Family of Launchers Approach vs. "Big-Size-Fits-All"

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One option for future space transportation concepts could be a family of launchers supporting a wide range of payload performance. The idea bases upon using "building blocks" of common stages or main propulsion rocket engines and applying them in a modular way. A somehow contrarious option is a single TSTO launch vehicle serving all kinds of missions, even those which have payload mass requirements much below the design capacity.

The technical investigations described in this paper evaluate the two antipodal design approaches of either establishing a launcher family consisting of modular building blocks or choosing a full-size launcher which serves all missions with minimal adaptations of the upper- and kick-stage selection.

The paper summarizes major results of the preliminary technical design process. The overall shape and aerodynamic configuration, the propulsion and feed system, the architecture of the stages are described and different technical solutions are compared. Payload performance is optimized for the different concepts in the GTO-mission, manned flight to ISS and to SSO. The winged configurations' controllability in hypersonic reentry and subsequent subsonic flight is assessed. The study is completed by a relative comparison of to be expected RC/NRC of the different launcher concepts.

Keywords: RLV, LOX-LH2-propulsion, LOX-LCH4-propulsion, VTHL, VTVL, in-air-capturing

Nomenclature

D	Drag	N
I_{sp}	(mass) specific Impulse	s (N s / kg)
L	Lift	N
M	Mach-number	-
T	Thrust	N
W	Weight	N
g	gravity acceleration	m/s^2
m	mass	kg
q	dynamic pressure	Pa
v	velocity	m/s
α	angle of attack	-
γ	flight path angle	-

Subscripts, Abbreviations

3STO	Three-Stage-To-Orbit
AEDB	Aerodynamic Database
ALM	Additive Layer Manufacturing
AOA	Angle of Attack
BEO	Beyond Earth Orbit
CAD	Computer Aided Design
DOF	Degree of Freedom
DRL	Down-Range Landing site
ELV	Expendable Launch Vehicle
GLOW	Gross Lift-Off Mass
IAC	In-Air-Capturing
ISS	International Space Station
LAS	Launch Abort System
LCH4	Liquid Methane
LEO	Low Earth Orbit
LFBB	Liquid Fly-Back Booster
LH2	Liquid Hydrogen
LOX	Liquid Oxygen
MECO	Main Engine Cut Off
MR	Mixture Ratio
RCS	Reaction Control System
RLV	Reusable Launch Vehicle
RTLS	Return To Launch Site
TPS	Thermal Protection System
TRL	Technology Readiness Level
TSTO	Two-Stage-To-Orbit
TVC	Thrust Vector Control
VTHL	Vertical Take-off and Horizontal Landing
VTVL	Vertical Take-off and Vertical Landing
CoG	center of gravity
cop	center of pressure

1 INTRODUCTION

The rapid development in the field of international space transportation systems requires an updated and refined European strategy with a new generation of modern, high-performance launchers in the medium to longer-term. These should be capable of serving all major missions relevant to Europe - in a highly cost-efficient, sustainable, flexible and performance-oriented manner.

This ambitious goal is unlikely to be achieved by minor modifications to existing systems alone, but is requiring more wide-ranging steps. Partial reusability is probably the most promising approach for such a new launcher system that, nevertheless, could exploit synergy potential with Ariane, already containing important technological building blocks.

One of the launcher concept options could be a family of launchers supporting a wide range of payload performance as proposed in ESA's 2021 program *New European Space Transportation Solutions* (NESTS) [1] where "all studies state that future needs shall be answered by modular, re-usable, agile, flexible, robust and affordable solutions". The idea bases upon using "building blocks" of common stages or main propulsion rocket engines and applying them in a modular way with up to four different launcher classes from micro, through intermediate, heavy and "super heavy" types.

SpaceX, at the same time, is following a somehow contrarious approach. A single TSTO launch vehicle should serve all kinds of missions and should be economical even for those which have payload requirements much below the design capacity. This is already the case today with small satellite missions transported on the Falcon9 but will become in the future even more explicit with the Starship&SuperHeavy combination. This vehicle is intended to become soon an operational ultra-heavy launcher well exceeding the size of all rockets ever built to date (see independent technical analyses in reference [2]!) and it is unlikely, SpaceX would withdraw itself from its current key-market with comparatively small payloads.

The question coming up in this landscape can be formulated: What is the best approach for Europe and its space transportation needs? Building human settlements on Mars requiring a powerful deep space missions' capability is not in the top position of the agenda. However, huge and heavy single payloads are still required to be transported into orbit while constellation deployment, smaller Earth-observation satellites or even human space transportation [3] could be part of the portfolio mix.

2 COMMON MISSIONS AND ELEMENTS

2.1 Mission assumptions

All presented RLV-configurations in this paper are assuming similar key mission requirements:

- GTO: 250 km x 35786 km
- ISS crew, 200 km circ., 51.6°
- SSO: 500 km x 500 km, 97.4°
- Launch site: CSG, Kourou, French Guiana

The vehicles should be capable of performing secondary missions to LEO, MEO or BEO.

All upper stages are to be actively deorbited at the end of their mission into Earth orbits to reduce the buildup of additional space debris. A contingency of fuel mass is reserved for this final part of the mission.

2.2 Main propulsion systems

The different launcher systems studied make use of a portfolio of different liquid rocket engines, either in production, in development or in conceptual study level. All these engines make use of LOX as oxidizer and the fuel options hydrogen (LH2) and methane (LCH4).

