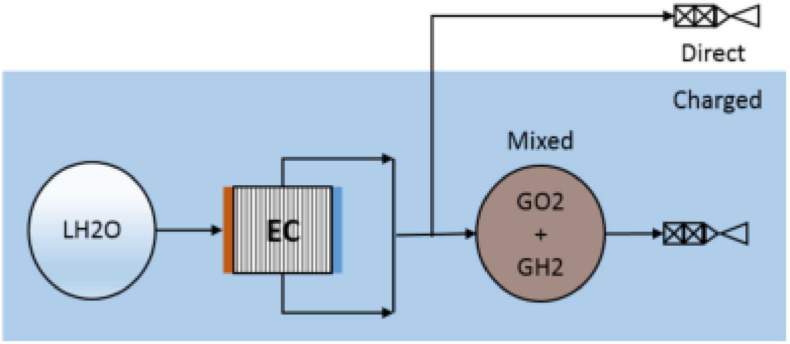
and the terminology “Electrolyses based Water Propulsion used to distinguish from other forms of water propulsion, such as steam rockets (ARCA Space), water arcjets or water electrothermal thrusters (AVS, Comet).

The basic working principle of water propulsion is as per the figure below



Configuration and synergies (1/2)

Multiple options were screened, such as mixed oxygen and hydrogen storage with posterior firing or even, potentially, direct firing without storage.

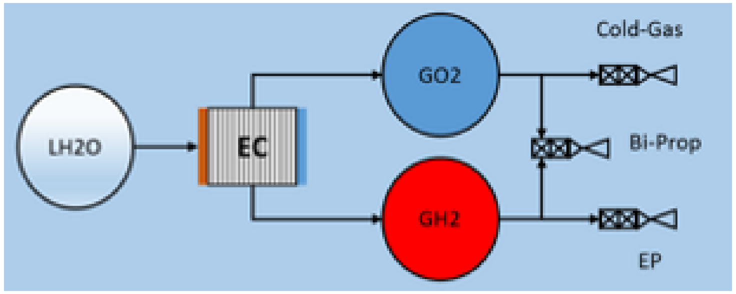


Direct firing would require 4-5kW for a 1N thruster, or ≈10kW for the 2N thruster under development

Mixed storage could bring an advantage in terms of mass penalty but due to safety concerns was abandoned; likely also, with totally separate gas feed lines, it is easier for ArianeGroup to implement the thruster film cooling strategy.

Several options arise if non-stoichiometric firing is assumed:

- Venting excess O2 gas through a cold gas thruster, or
- Should excess electric power be available, then complementary electric propulsion such as a resistojet or an arc-jet could be used. This option would have the advantage of increased Isp.

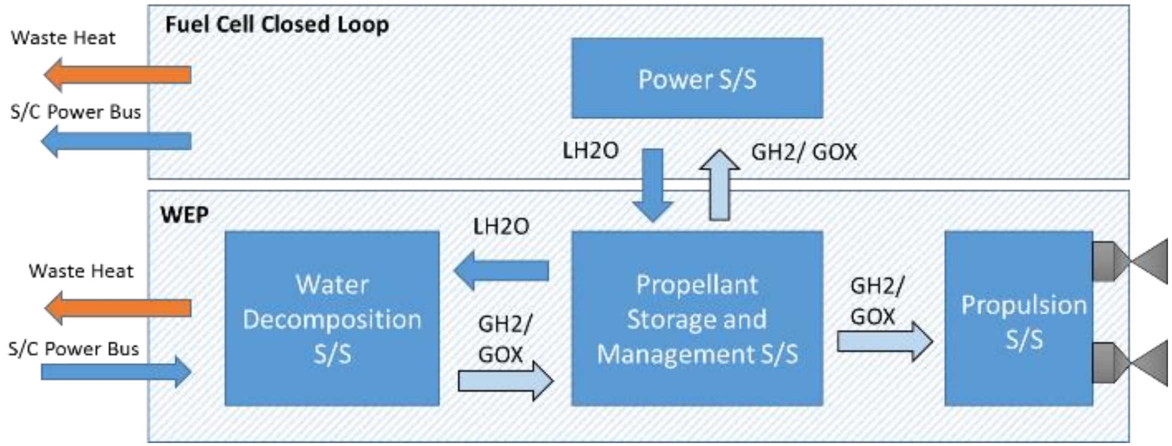


From early on ArianeGroup had a purpose for the “extra” O2, using it to achieve a relatively high Isp at stechiometry without allowing the nozzle to melt.

Venting excess O2 through a cold gas thruster does not provide a significant gain (much lower isp for no increase in max. thrust vs. bi-prop thruster)

Using electrical propulsion along the bi-prop thruster will now be investigated in a new ESA AO (10609)

Configuration and synergies (2/2)



During the project it was briefly equated if it would make sense to use a fuel cell (i.e. beyond the electrolyser) in order to investigate potential synergies of performing on-board thermal management also through the water system.

If the use of a fuel cell is combined with a thermal management using a water fluid loop, the hot water coming out of the fuel cell may be routed directly to radiators to be cooled down before entering the tank or the rest of the piping. A reasonable amount of water onboard may also be used as a heat sink for peak heat dissipation demands, such as the operation of the fuel cell during an eclipse period (in case of S/C batteries being replaced). However, this capability would be reduced as water is expended as propellant.

Some considerations on 2 possible layouts layouts:

1. electrolyser and fuel cell as separate units. This allows each of them to be optimized for their function, but requires doubling the electrochemical stacks.

2. Use of a single ***unified regenerative*** fuel cell, in which the same electrochemical unit performs simultaneously the role of a fuel cell and an electrolyser; in this case there is an efficiency (and lifetime) penalty to pay, plus potential design compromises

Mission analysis and selection (1/2)

3 main mission types were studied: LEO, GEO and Science Exploration.

