# MULTIFUNCTIONAL UPPER STAGE EXPRESS PROPULSION SYSTEM CONCEPTS AND TECHNOLOGIES

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**Christian Hessel [(1)], Sylvain Goek [(1)], Armin Isselhorst [(1)], Jörg Klatte [(1)], Nora Volling [(1)], Bart van Zegbroeck [(1)]**

*[(1)] ArianeGroup GmbH, Bremen, Germany, Email: christian.hessel@ariane.group*

**KEYWORDS:** upper stage, propulsion system, evaporation cooler, deorbitation

**ABSTRACT:**

In general, this paper describes the performed investigations of the propulsion system concepts based on new upper stage configurations in the frame of the Multifunctional Upper Stage Express (MUSE) study. Due to the specific requirements of a cryogenic upper stage, the identified technologies primarily serve to reduce non-propulsive masses and helium consumption. In addition to the new composite based materials for propellant tanks and their insulation concepts, the adapted total propellant loading of the upper stages and an optimized operating point of the main engine, the required primary mission profile and also the disposal phase plays a crucial role in the propulsion system concepts and examined technologies.

## 1. INTRODUCTION

Hydrogen/Oxygen cryogenic propulsion has been mastered on upper stages since the very first launch of Ariane in 1979. The H10-III upper stage of Ariane 4 can still be considered today as benchmark in terms of structural index and performance. The storable upper stage of Ariane 5 EPS (Étage à Propergols Stockables) is an excellent example of an efficient versatile solution. The Ariane 5 ESC-A (Étage Supérieur Cryotechnique Type A) upper stage being essentially an adaptation of the Ariane 4 upper stage propulsion system to the constraints of Ariane 5. In this study, the reference to be taken as the receiver of future improvements is the Ariane 6 ULPM (Upper Liquid Propulsion Module).
By now, European launchers have a large and excellent background on upper stages, but must cope with the evolution of satellites and their increasing mission profiles. The Future Launchers Preparatory Programme (FLPP) of the ESA Space Transportation Directorate anticipates these trends, preparing the architectures of future launcher systems necessary for identified institutional missions and enabling technology maturation through integrated demonstrators. The upper stage is the key element of the launcher competitiveness, mainly regarding payload performance and the range of possible mission profiles. In this context, the initiated Multifunctional Upper Stage Express study is used as an incubator for Ariane 6 upper stage architecture concepts and required propulsion system technology identification.

## 2. OBJECTIVES and MAIN MISSIONS

- General Context of the MUSE Upper Stage Study

Multifunctional Upper Stage Express is a system study with the task to trade, optimize and consolidate credible architectures for the future evolutions of the Ariane A6 upper stage. The activity addresses the setting-up of the upper stage system requirements, the investigation of stage architectures, the identification of candidate upper stage concepts, methodologies and technologies, and the definition of maturation plans.

- Objectives of the MUSE Upper Stage

The overarching objective of the MUSE upper stage is to achieve significant payload performance gains with respect to the current A6 configuration (in the order of +2 tons for a GTO mission), while maintaining the priorities as follows:

- Priority 1: Payload performance increase
- Priority 2: Production cost reduction
- Priority 3: Mission versatility and extended lifetime

- Main Missions of the MUSE Upper Stage

Although GTO is the mission reference to evaluate the performance, MUSE shall be able to cover a large spectrum of missions (mono resp. multi-boost) in different launcher configurations (A64 resp. A62). Mono-boost means that a single boost is requested from the upper stage in order to place the payload into the targeted orbit (e.g. GTO, LTO). It should be noted that after payload injection an additional boost may be necessary if the upper stage must be deorbited.

On the contrary, in case of a multi-boost mission, the upper stage main engine has to re-ignite once or several times in order for the payload to reach the targeted orbit (e.g. LEO, MEO, GEO). Between two boosts, the upper stage has to manage a so-called ballistic phase whose duration depends on the mission.

## 3. Global Design Overview

### 3.1. Methodology and Upper Stage Design

During the concept phase, five reference upper stage configurations have been identified and subjected to trade-offs. The figure below shows the investigated architectures.

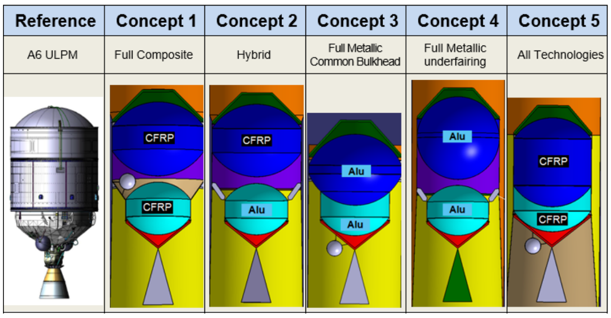


*Figure 2. MUSE Concept Configurations*

Based on the preselected criterias: Mission Performance, Industrial aspect & recurring costs, Risk & Opportunities and Versatility aspects the concept 1 “Full Composite” was preselected.