2.2.1 Closed staged-combustion cycle engine

Staged combustion cycle rocket engines with a moderate 16 MPa chamber pressure are a good choice of an RLV propulsion system. A Full-Flow Staged Combustion Cycle with a fuel-rich preburner gas turbine driving the LH2-pump and an oxidizer-rich preburner gas turbine driving the LOX-pump has been defined by DLR under the name SpaceLiner Main Engine (SLME) [4]. It is interesting to note that the ambitious full-flow cycle is currently developed by SpaceX for its Starship&-SuperHeavy with the Raptor-engine [2]. The expansion ratios of the booster and passenger stage/ orbiter engines are adapted to their respective optimums; while the turbo-machinery, combustion chamber, piping, and controls are assumed to remain identical in the baseline configuration.

The SpaceLiner 7 has the requirement of vacuum thrust up to 2350 kN and sea-level thrust of 2100 kN for the booster engine at a mixture ratio of 6.5 with a nominal operational MR-range requirement from 6.5 to 5.5. Table 1 gives an overview about major SLME engine operation data as obtained by latest cycle analyses [4] for the MR-requirements of the semi-RLV-configurations studied here. The intended SLME architecture allows the booster engines after a certain number of flown missions to be expended on the ELV-core segment by exchanging the nozzle extension.

Table 1: SpaceLiner Main Engine (SLME) technical data [4] as used by reusable and expendable main stage

	RLV Booster	2nd ELV stage
Mixture ratio [-]	6.5	5.5
Chamber pressure [MPa]	16.95	15.1
Mass flow per engine [kg/s]	555	481
Expansion ratio [-]	33	59
Specific impulse in vacuum [s]	433.4	450.6
Specific impulse at sea level [s]	386.1	*352.6*
Thrust in vacuum [kN]	2359	2110.5
Thrust at sea level [kN]	2101.6	1651.6

The size of the SLME in the smaller booster type is a maximum diameter of 1800 mm and overall length of 2981 mm. The larger second stage SLME has a maximum diameter of 2370 mm and overall length of 3893 mm. Both engine variants are shown with their Integrated Power Head architecture of turbo-machinery and two preburners as simplified CAD-models in Figure 1.

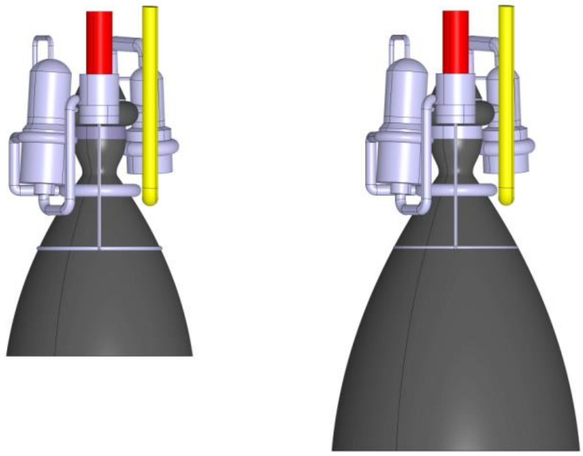


Figure 1: SLME simplified CAD geometry with nozzle expansion ratio 33 (left) and 59 (right) [4]

The engine masses are estimated at 3375 kg with the large nozzle for the upper stage and at 3096 kg for the booster stage. These values are equivalent to vacuum T/W at MR=6.0 of 68.5 and 72.6 [4].

2.2.2 Closed expander cycle engines

An advanced rocket engine already qualified today is the closed expander cycle Vinci which is to be used in the upper stage of Ariane 6 [5, 6]. Currently, Vinci is the most powerful engine of its type worldwide. The good performance data of this engine (Table 2) makes it attractive for powering smaller Building-Block stages (see section 3.2) or the upper or kick-stages of the 3STO- and Mini-TSTO-concepts described in sections 4.2 and 4.4.

The M10 *Mira* engine is a European methane rocket engine, conceived for use on upper stages of future Vega-E launchers. This type is a derivation of the Russian RD-0146 engine of

CADB as closed expander cycle for the LOX-LH2 propellant combination. [7, 8] The M10 engine has started recently its hot firing tests in Sardinia.

Table 2: Vinci technical data as used for expendable upper stage

Mixture ratio [-]	5.8
Chamber pressure [MPa]	6.1
Mass flow per engine [kg/s]	39
Expansion ratio [-]	175
Specific impulse in vacuum [s]	457
Thrust in vacuum [kN]	174.8

Table 3: Mira M10 technical data as used for expendable upper stage [7, 8]

Mixture ratio [-]	3.4
Chamber pressure [MPa]	
Mass flow per engine [kg/s]	28.16
Expansion ratio [-]	40
Specific impulse in vacuum [s]	362
Thrust in vacuum [kN]	100

2.2.3 Open cycle gas generator engines

PROMETHEUS is the precursor of a new European large-scale (100-tons class) liquid rocket engine designed for low-cost, flexibility and reusability [4] and the abbreviation stands for "***P**recursor **R**eusable **O**xygen **Meth**ane cost **E**ffective prop**ul**sion **S**ystem*". This engine is planned to be operated in open gas generator cycle. Baseline propellant combination of the PROMETHEUS-engine is LOX-LCH4.

Currently, the precursor of PROMETHEUS is under development. The calculated data in Table 4 have been generated by DLR to make realistic performance of a full-scale engine available for the launcher system design (compare also [4, 13] for similar gas-generator type but slightly different assumptions on nozzle expansion). The intention of this paper is *not* to provide an accurate prediction of the future PROMETHEUS for which technical characteristics are not yet all frozen. The purpose of the data in Table 4 is to offer performance estimates which are based on similar assumptions as the advanced hydrogen gas generator engine presented below and in Table 5. The data do not necessarily reflect the characteristics of the precursor or any future operational PROMETHEUS-M.

Different application might require selection of different engine characteristics. Therefore, slightly different engine assumptions have been selected for the VTHL and VTVL launcher concepts, both in the reusable first stage as well as the expendable upper stages. Note the different nozzle expansion ratios and chamber pressures for the 2nd stage application. Throttling could allow for a notable improvement in vacuum specific impulse, however, at the expense of reduced thrust levels. As will be described in the following sections 3.3 and 4.2 the relatively heavy weight of methane stages requires increased thrust which prohibits the use of the throttled PROMETHEUS-M operation in all investigated cases of TSTO and high-performance missions.