Parameter	LEO
Wet mass (at BoL)	1250 kg
Operating lifetime	10 years
Orbit	814 km x 98.6º
Thermal capacity	1 kW
Total Delta V	690 m/s
Thrusters - main	200 N
Thrusters - AC	10 N
Available power (de-orbit)	1.96 kW
Available power (on station)	2 kW
Longest single eclipse time	989 s

Parameter	Exploration
Wet mass (at BoL)	3700 kg
Operating lifetime	7 years
Thermal capacity	1 kW
Delta V – DSM	350 m/s
Delta V – MOI	1500 m/s
Delta V – scientific mission	250 m/s
Total Delta V	2100 m/s
Thrusters - main	420 N
Thrusters - AC	10 N
Available power	2 kW

Parameter	GEO
Wet mass (at BoL)	6500 kg
Operating lifetime	15 years
Orbit	35786 km x 0º
Initial orbit	300 x 35786 km x 5º
Inclination correction	280 m/s
Thermal capacity	11.5 kW
Delta V – transfer	1500 m/s
Delta V – attitude control	90 m/s
Delta V – station keeping	900 m/s
Delta V – momentum wheel unloading	150 m/s
Delta V – de-orbit	15 m/s
Total Delta V	2655 m/s
Thrusters - main	600 N
Thrusters - AC	10 N
Available power	18 kW
Longest single eclipse time	4170 s

Mission analysis and selection (2/2)

Parameter	Weight	Score		
		LEO	GEO	Mars
Mass budget	2	1	1	-3
Lifetime	1	0	0	0
Resources	1	1	1	-1
Other subsystems	1	1	-1	0
Spacecraft layout	1	-1	0	-3
Mission profile	2	0	-1	0
Mission frequency	2	1	0	-1
Total		**5**	**0**	**-12**

LEO provides the easiest implementation case, with only spacecraft layout being slightly affected due to the need to accommodate 1 electrolyser and 2 gas tanks.

Mass budget is positively impacted due to the significant reduction in propellant weight while

mission frequency is positively impacted due to the much quicker fuelling with water,

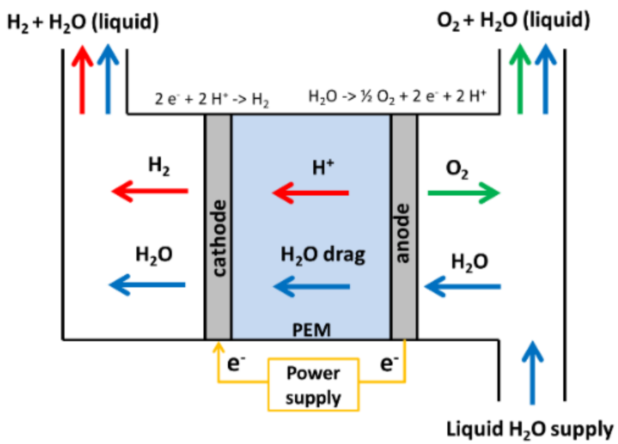
Resources also benefitting from the much lower costs for the fuelling itself and costs of the loading operation.

The GEO mission has an overall neutral score since slight advantages on mass budget (slighter higher Isp plus the lower fact that, vs bi-prop, WPS has a single prop. tank. Resources are outweighed by the significant longer insertion time;

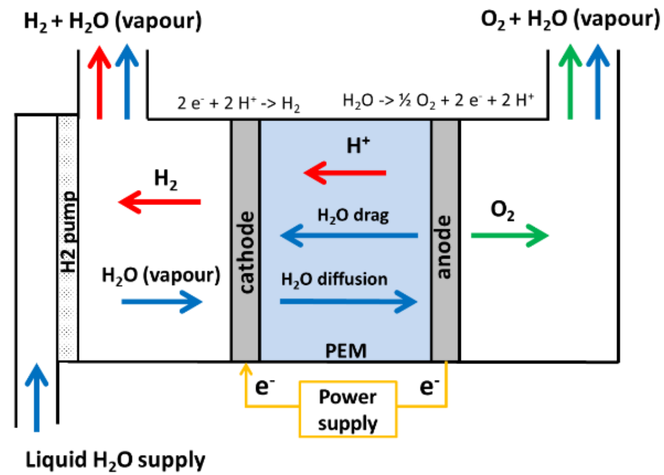
In conclusion: while WPS can be used for GEO/GTO, advantages are cancelled out by the longer insertion time, with WPS taking much longer than storable bi-propellant to perform GTO while lacking the Isp advantage of HET/GIT electric propulsion.

For the Mars mission, the impacts are all negative; by far the biggest issue is that, due to the very low volumetric density of GH2 (even @100bar) performing planetary capture is nearly impossible; the same problem of volumetric density is reflected in a spacecraft with a big mass budget penalty (extra weight of colossal gas storage tanks) and very bad spacecraft layout (to accommodate the gaseous tanks). This would imply negatively also on resources and mission frequency.

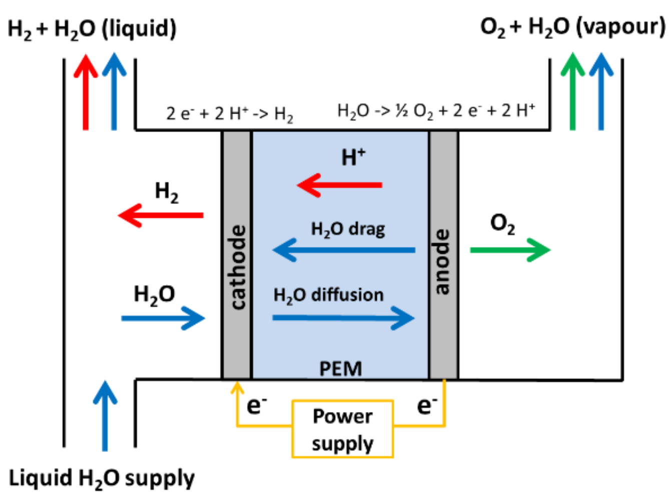
Eletrolyser studies and trade-off (1/4)