For determination of the best parameters for the propulsion system reference (e.g. thrust, mixture ratio and propellant loading), an overall optimization approach on stage level was performed. The objective of the optimization was to maximize the Net Present Value (NPV) resp. the average profit achieved per launch over the specified period of time. Based on the engine and functional propulsion characteristics, the theoretical optimum for thrust and propellant loading was calculated for Concept 1 "Full Composite" of the upper stage configuration.

Among several upper stage architectures and after evaluation and trade-off, one baseline configuration of the Concept 1 "Full Composite" was selected.

Main Design Features of the Upper Stage Study

- Engine Thrust Range: 115 - 180kN
- Engine Mixture Ratio Range: 5.5 - 6.0
- Engine Type: Expander Cycle
- Propellants: LOX / LH2
- Propellant Loading Range: 20 - 30 tons
- LH2 side:
  - Autogenous pressurization
  - Composite tank, Ø5.4 m
- LOX side:
  - Full Helium pressurization
  - Composite tank, Ø3.6 m
- Propellant Management options:
  - None
  - LH2 Boost Pump
  - LH2 Evaporation Cooler
- Deorbitation Kit

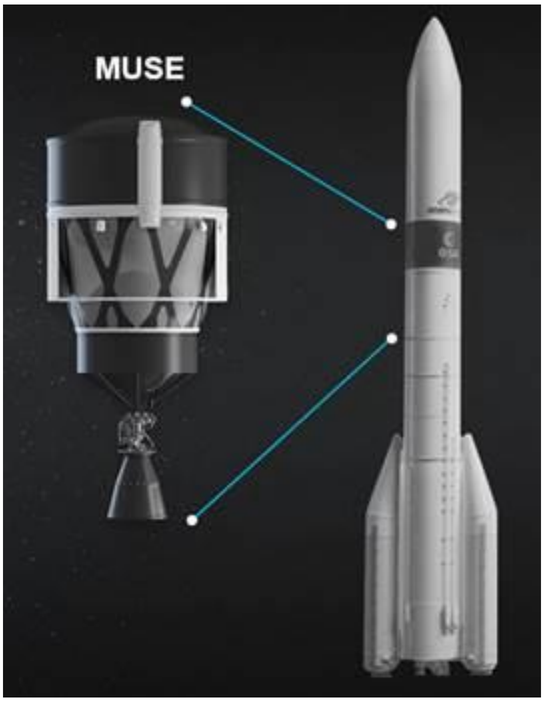


*Figure 3. MUSE Upper Stage Concept 1*

### 3.2. Flight Engine Image Main Design Features

In order to assess the impact on upper stage level using simplified models, an engine performance mapping is established as a function of key parameters (Thrust, ISP, Mass, Dimensions). This mapping takes into account the evaluation of the TCA and the Power Pack mass models elaborated during the concept phase and so-called Flight Engine Image Mapping. For the MUSE Upper Stage the Flight Engine Image from the FLPP ETID project was derived and extended with different thrust classes in the range of 115kN – 180kN.

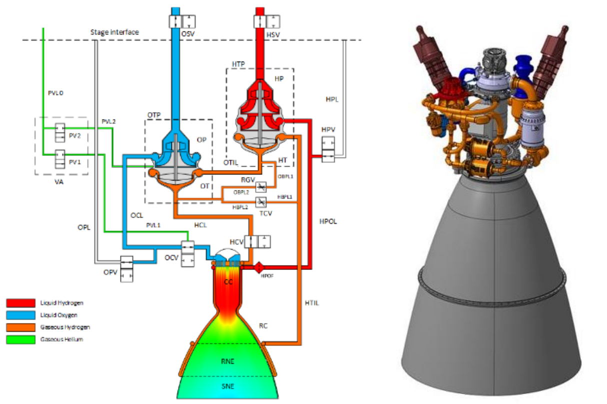

*Figure 4. Flight Engine Image FEI FLPP ETID Project [1]*

The stage system trade-off applied on a specific mono- and dual-boost mission portfolio identified a preliminary optimized thrust level of the upper stage at 150kN with mixture ratio of 5.9 and ISP of 463s. The dry mass of the engine including optimized power pack and nozzle is 365kg with total engine length of 3.50m.

For the 150kN class engine (upper thrust level of FLPP ETID FEI [1]) the following characteristic of the flight engine image is shown.

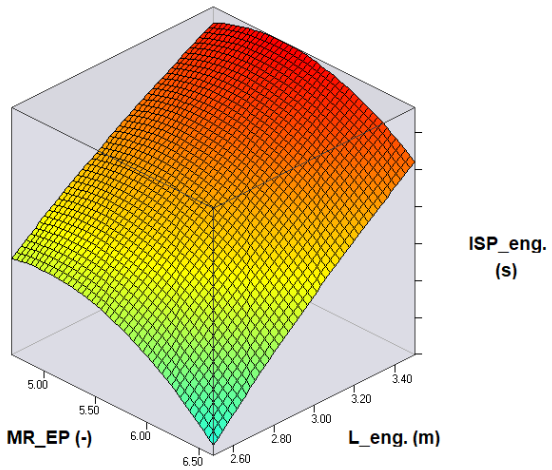


*Figure 5. Specific Impulse (vac.) Mapping*

The engine mass characteristic can be described with a function of thrust level (F) and engine length (L_eng. from Cardan to Nozzle Exit Diameter) according to the following equation