The engine masses are estimated at 1675 kg with the large nozzle for the upper stage and at 1382 kg for the booster stage. These values are equivalent to vacuum T/W of 85 and 97.5.

Table 4: Calculated technical data of Gas Generator Methane in different design and operation modes as used by reusable and expendable main stage

	RLV-C4-Booster	RLV Building Block	2nd ELV stage RLV-C4	2nd ELV stage Building Block	
Mixture ratio [-]	2.68	2.67	2.68	2.8	2.66
Chamber pressure [MPa]	12	11.772	12	4.866	9.75
Mass flow per engine [kg/s]	422.5	421	422.5	174.7	334.5
Expansion ratio [-]	20	16.4	59	100	100
Specific impulse in vacuum [s]	319	316	337	348	339.6
Specific impulse at sea level [s]	287	288	*251*	-	-
Thrust in vacuum [kN]	1322	1305	1397	596.4	1114
Thrust at sea level [kN]	1190	1200	1040	-	-

An interest has been proposed in using the advanced low-cost additive manufacturing processes to be implemented for PROMETHEUS but transferring them to an engine with the higher performing LOX-LH2 propellant combination. Such a hypothetical advanced Vulcain or PROMETHEUS "H" has also been calculated for this study and data are listed in Table 5. If the engine would be equipped with a larger nozzle of expansion ratio 100 and mixture ratio is shifted for throttled operations, Isp can be significantly raised to 442 s.

Many design similarities exist to the methane type PROMETHEUS with the main architecture change being two separate turbopumps for the LOX and LH2 sides. If such an engine would be developed its design could be based on the huge European heritage with Vulcain of which the latest and final variant 2.1 is already qualified for Ariane 6 [5]. Performance data of PROMETHEUS "H" would be very close to the existing Vulcain 2(.1) of which it is sharing many characteristics. Technical progress is expected mostly on the side of advanced, simplified manufacturing and digital controls.

Table 5: Calculated technical data of PROMETHEUS H(ydrogen) as used by reusable and expendable main stage

	RLV Booster	2nd ELV stage engine options	
Mixture ratio [-]	6.0	6.0	5.48
Chamber pressure [MPa]	12	12	10
Mass flow per engine [kg/s]	325	325	209.9
Expansion ratio [-]	20	59	100
Specific impulse in vacuum [s]	405	431	442
Specific impulse at sea level [s]	365	*317*	-
Thrust in vacuum [kN]	1292	1375	911
Thrust at sea level [kN]	1164	1011	-

The engine masses are estimated at 1750 kg with the nozzle of ε= 59 for the upper stage and at 1385 kg for the booster stage.

These values are equivalent to vacuum T/W of 80 and 95. The larger nozzle with ε= 100 is estimated at 2100 kg mass.

2.3 RLV recovery methods considered

The question of the best recovery method for an RLV-stage is subject of intensive debate and also to systematic investigations [19, 20]. Criteria for selection are performance and cost as well as technology availability which is linked to development risk. Two recovery and return strategies offer attractive conditions for high performance missions. Both are related to a Down-Range "Landing" (DRL) and these are baseline for the first stage RLV investigated in this paper. Current European TRLs are roughly the same for both methods.

2.3.1 VTHL using "in-air-capturing" (IAC)

The patented "In-air-capturing" intends the winged reusable stages to be caught in the air, and towed back to their launch site without any necessity of an own propulsion system [21, 22]. The idea has certain similarities with the vertical Down-Range Landing (DRL)-mode (section 2.3.2), however, initially not landing on ground but "landing" in the air. Thus, additional infrastructure is required, a capturing aircraft of adequate size for the to be towed RLV. Used, refurbished and modified airliners should be sufficient for the task.

A schematic of the reusable stage's full operational circle when implementing IAC is shown in Figure 2. At the launcher's lift-off the capturing aircraft is waiting at a downrange rendezvous area. After its MECO the reusable winged stage is separated from the rest of the launch vehicle and afterwards performs a ballistic trajectory, soon reaching denser atmospheric layers. At around 20 km altitude it decelerates to subsonic velocity and rapidly loses altitude in a gliding flight path. At this point a reusable returning stage usually has to initiate the final landing approach or has to ignite its secondary propulsion system.

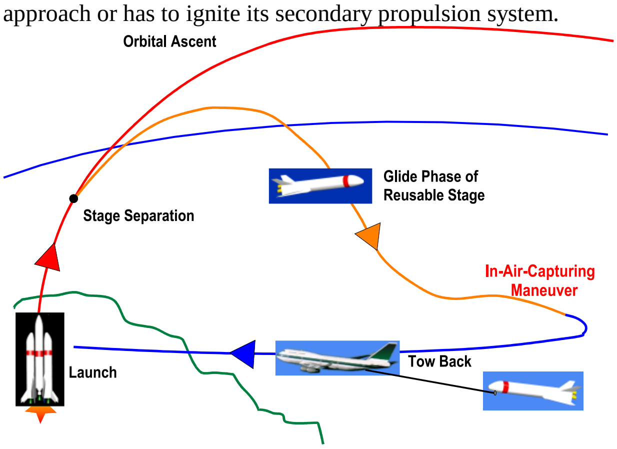


Figure 2: Schematic of the proposed in-air-capturing

Differently, within the in-air-capturing method, the reusable stage is awaited by an adequately equipped large capturing aircraft (most likely fully automatic and potentially unmanned), offering sufficient thrust capability to tow a winged launcher stage with restrained lift to drag ratio. Both vehicles have the same heading still on different flight levels. The reusable un-powered stage is approaching the airliner from above with a higher initial velocity and a steeper flight path, actively controlled by aerodynamic braking. The time window to successfully perform the capturing process is dependent on the performed flight strategy of both vehicles, but can be extended for up to more than one minute. The entire maneuver is fully subsonic (around 160 m/s) in an altitude range from around 8000 m to 4000 m [21]. In order to keep the two large vehicles always in a safe distance to each other, the actual contact and towing rope connection is established by a small agile vehicle [21, 22]. After successfully connecting both vehicles, the winged reusable stage is towed by the large carrier aircraft back to the launch site. Close to the airfield, the stage is released, and autonomously glides like a sailplane to Earth.