Anode Liquid Feed concept (LAF)



Cathode Vapour Feed (CVF, aka Static Water Feed)



Cathode Liquid Feed concept (LCF)

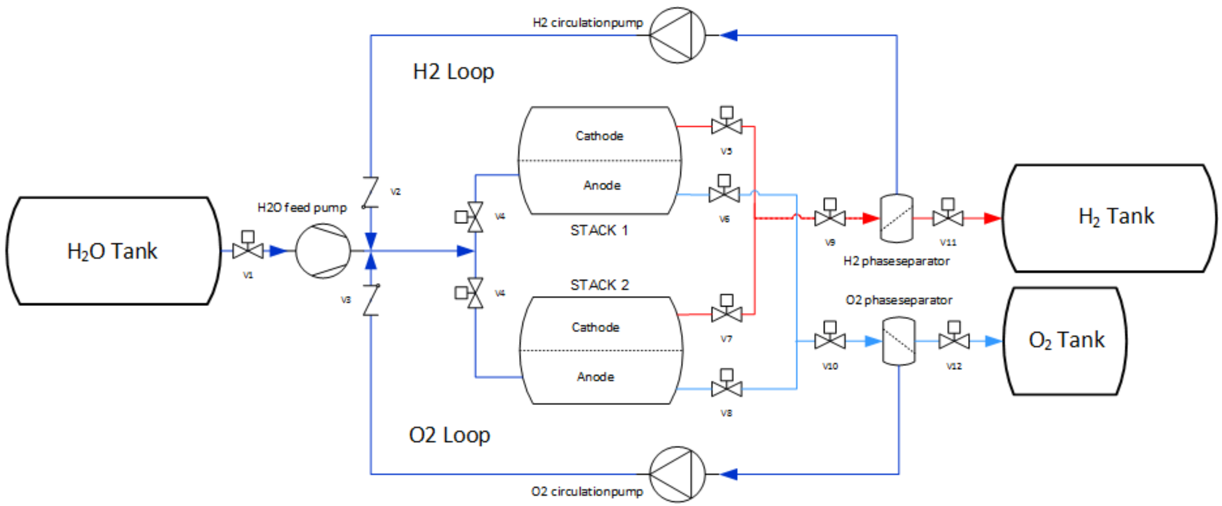
3 main mission types of pressurized electrolysers were studied and trade-off.

LCF was abandoned for study from early on, due to the fact that it suffers from most of the problems of LAF (without any significant performance gains) while offering none of the advantages of CVF/SWF.

Eletrolyser studies and trade-off (2/4)

2 types conceptually designed:

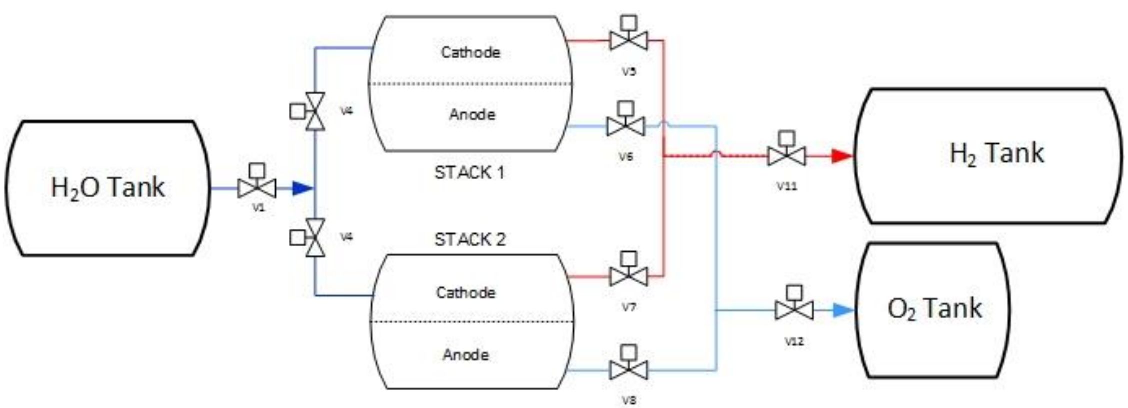
LAF (Liquid Anode Feed)



LAF represents the most studied/developed technology in EU, with best efficiency and lowest overall mass,

but with much higher complexity due to the need of feed pump, circulation pumps and phase separators

CVF/SWF Cathode Vapour Feed



CVF is relatively undeveloped in EU but presents a much simpler overall system; nonetheless this comes with a severe efficiency reduction and much higher overall stack mass, besides being far from proven for higher pressures and longer lifetimes

Eletrolyser studies and trade-off (3/4)

Conclusions:

Cell: LAF concept	
current density (A/cm²)	0.5 -2 A
active area (cm²)	23.4
current (A)	11.75- 46.8
voltage (V)	1.65 – 1.7
temperature (K)	353
pressure (bar)	100
mass (kg)	0.045
Stack	
Cells per stack	10
Stack voltage (V)	16.5 – 19.5 V
Stack input power (W)	796
Stack heating power (W)	100
Mass (kg)	1.1
Total	
Total Number of stacks	2
Total input power (W)	1595
Total heat dissipation (W)	199
Mass (kg)	2.2
Propellant generation rate (kg/s)	8.301E-5