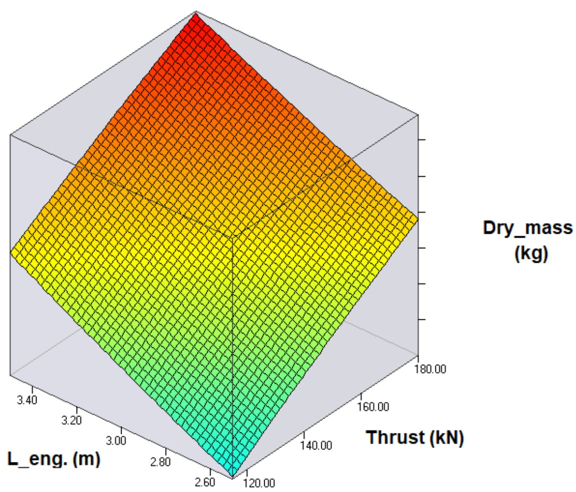


*Figure 6. Engine Dry Mass Mapping*

## 4. Propulsion System Architecture Overview and Trade-offs

The imposed technological choices, from upper stage high-level requirements, that are used as input for the concept sizing of the propulsion system are:

- ✓ The use of MUSE engine based on ETID technologies as Flight Engine Image
- ✓ The use of a Propulsion Kit on LH2 side as baseline, either Evaporation cooler or Boost Pump (see technology chapter)
- ✓ Helium Pressurization System for LOX side
- ✓ Electric command system for valve actuation (motor valves)
- ✓ Cold Gas Reaction System for Ballistic phase and Roll Control

Furthermore, specific trade-offs have been done in order to identify the most promising functional propulsion architecture in combination with the need for deorbitation as well as the possibility of a second boost scenario with a short ballistic phase between the two boosts:

- ✓ Deorbitation System
- ✓ 2nd boost Capabilities (Flushing of Lines)

### 4.1. General Overview

The functional propulsion system architecture trade-off was divided in two phases. First the optimization for mono-boost capability and second the re-boost capability. On the one hand the dry mass savings with carbon structures was the major objective for the upper stage and on the other hand the reduction of non-propulsive masses or the maximization of propulsive mass. Two major pressurization architecture options for the trade-off are identified:

- Full Helium pressurization architecture
- Full autogenous pressurization architecture

### 4.2. Functional Propulsion System Performance Evaluation

Evaluation of the different pressurization strategies is performed by comparing the different pressurant and technology options with focus on the associated impact on stage level. The evaluation takes into account the following mass contributors:

$$m_{FPS} = m_{TRM} + m_{Tank} + m_{Press} + m_{Storage} + m_{Equipment} + m_{Additional} \quad \text{Eq.1}$$

with,

- $m_{FPS}$, Functional propulsion system mass,
- $m_{TRM}$, Thermal residual mass,
- $m_{Tank}$, Tank dry mass,
- $m_{Press}$, Pressurant mass,
- $m_{Storage}$, Storage of pressurant
- $m_{Equipment}$, Propulsion equipment for main function pressurization and
- $m_{Additional}$, Additional equipment to fulfill a 2nd function if it is not compatible with or taken into account by the main function of the device.

For the pressurization system trade-off, the limit cavitation temperature was considered in order to determine the associated tank pressures and their impact on the tank dry mass and related equipment (vessels, heaters, lines, and valves), as well as the non-propulsive masses such as thermal residuals and pressurant mass.

Figure 7 and Figure 8 show the applied characteristic of tank pressure vs. pump inlet temperature for the trade-off of the propulsion system and the kit technology investigations.