From a performance perspective, the IAC mode is highly attractive. In a systematic comparison of different RLV-stage return modes [19, 20, 21] with all launchers generically sized for the same GTO mission, the IAC-mode constantly shows a performance advantage compared to alternate modes. Costs for recovery of RLV-stages have been estimated and are found to be very similar for the IAC and DRL modes without any significant edge for one of them [21]. In combination with the performance advantage, the "in-air-capturing"-method based on current analyses seems to be an attractive technology for future RLV.

DLR together with European partners is currently preparing for flight testing the "in-air-capturing"-method on a laboratory scale by using two autonomous test vehicles. The EC-funded project FALCon should bring the TRL of the advanced IAC-recovery method beyond 4 in 2022. The project does not only address the lab-scale but also more sophisticated and refined simulations of a full-scale launcher-capturing. An existing airliner (study reference Airbus A340-600) with typical constraints is regarded for towing and the effects of its generated turbulent wake field are taken into account [21, 23].

2.3.2 VTVL with down-range sea-landing

Vertical Landing downrange is another viable option for future RLV proposals which has also been considered for the Ariane 6 Evolution option with liquid boosters described in [3]. Currently, SpaceX is using this method to land Falcon 9 and Falcon Heavy booster stages on the so-called autonomous spaceport droneships (ASDS) which are positioned downrange of the launch site either in the Atlantic Ocean (Cape Canaveral launches) or in the Pacific Ocean (Vandenberg launches).

VTVL require engine reignition capability to perform several maneuvers following MECO of the returning booster. First, the stage continues to travel on a ballistic trajectory up to its apogee, where it starts falling back to the earth's surface again. At a certain altitude, dependent on the mission profile and aerothermal loads experienced, one or more engines reignite to slow the stage down and thus limit re-entry loads (re-entry burn). After shutting down the engines and using the denser parts of the atmosphere to further slow-down the stage aerodynamically to subsonics, an engine is again reignited to gradually decrease the speed to a safe landing velocity coin-ciding with touchdown on the barge.

Compared to VTHL, vertical landing stages are not equipped with conventional wings or rudders and flaps. Instead, landing legs are required and some kind of aerodynamic controls, like grid fins for the Falcon 9, which usually are adding less dry mass as the VTHL recovery hardware. However, VTVL instead require a certain amount of propellant to be kept for the return maneuvers, thus adding to the inert mass of the launcher acceleration mission and hence reducing payload performance [13, 21].

3 OPTION 1: FAMILIES OF "BUILDING-BLOCK" LAUNCHER SYSTEMS

Europe's Ariane 6 developments in two different configu-rations, A62 with two solid strap-on boosters and A64 with four solid strap-on boosters, are ongoing [5]. Further performance enhancements by increasing the propellant loading of the solid boosters are in preparation for a Block 2 version. Nevertheless, activities on the next generation of completely new launch vehicle stages are constantly pushed forward [1 - 3].

Two of these potential future launcher options are described in this section 3 and the following section 4. On the first look they seem to follow an antipodal approach afterwards evaluated in the conclusion of section 6.

ArianeGroup's view of the future has been recently announced [10]: A small, partially reusable Maïa rocket starting its operations already in 2026, followed by reusable versions of the medium-lift Vega and heavy-lift Ariane 6 rockets. Common "Building-Blocks" of similar size stages for different size launchers should become a family spanning a significant payload range. Figure 3 shows how such configurations could look like if based on LOX-Methane propulsion and the PROMETHEUS liquid rocket engine [4].

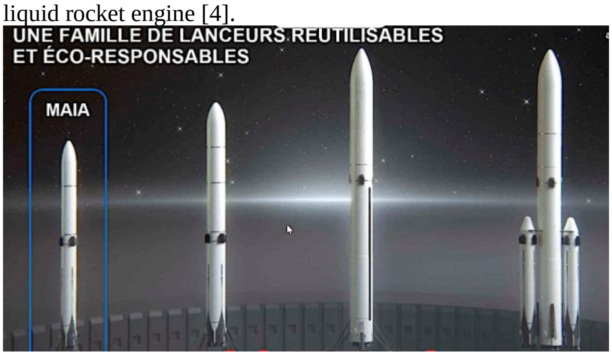


Figure 3: French industrial proposal of a future "Building-Block" launcher family [10]

Currently, no technical publication on the type of family as presented in Figure 3 is available. DLR initiated its own independent preliminary design study with the two propellant options LOX-LH2 and LOX-LCH4. High-level mission requirements might be somehow different to the launchers of Figure 3. This is not relevant as the comparison is made here between the "Building Block" Option 1 and the large size winged Option 2 approach.

3.1 Sizing approach

The optimum sizing of a "Building-Block" launcher family could be quite complicated. Minimization of life-cycle cost is a potential target, however, would require good knowledge of future launch scenarios. The latter is hard to reliably estimate, the more as projections are 30 to 50 years in the future.

Therefore, a pragmatic approach is followed which already delivers a suitable selection of building block stages which can be used as the baseline of any launcher family. These stages might not be exactly at a theoretical optimum but with the uncertainty in several assumptions this is not relevant for the launcher designs. A strong impact on the stage architecture choices comes from the definition of the rocket engines and their respective thrust levels.