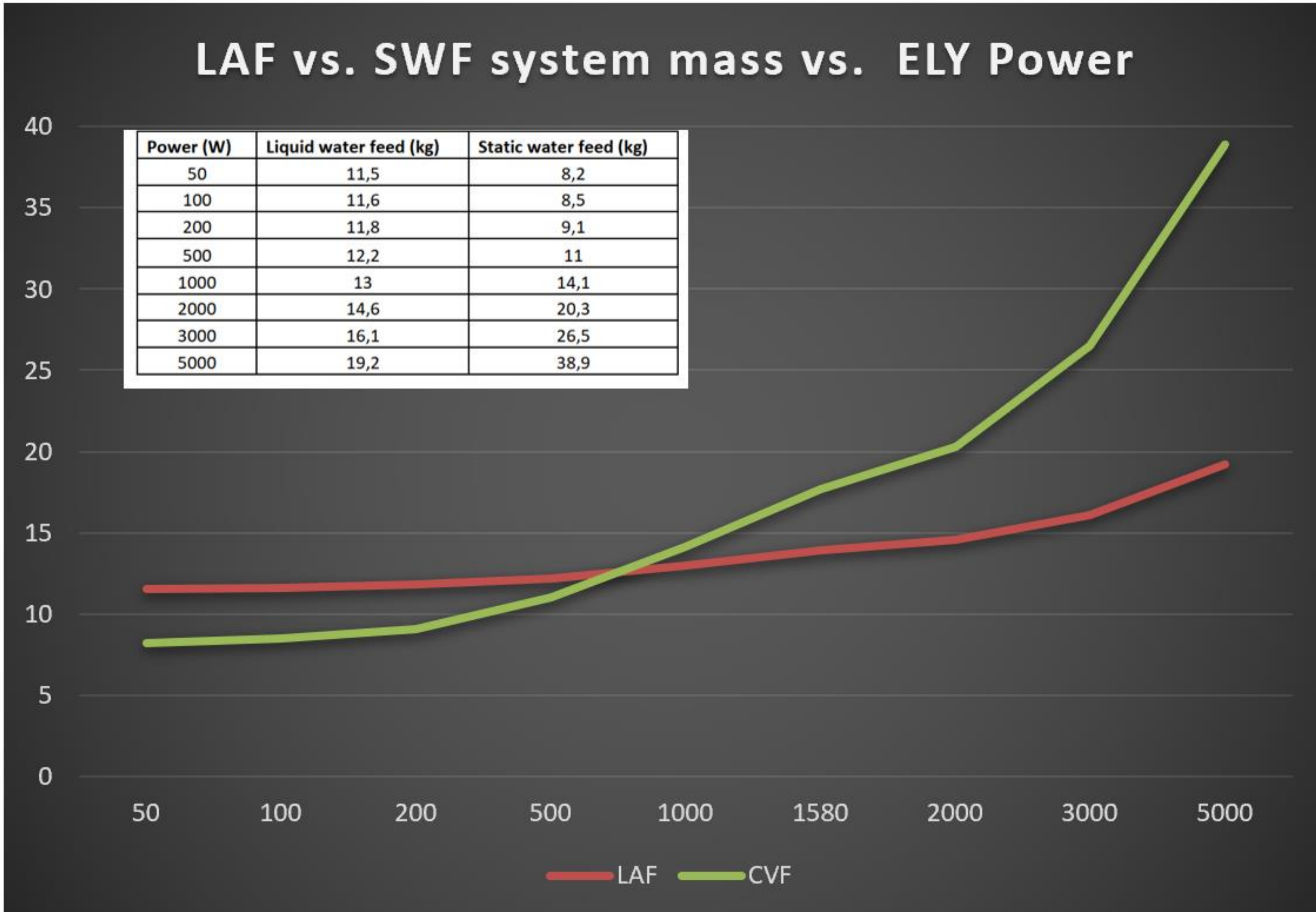
LAF (Liquid Anode Feed)

Cell: CVF/SWF concept	
current density (A/cm²)	0.125 – 0.5 A
active area (cm²)	23.4
current (A)	2.9 -11.7
voltage (V)	1.65 – 1.7
temperature (K)	353
pressure (bar)	100
mass (kg)	0.09 *
Stack	
Cells per stack	40
Stack voltage (V)	66 - 68V
Stack input power (W)	796
Mass (kg)	4.7
Total	
Total Number of stacks	2
Total input power (W)	1595
Total heat dissipation (W)	199
Mass (kg)	9.4
Propellant generation rate (kg/s)	8.301E-5

CVF/SWF Cathode Vapour Feed

Eletrolyser studies and trade-off (4/4)

Conclusions:



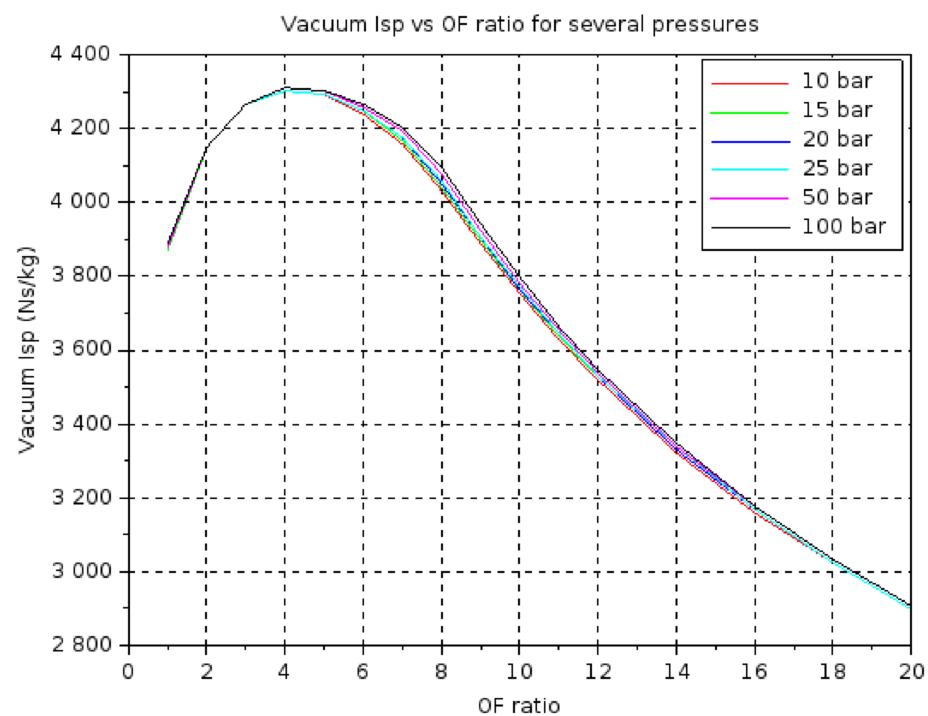
Power (W)	Liquid water feed (kg)	Static water feed (kg)
50	11,5	8,2
100	11,6	8,5
200	11,8	9,1
500	12,2	11
1000	13	14,1
2000	14,6	20,3
3000	16,1	26,5
5000	19,2	38,9