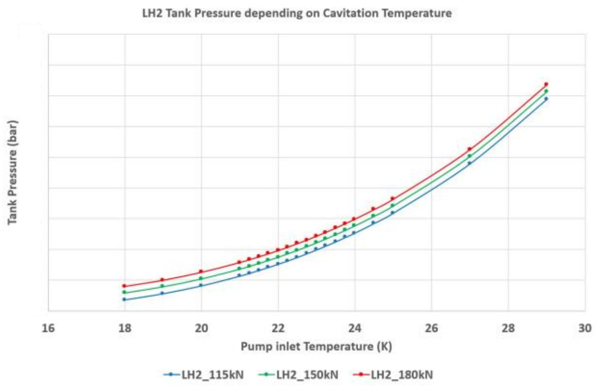


*Figure 7. LH2 tank pressure vs. cavitation limit temperature at pump inlet*

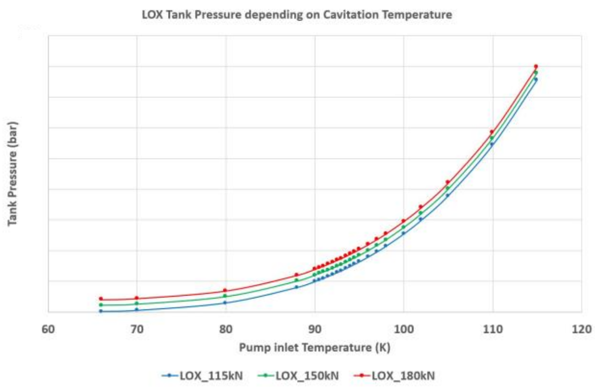


*Figure 8. LOX tank pressure vs. cavitation limit temperature at pump inlet*

### 4.3. Propellant Tank Stratification

One of the major contributors of the performance of the functional propulsion system is the tank propellant stratification and the resulting thermal residuals for the main feed pump of the engine power pack. The characteristic tank pressure vs. pump inlet temperature shown in Figure 7 & Figure 8 are one of the inputs for LH2 and LOX side analysis regarding thermal residuals.

An in-house transient 1-D model with fine discretization of liquid, gas phase & wall has been used to evaluate the thermal tank stratification & tank outlet temperatures as input for the propulsion architecture [2] & [3].

The equations are defined for an axis-symmetrical tank geometry with cylindrical section and user-defined domes. The 1-D model conserves heat & mass for liquid, gas & wall. It is validated via heritage large scale cryogenic tests & launcher data (Ariane 5 ECA). Figure 9 exemplifies the temperature stratification evolution for LH2 tank liquid phase.

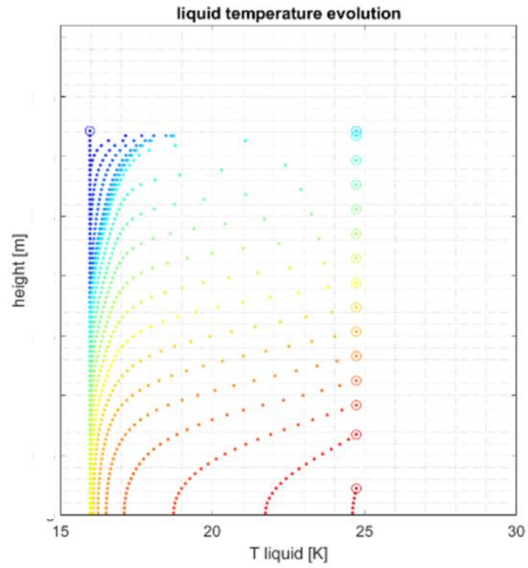


*Figure 9. LH2 Tank stratification evolution*

All parameters have been examined for constant tank pressure level.

- **LH2 thermal residuals**

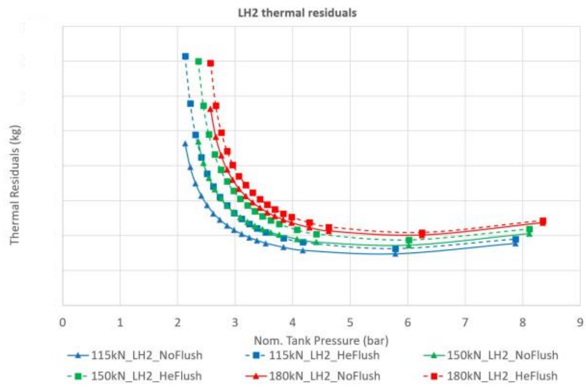


*Figure 10. LH2 tank thermal residuals*

- **LOX thermal residuals**

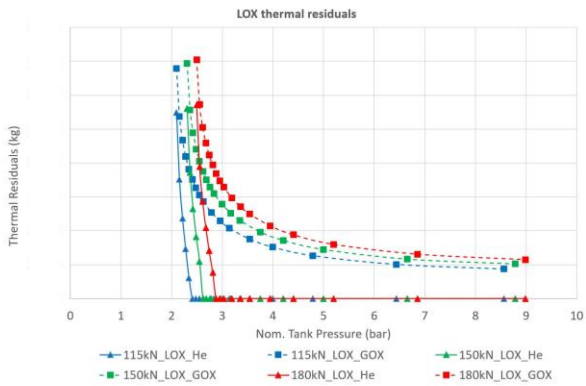


*Figure 11. LOX tank thermal residuals*

- **System analysis conclusion**

For autogenous pressurization with higher tank pressures the thermal residuals get lower, but on the other hand the higher tank pressure will increase the tank mass and the required pressurant mass. Full Helium pressurization at LOX tank could show no thermal residuals after certain pressure level – no improvement with higher pressure levels on thermal residual side. Lower pressure tank operations result in lighter tank mass and lighter pressurant mass but increase the thermal residuals and if the residuals are also drained further to the engine cavitation could occur.

In order to avoid an exceeding of cavitation limit at the main pump inlet, there are two possibilities: pressure increase or temperature reduction. Both can be achieved or partially achieved with kit technologies or core stage functionalities.

For the temperature reduction option, it is possible to prevent heating and remain sub-cooled for as long as possible. Thermal insulation is also limited to protect the tanks from external heat, and on the other hand, the pressurization system introduces warm autogenous or inert pressurized gas into the tank compartment. The pressurized gas will condensate or exchange with liquid / gas layer to heat the propellant. The local temperature reduction between tanks and motor unit could be achieved with a feedline heat exchanger, a so-called evaporative cooler, to cool the main flow to the main feed pump in the feedline by evaporating a small tap.