In good accordance with the ongoing European discussions, only open gas-generator type engines have been selected for the "building block" families. Data for these engines all summarized in section 2.2.3 show that sea-level thrust (at take-off) is in the range 1150 kN to 1200 kN. As these engines' sizes are unlikely to be significantly changed in the future, the preliminary launcher sizing is adapting take-off thrust levels only in discrete steps by adding or removing a full engine. Thus, the iteration process is simplified because suitable initial acceleration is achieved only by a limited number of combinations.

Another baseline requirement is the preference of TSTO configurations over alternative architectures in order to reduce costs. The DLR-defined families do not strictly limit their configurations to TSTO but allow also 3STO or "common core booster" architectures for high-performance missions as derivatives. Nevertheless, also in the stage sizing of DLR the TSTO concepts play the dominant role in the pre-definition of the common stages.

Initial intention of the sizing procedure is finding a very broad range in payload performance of the complete launcher portfolio by combining only three baseline stages and the available engines described in sections 2.2.2 and 2.2.3. The powerful staged-combustion cycle SLME is not considered in the building blocks. It has been tried to find up to five different members of the launcher families, named S, M, L, XL, and XXL. The S, M and L are always TSTO while the highest performance XL and XXL are sized either as 3STO or as core TSTO with two common core boosters attached.

The search for suitable stage sizes has always been based on the M and L categories. The more extreme configurations to the left of the range (S) and to the right (XL and XXL) are derivatives and should not impact the stage sizing. For better convenience of the investigation process, the pre-definitions have been designed in the first step as ELV. The M-variant is to target the SSO mission and is to be based on a large first stage and a small expendable upper stage. The L-variant's design mission is GTO for single satellite deployment consisting of a large first stage with several liquid engines and a medium size second stage using the same engine in its vacuum variant.

Both M- and L-launcher variants are optimized for minimum GLOW individually. However, this optimization is under the constraints of the available engines and while the first stage includes several motors, the upper stage should have only a single engine with large nozzle expansion. Due to the latter's size, hardly a second engine could be accommodated.

After individual stage pre-sizing the data are compared and the size of the "Building Block" stage elements are frozen. It turned out both for hydrogen and methane BB-concepts that the choice is rather straight-forward and that the M- and L-variants remain close to their optimum size when adopted to the common stages. The three defined stages are transferred afterwards to the S-, the XL- and the XXL-variants.

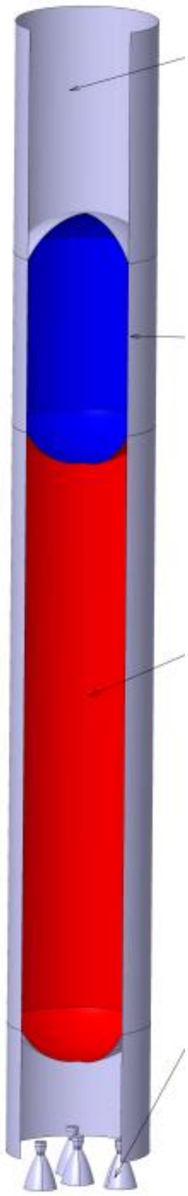
3.2 Family of hydrogen "Building Block" launchers

Following the described design approach and choosing the hydrogen engines Vinci (Table 2) and PROMETHEUS "H" in two versions for RLV-Booster and 2nd stage with expansion ratio 100 (Table 5) a total of five different launchers (Figure 7) based on three building block elements has been defined:

- S-Type: H61 + H15
- M-Type: H240 + H15
- L-Type: H240 + H61
- XL-Type: H240 + H61 + H15
- XXL-Type: 2 H240 + H240 + H61

for which the numbers represent total nominal propellant loading in tons.

Although the BB-launcher sizing is part of a Phase-0-study to show feasibility of the concepts, a tank geometry sizing and generation of simplified CAD-models is included and supports the overall size estimation and the approximation of launcher aerodynamics.



All large core stages (H240 and H61) have been sized for tank diameters of 5.4 m already used with Ariane 5 and Ariane 6. The small H15 is more mass efficient when designed with 3.6 m stage diameter.

The large first stage H240 (Figure 4) has an overall length of 37.8 m without 1/2-interstage and is using four PROMETHEUS "H" engines delivering more than 4.6 MN at lift-off. A common bulk-head is separating both propel-lants. In case of being used as VTVL-RLV, landing legs are installed to the aftskirt and movable fins at the forward interstage not shown in Figure 4.

The medium size H61 (see Figure 5 is defined in two variants: as first stage with a single PROMETHEUS "H" and small nozzle and as an upper stage with the same engine but large nozzle expan-sion of 100. Overall lengths are between 16.5 m and 23.3 m. Note the tank arrangement with separate bulkheads and the LOX-tank in the forward position is driven by the first stage application but could easily be adapted if beneficial for refined launcher lay-out.

The upper stage H15 (Figure 6) could be loaded with up to 15.7 tons of propellant and is to be accelerated by a single Vinci engine.

Figure 4: BB-element H240 (LOX-LH2)

3.2.1 Mission performances

The reference design missions are GTO for L-class and larger and SSO for S- and M-class as specified in section 2.1. All launcher types performances have been calculated in pure ELV- and partial RLV-operation. As partial reusability of future European space transportation is in the focus of this paper, only the payload capabilities of launchers under the assumption of returning first stages are provided

The L-launcher could deliver 2250 kg separated payload to GTO with the first stage performing a Down-Range Landing (DRL). The XL-configuration with two expendable upper stages more than triples this performance to around 7250 kg again using DRL. This capacity would already allow the trans-portation of super heavy satellites. The XL-variant shows im-pressive performance in reusable mode as GLOW is still below 400 Mg, however, at the expense of increased complexity of the three-stage launcher making use of all three BB-elements. The XXL-version with the three similar H240 stages has more than double GLOW compared to XL but in case of all its lower stages reused would exceed the XL-performance only by about 1 ton (+ 14%). In case the core stage becomes expendable, payload to GTO is strongly elevated and would allow to beat Ariane 5/6 in double launch assuming RTLS of the side boosters. If the boosters are performing a DRL-mode return, the payload in GTO could be up to 14800 kg.