Thruster Performance (1/3)

To assess thruster performance, it is first relevant to determine propellant performance. The relevant parameters of thruster performance are Isp and combustion temperature. Calculations were made using CEA for a thruster with a nozzle expansion ratio of 100 and varying chamber pressures and OF ratios.

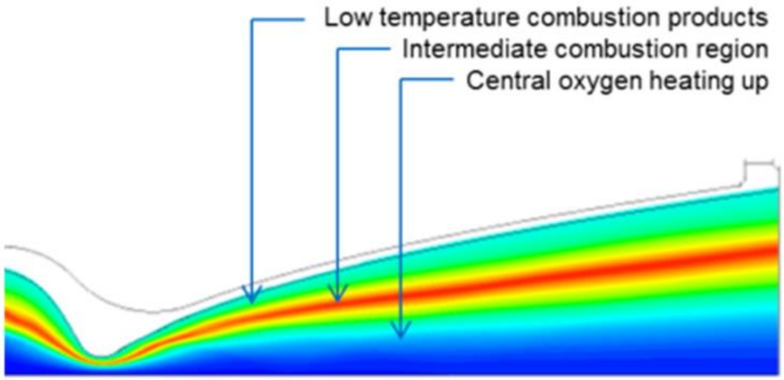
The calculations show that vacuum I_{sp} increases little with chamber pressure, especially near the peak. This peak region is offset from the stoichiometric value (OF=7.94) due to the influence of unburned hydrogen in the rich mixture that due to a smaller average molar mass improves I_{sp}.

$$I_{spcomb}=\begin{cases} I_{spbp}\cdot\left(\dfrac{OF_{bp}+1}{OF_{st}+1}\right)+I_{spcg}\cdot\left(1-\left(\dfrac{OF_{bp}+1}{OF_{st}+1}\right)\right) & OF\leq OF_{st} \\ I_{spbp}\cdot\left(\dfrac{OF_{st}}{OF_{st}+1}+\dfrac{OF_{st}}{(OF_{st}+1)\cdot OF_{bp}}\right)+I_{spcg}\cdot\left(1\left(\dfrac{OF_{st}}{OF_{st}+1}+\dfrac{OF_{st}}{(OF_{st}+1)\cdot OF_{bp}}\right)\right) & OF\geq OF_{st} \end{cases}$$

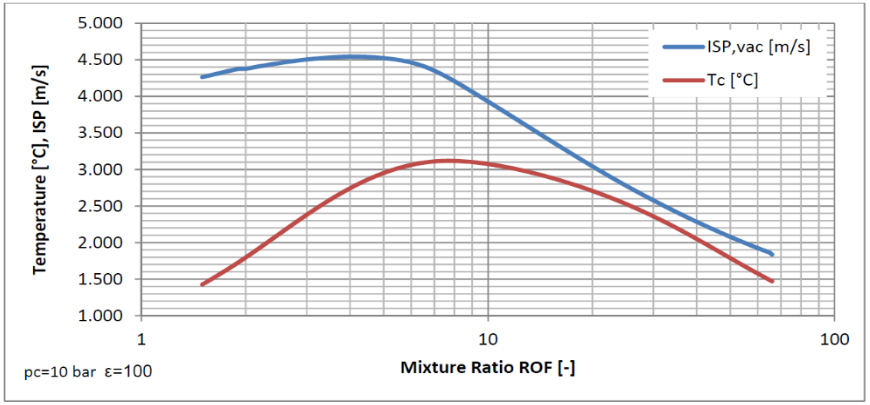


Thruster Performance (2/3)

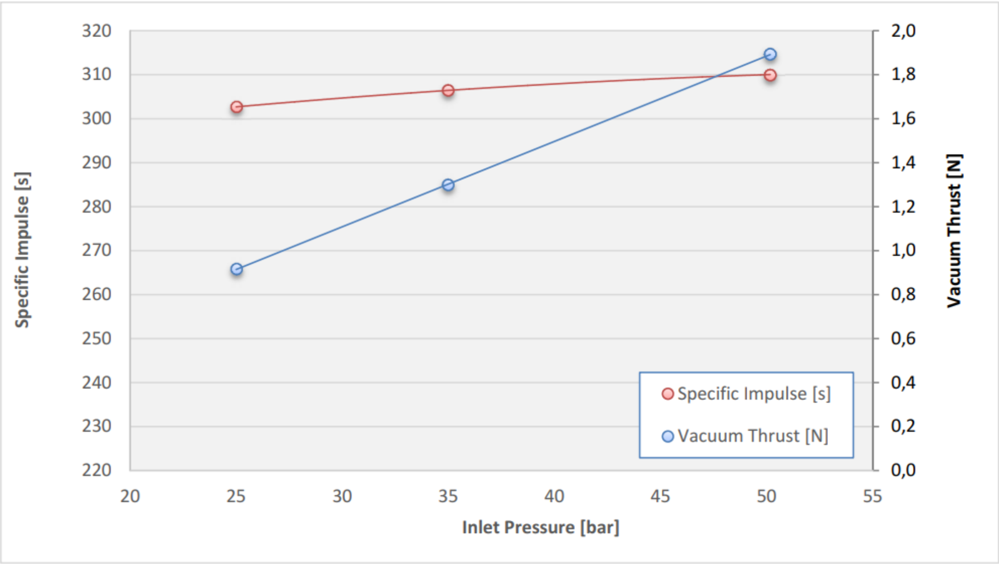
The key technology for the GO_2-GH_2 thruster is proprietary to ArianeGroup. Below the under development 2N thruster is presented, which was hot fire tested during this project. The stoichiometric operation is realized via a stratified combustion, where part of the hydrogen is fed and ignited via a catalytic bed, and the remaining oxygen is centrally injected.