### 4.4. Technologies for elimination of thermal propellant residuals

A method to perform the propellant conditioning is a new challenge in the development of cryogenic upper stages with re-ignitable engines. The MUSE engine requires for its operation propellant under specific conditions w.r.t. pressure and temperature to avoid cavitation inside the propellant pumps.

- **Evaporation Cooler**

Propellant conditioning for main feed pump inlet via temperature reduction can be performed at tank side using global propellant conditioning based on de-pressurization of the tank or locally performed based on an evaporation cooler device. The strategy based on de-pressurization is only possible during ballistic phase and not during the boost phase due to bubble formation and stable operations of the engine. The strategy based on local cooling of propellant via evaporation cooler is possible during the boost phase.

a. <u>Working Principle</u>

The evaporation cooler is a device, mounted between the propellant tank outlet and the turbo-pump inlet in order to simultaneously cool down the main engine propellant flow.

The cooling is performed by propellant counter flow, tapped off from the main flow. The pressure of this counter flow is reduced through a junction plate, so that the liquid evaporates, resulting in a phase-change heat exchanger. In order to provide sufficient cooling areas, the counter flow is routed through cooling channels. The number and length of those cooling channels are defined by the required cooling performance.

The tap-off is a loss from the main flow, but only the propellant mass running through the engine needs to be cooled when the inlet temperature at pump inlet exceeds the operational limit of the turbo-pump cavitation (mono-boost at the end of the boost, multi-boost only for 2nd or 3rd boost with much less boost duration compare to 1st boost).

Since there is no tank de-pressurization for reconditioning, there is no need for tank re-pressurization with helium. In addition, the tank MEOP can be reduced (lighter tanks) because thermal residuals can be used with evaporation cooler operations.

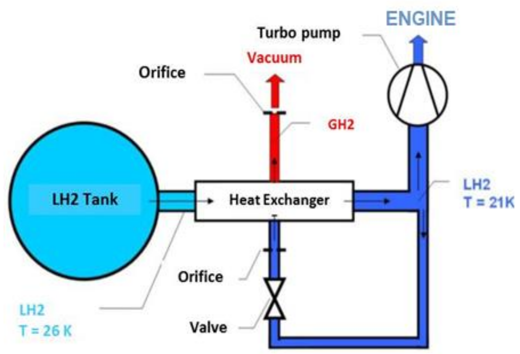


*Figure 12. Evaporation Cooler Schematic System View*

The LH2 Evaporation cooler consists of:
- Heat Exchanger (HX)
- Valve
- Tubing system
- Exhaust

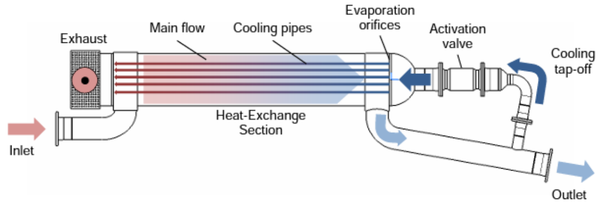


*Figure 13. Feedline Heat Exchanger - Evaporation Cooler*

b. <u>Performance Characteristic</u>

To assess the performance of nominal operational point for MUSE, a simulation model was used (Ref. EcosimPro v5.10.2).

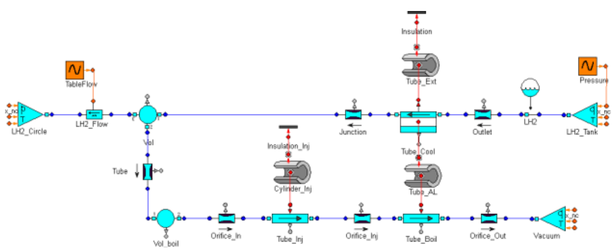

*Figure 14. EcosimPro ESPSS Evaporation cooler model*

c. <u>Maturation Activities</u>

- Functional Maturation (2016)

In past developments, the functional aspects of the evaporation cooler have been shown already at a ~1:180 downscale with a direct scaling to full scale via the number and length of tubes [4].

- Mechanical Maturation (2023)

The MUSE maturation Project FLASH has the objective to demonstrate the structural and manufacturing aspects of the evaporation cooler.
Two version of the demo versions was manufactured:

- GLUED Version
- BRAZED Version

The following verification tests have been performed at AGG facilities:

- Incoming Inspection
- Cleanliness check
- Leak test (initial)
- Cryogenic cycling test
- Leak test (final)

*Figure 15. Cryogenic cycling test with Sub-scale Heat Exchanger*

Result:

- **Brazed version** is selected for full scale model

### 4.5. Orbital Phase Performance Aspects

The orbital phase of an upper stage is particularly demanding especially if a second boost is to be undertaken with the main engine. Propellant management in the tank plays a major role here in order to maintain the correct temperature, pressure and properties (e.g. bubble-free) for the main engine power pack. In the ballistic phase, the manoeuvres and zero-g usually cause the propellant to leave its tank settling conditions after the first boost phase and disperse in the tank ullage and on the tank walls. The figure after shows the typical behaviour of propellant ullage mixing during payload release at distancing manoeuvre from the payload.