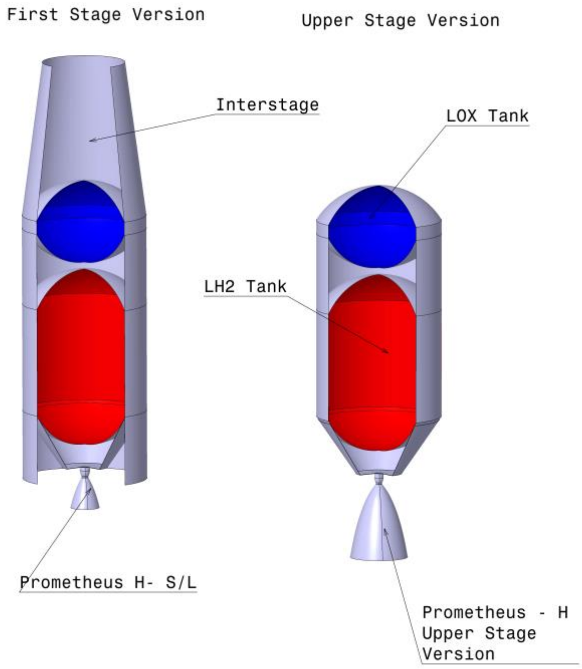


Figure 5: BB-element H61 as lower- (left) and upper-stage variant (LOX-LH2)

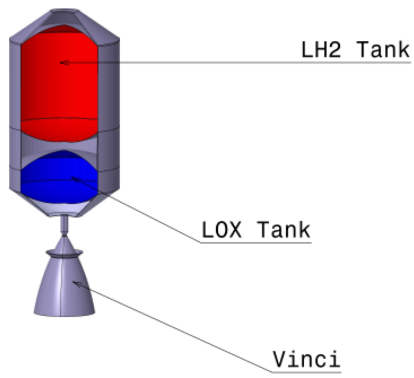


Figure 6: BB-element H15 as upper-stage (LOX-LH2)

It has been impossible to find an S-class BB-launcher with any meaningful payload assuming RLV first stages as VTVL. Further, T/W at the landing burn would exceed 3, even with the engine deeply throttled-down to 30% nominal thrust level. The safe vertical landing of the first stage by closed-loop control under these conditions is almost impossible. SSO-payload performance as an ELV is at 1.5 tons and comparable to Vega. With the S-Launcher based on the defined BB elements as RLV hardly feasible and at best with minimal performance, this concept is obviously the least attractive of the whole family.

Beyond the reference SSO- and GTO-mission also the capabilities of the XXL-configuration in an ISS-transfer orbit for crewed missions have been assessed. Mission constraints are similar to the previously analyzed DLR - ArianeGroup study EURASTROS (European Astronautical Space Transpor-tation) [3, 25]. Flight performance and safety quality are superior to those of Ariane 64, therefore, the most powerful

XXL-variant would be ready to support also independent crewed European space flight missions.

In Figure 7 an overview of the complete LOX-LH2 family is provided with its major dimensions and internal architectures.