This equates to “film cooling” as the gas part in direct contact with the nozzle walls is relatively cool, thus protecting the nozzle materials from melting since the adiabatic flame temperature is ≈ 3400K.

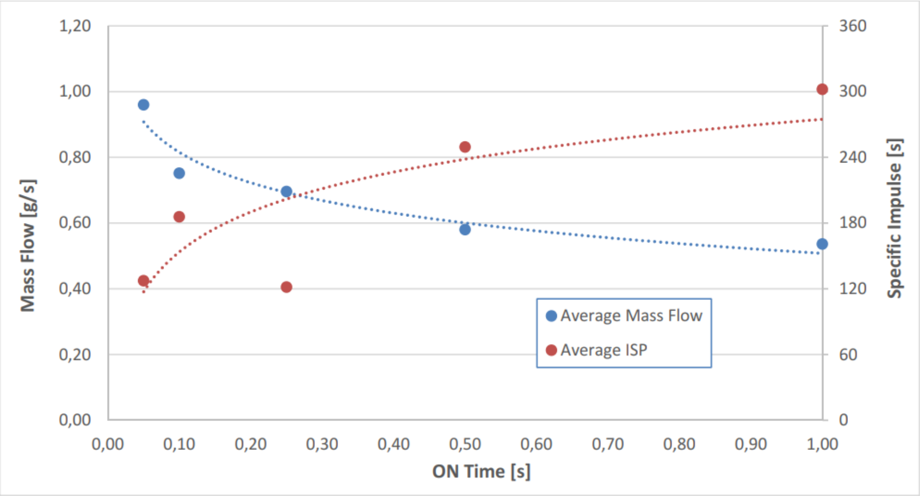


Thruster Performance (3/3)



The results for pulse mode firing (PMF) returned a lower Isp (close to 300s) on longer firing durations (i.e. close to 1s), while performance for very short firing, e.g. 50ms, can be significantly lower.

The thruster was tested in static firing (SSF) and pulsed firing modes, at different inlet pressures



Propulsion Architecture and Mission Performance Calculations (1/5)

During mission definition, the consortium took into consideration almost all attributes required by ESA's SoW; also the very low hydrogen storage density of hydrogen gas at room pressure makes it necessary that, for a 1-ton class satellite, any meaningful manoeuvre which is to be performed as an "Hohmann type manoeuvre" will greatly benefit from really high pressure storage of the gases under consideration, especially for Hydrogen;

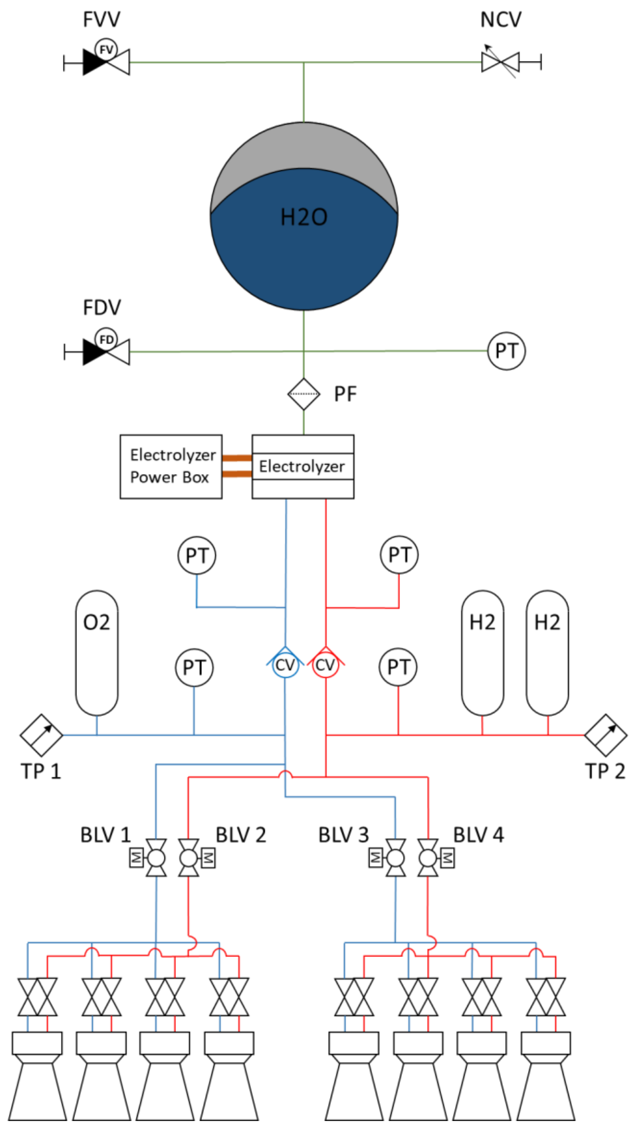
Since 100bar is currently seen as the maximum theoretically achievable for high pressure electrolysers and since the thrusters can fire at this pressure without need (or performance reduction) for a pressure regulator, the top level requirements were set off with 100bar as the target maximum pressure.