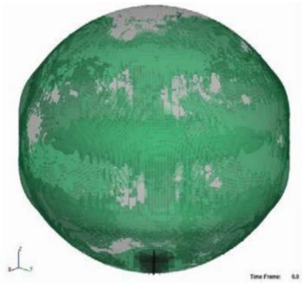

*Figure 16. Typical sloshing and mixing of liquid propellant and ullage during payload release phase*

Avoidance of wetting warm tank areas with cryogenic liquid at upper tank segments (cylinder, y-ring etc.), preventing of boil-off and ullage cooling down as well as bulk warming is essential in order to reduce non-propulsive masses.

Two Mission scenarios were investigated – Mono-Boost Mission CASE 1A and Dual-Boost Mission CASE 2B. The first mission that has been analysed, case-1a, is an A62 GTO mission with a singular, long boost phase followed by a deorbitation boost provided by a monopropellant deorbitation kit. The second mission that has been analysed, case-2b, is an A62 mission to MEO with 2 boost phases and no kick stage or deorbitation (only passivation).

The performance study of the orbital phase focuses in particular on the two-boost missions and examines the effects on the boil-off behavior and the pressurant mass consumption for the options with and without a propulsion kit (boost pump or evaporative cooler), as well as the effect of tank materials between CFRP tanks and metallic aluminium tanks on these effects.

The following sections show the pressurization mass and boil-off mass for the Mission 2b for LH2 tank compartment and LOX tank compartment.

- **LH2 tank compartment**

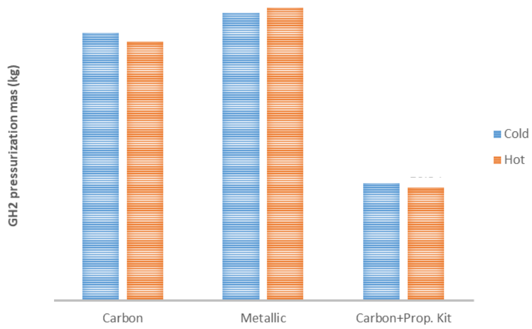


*Figure 17. Synthesis of GH2 pressurization mass for different cases, mission 2b*

The effect of GH2 pressurization mass consumption between carbon (CFRP) tank and metallic tank configuration is quasi identical. Figure 17 shows a large delta in GH2 pressurization mass requirement when using a propulsion kit. This is because without a kit, the propellant must be reconditioned prior to engine re-ignition to make sure that the engine turbine inlet temperature will be well below the cavitation temperature throughout the coming boost. This is done by venting the tank and forcing the propellant to boil-off to a lower saturation temperature. When the tank is then later re-pressurized, the propellant will be sub-cooled. When the stage is configured with a propulsion kit in the form of a heat exchanger or a boost pump, this procedure is no longer required as the avoidance of cavitation in the engine is assured by other means than propellant tank pressurization.

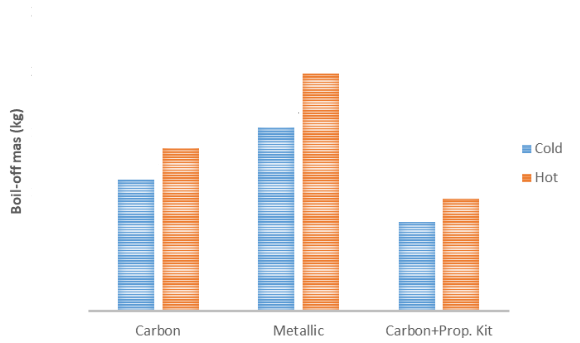


*Figure 18. Synthesis of LH2 boil-off mass for different cases, mission 2b*

Figure 18 illustrates the improved thermal characteristics of CFRP relative to their metallic counterparts when applied to the structural LH2 tank. The lack of reconditioning (i.e. forced boil-off) allows the configuration with propulsion kit to perform even better in terms of boil-off.

- **LOX tank compartment**

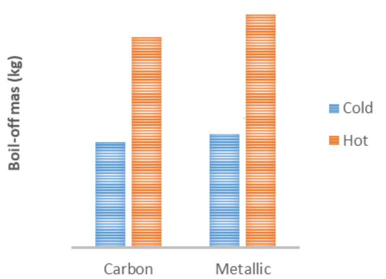


*Figure 20. Synthesis of LOX boil-off mass for different cases, mission 2b*

When in thermal equilibrium, the thermal radiative performance of the tank is considered to be equal to its metallic counterpart, as the same optical coating is applied. Because the LOX tank is not a structural tank, it does not have a large cylindrical element directly exposed to the external thermal environment. The small delta is mainly a result of the reduced thermal conductivity of CFRP at the structural interfaces, which are limited in size and covered in MLI.