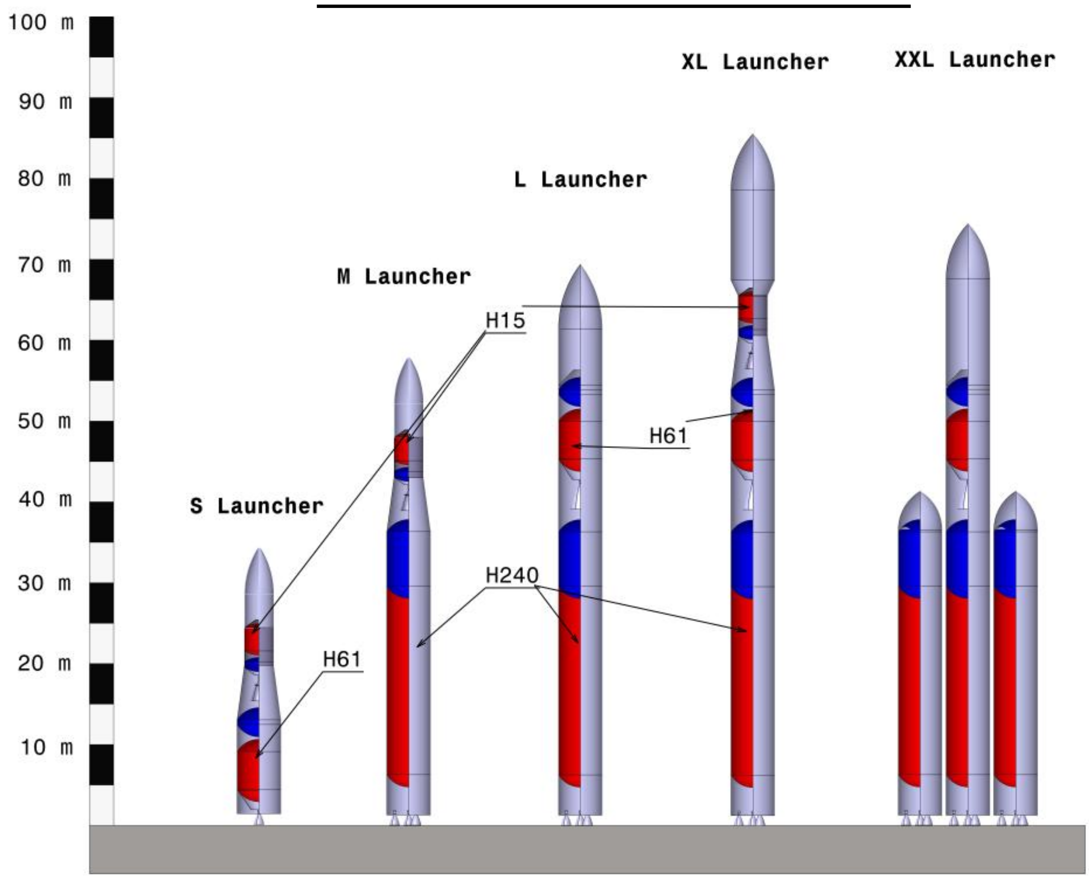


Figure 7: Building-Block launcher family LOX-LH2 combination

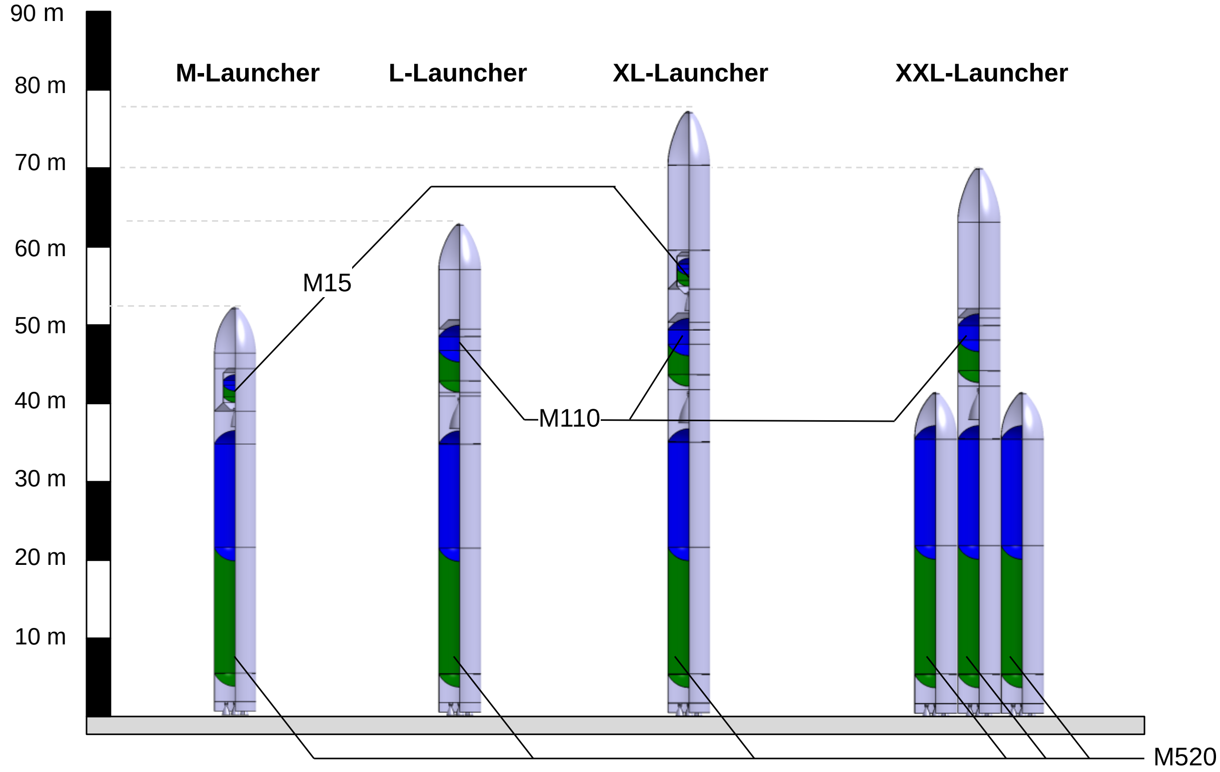


Figure 8: Building-Block launcher family LOX-LCH4 combination

3.3 Family of methane "Building Block" launchers

Following the same design approach as for LOX-LH2 and choosing the methane engines M10 (Table 3) and Methane Gas Generator in two versions for RLV-Booster and 2nd stage with expansion ratio 100 (Table 4) a total of four different launchers (Figure 8 shows an overview of the complete family with its major dimensions and internal architectures.) based on three building block elements has been defined:

- M-Type: M520 + M15
- L-Type: M520 + M110
- XL-Type: M520 + M110 + M15
- XXL-Type: 2 M520 + M520 + M110

for which the numbers represent again the total nominal propellant loading in tons.

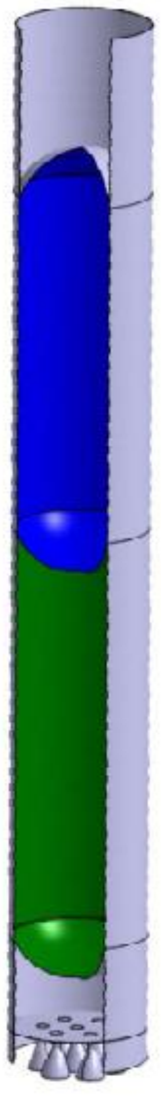


Figure 9: BB-element M520 (LOX-LCH4)

An S-Type was found unfeasible both in ELV and RLV-mode due to insufficient thrust capabilities of the M110 with single methane gas generator engine because T/W at lift-off is merely exceeding 1.0.