Parameter	Units	Value
I_{sp} (SSF)	[s]	310
Mass of propellant	[kg]	80.3
Mass of prop. system dry	[kg]	<47,5
Water Tank Sizing		
Initial pressure	Bar	20
Final Pressure	Bar	2
Water Tank Size	[dm3]	> 85
Gas Tank Sizing		
Gas Mass/manoeuvre	[kg]	3.7
Cycles		> 100
Initial Pressure	Bar	100
Final Pressure	Bar	10
Gas Tank Size O_2	[dm3]	25.6
Gas Tank Size H_2	[dm3]	51.2
Electrolyser Sizing		
Electrical Power	[W]	1600
Gas generation rate	[g/h]	270
Lifetime	years	10

Propulsion Architecture and Mission Performance Calculations (2/5)

- After selection of the LEO profile, reference mission was defined as per Sentinel-3 (≈1,1 ton, 10 years, SSO @ 814km)
- mission analysis gave a delta-V around of 215m/s;
- from known delta-V and satellite weight, total impulse was derived
- and with known Isp (≈310) so was propellant mass.
- The Isp achieved during test firing (310s) is a realistic goal; although blowdown will decrease average actual Isp, room for improvement exists.

Manoeuvre	Total ΔV (m/s)	Total Impulse (Ns)	Propellant (kg)
Insertion & orbit acq.	≈25	≈31.000	10.2
AOCS (10 yrs)	≈90	≈107.850	35.5
De-orbiting	≈90	≈105.000	34.6
Total	≈215	< 244.000	80.3



Propulsion Architecture and Mission Performance Calculations (3/5)

Overall mass budget and components TRL

Qty	Name	Phase	Status (TRL)[5]	Est. Mass of comp.(kg)	Tol. (kg)	Total mass (kg)	Comments /source
1	Liquid Tank[6] (135L available)	SRR	9	15,5	±0,2	16	MT-A PTD 177
1	HP Electrolyser Power Box	SRR	3	0,500	10%	0,55	Prototech
1	HP Electrolyser	SRR	3	5,3	10%	5,8	Prototech
1	GO2 tank	SRR	6	5,5 11,4	±0,25	5,7 11,65	Ardé 4699 Luxfer T200A
2	GH2 tank	SRR	6	5,5 11,4	±0,25	11,4 23,3	Ardé 4699 Luxfer T200A
8	Thrusters	SRR	5	0,3	<0,05	2,55	Actual AG 2N flight design
1	Normally Closed Valve	SRR	9	0,18	<0,05	0,2	AG SMA valve
5	Pressure Transducers	SRR	9	0,26	<0,05	1,3	Bradford Eng. (inc. cable)
4	Branch Valves/isolation valves	SRR	9	0,52	<0,05	2,2	O-RTG BLV mkII
2	Fill and Drain / Vent Valve	SRR	9	210	<0,05	0,44	Moog UK valve
2	Test Ports/Fill and Vent	SRR	9	0,09	<0,05	0,2	O-RTG FVV
2	Check Valves	SRR	7	0,3	<0,05	0,7	Estimated from Cobham design
1	Propellant Filter	SRR	9	0,050	<0,05	0,06	O-RTG prop. filter or Sofrance
8-12m	Propellant piping	SRR	NA	0,04	<0,05	< 0,3	
			Total			**47,5**	**Assuming Ardé COPV's**
			Total			**65,4**	**Assuming Luxfer COPV's**

Power budget in Propellant production mode

Qty	Name	Phase	Status (TRL)	Est. power of comp.(W)	Tol. (W)	Total power (W)	Comments /source
1	Electrolyser Power Box	SRR	3	50	10%	55	Prototech
1	HP Electrolyser	SRR	3	1500	TBC[8]	1600	Prototech
			Total			**< 1700**	

Power budget in Firing mode

Qty	Name	Phase	Status (TRL)	Est. power of comp.(W)	Tol. (W)	Total power (W)	Comments /source
8	Thrusters	SRR	5	15	±0.5	60[9]	No preheating assumed
1	Normally Closed Valve	SRR	9	12	±0.5	12	AG SMA valve
4	Branch/isolation Valves	SRR	9	5	±0.5	11[10]	O-RTG's BLV
			Total			**< 90**	

Propulsion Architecture and Mission Performance Calculations (3/4)

Table 3-8: Orbit decrease calculations for 100 kg/m² ballistic coefficient

Case 1 (B=100 kg/m²)	Step 1	Step 2	Step 3	…	Step 11	Total
Initial orbit, perigee (km)	814	792	770		594	
Apogee (km)	814	814	814		814	
Final orbit, perigee (km)	792	770	748		575	80.8
time (h)	7.5	7.5	7.5		6.7	64.4
ΔV (m/s)	5.75	5.75	5.76		5.27	63.2
Used propellant (kg)	2	2	2	…	1.8	21.8