## 5. Deorbitation Kit

In the frame of the MUSE Deorbitation and Demisability Study a detail pre-selection of the MUSE Upper Stage Deorbitation Kit or Deorbitation System was performed. In this trade-off several mission parameters and scenarios are taken into account:

- Propulsion Kit impact on Deorbitation System
- Performance Reserve impact on Deorbitation System for initial mass assumptions
- Mission scenario GTO or SSO with short ballistic phase deorbitation need of 70 – 110m/s

The deorbitation concepts depend on architecture capabilities on functional propulsion system and avionics sides as well as the mission scenarios. There are two different options as deorbitation solution:

- Use of internal upper stage propellant (liquid or gaseous) to provide thrust for the required deorbitation delta-v
- Use of external equipment or kits for the required deorbitation delta-v

An external equipment kit based on a storable liquid monopropellant system fulfils the required needs and was preselected and further investigated.

### a. Description

The deorbitation system is part of the Functional Propulsion System (FPS) which shall provide adequate propellant and/or pressurant to the consumed assemblies – tank, engine, RCS and deorbit kit. The integration of the deorbitation system into the MUSE Functional Propulsion System is depicted functionally below.

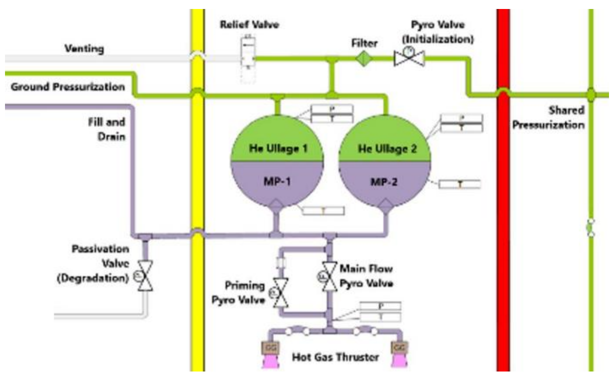


*Figure 21. Flow Schematic View of the Deorbitation Kit and the Upper Stage Interface*

### b. Performance Characteristics

For the deorbitation system with liquid storable monopropellant two types of storable propellants are investigated - monopropellant High Test Peroxide (HTP) and the monopropellant blended

hydrogen peroxide with ethanol and water (HTP-ETH-H2O).
Hydrogen peroxide is a proven propellant for monopropellants and has a high density and medium ISP. The substance can decompose, but due to its short-term use as a deorbiting kit, storage for several weeks or more is not intended and the upper stage use itself is completed in hours. An interesting trade-off option was the direct mixing of ethanol with a higher proportion of water and a lower proportion of pure H2O2 than the monopropellant HTP with a concentration of over 90%.
Ethanol does not visibly react with H2O2 when mixed at low temperatures and stored at or below room temperatures [8]. The detonability boundaries of H2O2/ethanol mixtures were already well known in 1943 and there is an increased risk of detonation if the mixture ratio is in a self-explosive detonation area. [9]. This blended monopropellant has already been studied and the latter research in the field showed a practicable application scenario [10], [11] and [12]. The storage duration, material selection of the tanks (AL5025 AM) and equipment as well as the storage conditions (pressure and temperature) also play a role here. As it only takes a few hours from loading to deorbiting, the fuel can optionally be stored under-cooled (super cooling) and the tanks are slightly insulated or are indirectly protected from heat input by the cryogenic tanks.
To compare both propellant options with respect to their performance, the ISP is determined depending on chamber pressure and expansion ratio.
For different concentrations of HTP ranging from 75% to 98%, different chamber pressures and expansion ratios, an ISP mapping is performed. Regarding the performance values and the availability of HTP, a concentration of 98% for the HTP is selected. This concentration is typical for HTP and can be easily sourced from suppliers. Again, the ISP is calculated as a function of chamber pressure and nozzle expansion ratio.

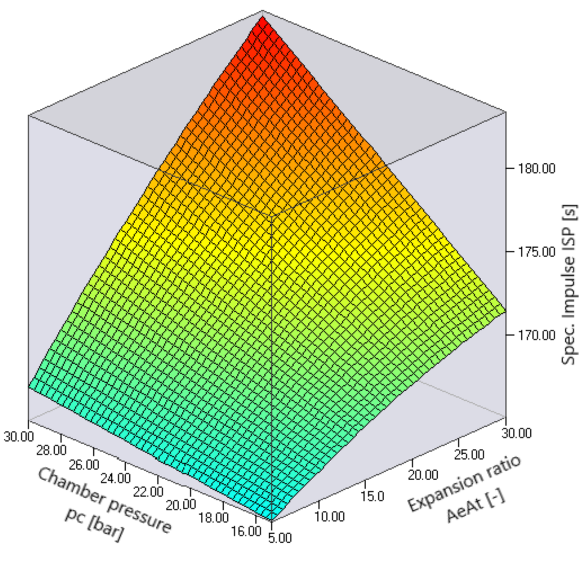


*Figure 22. Mapping of ISPvac. for 98% HTP based on CEA Calculation*

In order to compare the pure HTP propellant with the blended monopropellant, another mapping for the mixture with the highest ISP in relation to chamber pressure and expansion ratio is necessary. Before, the best HTP / ETH / H2O blend must be determined. Hence, the performance of different blends is calculated which are close to a detonable mixture. The blends used in the mapping are derived from the triangle plot of HTP, ETH and H2O, in which the detonable mixtures are marked. Along the red line, the mapping was conducted:

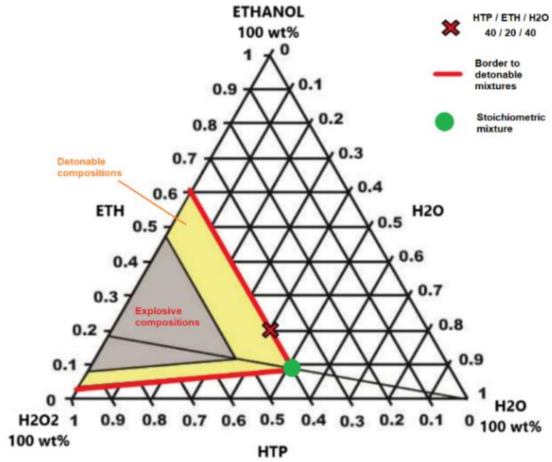


*Figure 23. Range of detonable compositions of HTP-ETH-H2O*

The calculation based on CEA cannot be carried out for all expansion ratios under the conditions of 10bar combustion chamber pressure and the variations of the fluid mixtures, as condensation effects occur in the nozzle, which are accompanied by a phase change (steam - liquid water), and the calculation with CEA cannot be continued. The table below shows different mixtures of HTP-ETH-H2O and their resulting specific impulse along the detonation line. Once for iso-expansion ratio representation at AEAT = 6 (for the stoichiometric mixture) and once at maximum expansion of the respective mixture before condensation occurs in the nozzle.

*Table 8. ISP mapping for blends along detonable border for different mass fraction*

| HTP (%) | ETH (%) | H2O (%) | $ISP_{vac}$ (@ $AEAT_{iso}$) (s) | $ISP_{vac}$ (@ $AEAT_{max}$) (s) |
|---|---|---|---|---|
| 40 | 60 | 0 | 196 | 229 |
| 40 | 50 | 10 | 193 | 265 |
| 40 | 40 | 20 | 192 | 249 |
| 40 | 30 | 30 | 196 | 246 |
| 40 | 20 | 40 | 212 | 266 |
| 40 | 10 | 50 | 156 | 162 |
| 40 | 9 | 51 | 147 | 147 |
| 50 | 8 | 42 | 154 | 159 |
| 60 | 7 | 33 | 161 | 171 |
| 70 | 6 | 24 | 168 | 181 |
| 80 | 5 | 15 | 175 | 192 |
| 90 | 4.5 | 5.5 | 189 | 226 |
| 96 | 4 | 0 | 205 | 244 |

It shows that the blend with the highest ISP values consist of 40% HTP, 20% ETH and 40% H2O. Therefore, this blend will be used in the following calculations shown in the figure below.

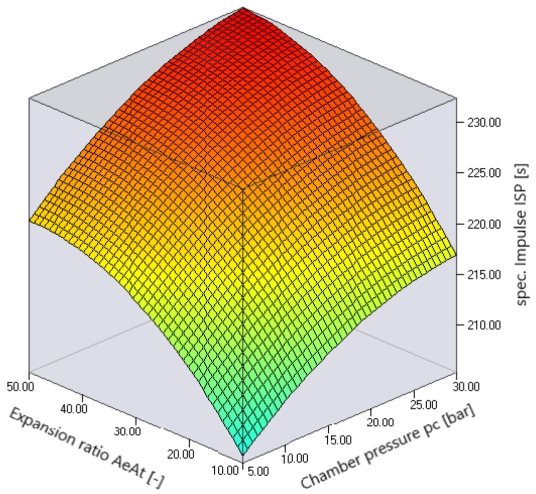


*Figure 24. Mapping of ISPvac. for HTP-ETH-H2O (40%/20%/40%) based on CEA Calculation*

The plot of the polynomial shows similar rising behaviour of the ISP with higher expansion ratios and higher chamber pressures.

c. <u>MUSE Stage Impact Assessment</u>

Using the two different propellant options for a MUSE stage deorbit kit results in the following savings with HTP-ETH-H2O propellant blends compared to 98% HTP for a range between 70 and 110 m/s deorbit performance:

- 70m/s case: 47kg
- 110m/s case: 75kg

## 6. CONCLUSION

The Multifunctional Upper Stage Express study is used as an incubator for Ariane 6 upper stage architecture concepts and required propulsion system technology identification. This will further foster the incremental development of Ariane 6 and beyond in terms of market share, performance increase and flexible mission scenarios of the launcher. The upper stage is the key element of the launcher competitiveness, mainly regarding payload performance and the range of possible mission profiles. The results of the upper stage study and the technologies identified was used to launch maturation projects and demonstrators to achieve TRL6 in an acceptable timeframe.

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