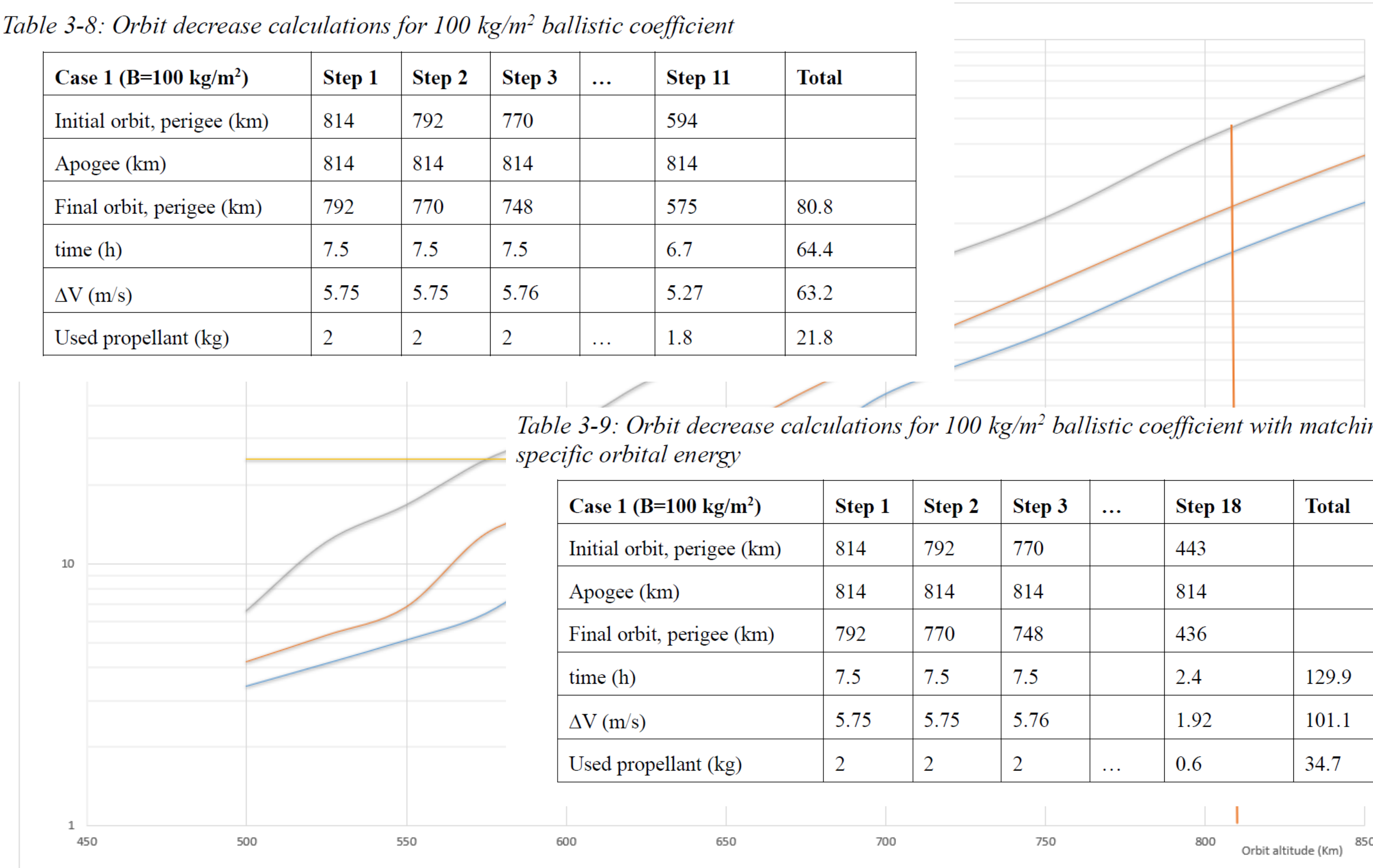


Table 3-9: Orbit decrease calculations for 100 kg/m² ballistic coefficient with matching specific orbital energy

Case 1 (B=100 kg/m²)	Step 1	Step 2	Step 3	…	Step 18	Total
Initial orbit, perigee (km)	814	792	770		443	
Apogee (km)	814	814	814		814	
Final orbit, perigee (km)	792	770	748		436	
time (h)	7.5	7.5	7.5		2.4	129.9
ΔV (m/s)	5.75	5.75	5.76		1.92	101.1
Used propellant (kg)	2	2	2	…	0.6	34.7

Propulsion Architecture and Mission Performance Calculations (4/4)

The increased Isp of the WPS allows for a total propellant reduction from 128kg to 80.3kg, with a reduction of ≈8% of total impulse, related only to the different mission lifetime (10 y vs. 12 on Sentinel 3).

	Dry mass	Propellant	Total
Hydrazine	23,5 kg	128[4] kg	**151,5 kg**
WPS	47,5 kg	84[5] kg	**131,5 kg**
difference	+24 kg	-44 kg	**-20 kg (13%)**

Lowering the minimum impulse per manoeuvre to between 300-500Ns was assessed; <u>such manoeuvres could be compatible with low pressure gas storage (@ between 5 to 20bar) at similar tank sizes</u>, while removing altogether the need for pressures above 30bars, making the following feasible:

- a conventional "hydrazine like" bladder tank could be used, without any disadvantage
- the ELY would then operate at pressures which, e.g. for SWF have been demonstrated, with significant less stresses on membrane
- and the gas storage tanks could employ conventional (i.e. no COPV) metallic construction

Such a system would be simpler and significantly closer to a TRL of 5 within 2-3 years making it entirely possible for a qualification well before the end of the decade. Nonetheless it could not perform active deorbit, even if it could still perform AOCS under most (if not all) restrictions imposed by ESA SoW.

Roadmap and recommendations (1/2)

During this study it became clear the electrolyser is the component with the lowest overall TRL, making it difficult to accurately have mass or volume budgets.

- At ELY level, consortium's suggestions are:
  - A more immediate development based on SWF, aiming for a demo mission towards 2026/27 with a possible commercial mission afterwards.
  - LAF development, aiming for a demo mission towards the 2030's, should be continued, more aimed at GEO, for which the LAF concept is best suited; this is because GEO intended spacecrafts are larger and can better take advantage of the superior LAF performance; on the other hand SWF is the best suited for LEO missions.
- Several components can and should be optimized, especially the water tank (which should be sized correctly for the WPS needs) and the gaseous storage tanks, where a decision between COTS or specifically developed components should be taken.
- Should LEO missions aim to fulfil the spacecraft controlled/active de-orbiting, or for potential GEO missions, 20N thrusters would provide extra flexibility.

Roadmap and recommendations (2/2)