All large core stages (M520 and M110) have been sized for tank diameters of 5.4 m already used with Ariane 5 and Ariane 6. The very small M15 is designed with 3.1 m stage diameter. Different to the LOX-LH2-upper modules, all these LOX-LCH4-stages are defined with a common bulkhead between the propellants because temperature difference is small and thus complicated insulation can be avoided.

The large first stage M520 (Figure 9) has an overall length of more than 37 m without 1/2-interstage and needs seven Methane Gas Generator engines which fill already the complete base area. This geometrical constraint is also one justification for the reduced expansion ratio of the CH4-engine compared to LH2. In case of being used as VTVL-RLV, landing legs are installed to the aftskirt and movable fins at the forward interstage not shown in Figure 9.

The medium size M110 (Figure 10) is defined only as an upper stage with the Methane Gas Generator engine and large nozzle expansion of 100. Any first stage or booster application is found unfeasible.

The small upper stage M15 is limited in its performance as building block because maximum thrust of the Mira M10 engine is not exceeding 100 kN. Availability of a more powerful variant with 200 – 300 kN vacuum thrust would significantly improve the attractiveness of the M-size launcher in SSO-missions.

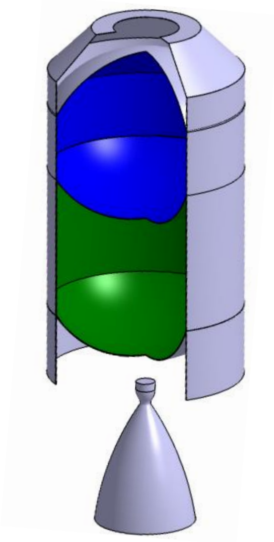


Figure 10: BB-element M110 as upper-stage (LOX-LCH4)

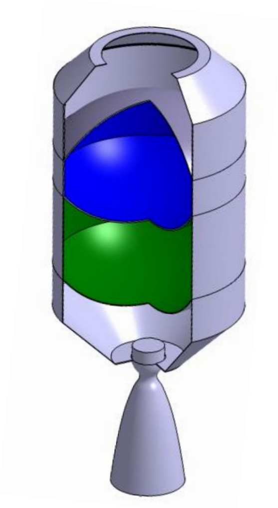


Figure 11: BB-element M15 as upper-stage (LOX-LCH4)

3.3.1 Mission performances

The reference design missions are GTO for L-class and larger and SSO for S- and M-class as specified in section 2.1. All launcher types' performances have been calculated in pure ELV- and in partial RLV-operation. As partial reusability of future European space transportation is in the focus of this paper, again only the payload capabilities of launchers under the assumption of returning first stages are provided

The L-launcher is deemed unattractive as RLV to GTO because hardly any separated payload mass can be delivered. The 3STO XL-configuration with two expendable upper stages achieves more than 6300 kg when using DRL. This capacity is below the LH2-XL-type and would not allow the transportation of all existing super heavy satellites. The XXL-version with the three similar H520 stages more than doubles GLOW to above 1800 Mg but with all its lower stages reused would not exceed the XL-performance. In case, both the core stage and the upper stage are expendable, payload to GTO is elevated and would reach more than 8300 kg in double launch assuming RTLS of the side boosters. If the boosters are performing a DRL-mode return, the payload in GTO could be up to 13900 kg.

Beyond the reference SSO- and GTO-mission also the capabilities of the XXL-configuration in an ISS-transfer orbit for crewed missions have been assessed. Mission constraints are similar to the previously analyzed DLR - ArianeGroup study EURASTROS (European Astronautical Space Transportation) [3, 25]. Flight performance and safety quality are again superior to those of Ariane 64, therefore, also the most powerful methane-variant would be ready to support independent crewed European space flight missions.

3.4 Critical points of "Building Block" launchers

Overall, the payload performance of the LOX-methane BB-launchers with reusable RLV-stages in VTVL-mode is significantly lower than that of similar LOX-LH2 variants. The roughly 90 s lower Isp of LOX-LCH4 compared to LOX-LH2 (compare Table 4 and Table 5) is to be compensated by a significantly higher propellant mass needed for reentry and landing maneuvers. Thus, less fuel is available for the ascent acceleration, reducing payload mass.

As minimum payload mass requirements are not defined for the reference missions and the focus of the investigation has been on the principal feasibility of a family of BB-launchers, LOX-LCH4-variants have not been iteratively sized for the same performance as their LOX-LH2 counterparts. It should be noted that only the strongest methane versions XL and XXL are capable of delivering any payload to GTO with the VTVL-RLV first stage although the L-size TSTO's GLOW is already approaching 750 Mg. A methane-based BB family with similar performance as a hydrogen-based BB family would need about twice the number of engines and roughly the same tank sizes despite its increased propellant densities.

In the past, separation velocities only up to 3.5 km/s (~Mach = 12) have been observed for VTVL first stages. Any separation conditions above this value require further analyses to determine the potential impacts on the system design. Such critically high speeds are relevant for the M- and the XXL-class of both investigated propellant combinations. Therefore, these classes might be unfeasible as VTVL or need to be modified resulting in reduced payload performance.

The preliminary sizing process of all building blocks assumes ambitious but still realistic stage masses. However, the BB with potential applications in several different launcher configurations and for a variety of missions would need to be designed for the most demanding structural load cases. In order to keep the family concept flexible for future evolution and potential growth and also considering the stiffness requirements for ascent control of all variants, the actual stage masses might significantly increase. Any definitive answer on the impact and potential restrictions require a considerably more detailed analysis followed by thorough evaluation of obtained results.

4 OPTION2: NEXT-GENRERATION BIG RLV-CONFIGURATIONS

A somehow different idea in defining the next generation of partially reusable heavy launchers has been under investigation in several system studies. Instead of creating a family with potentially different reusable first stages, the "Big-Size-Fits-All"- approach assumes one sufficiently large reusable stage as the baseline element to be combined with different types of expendable upper stages. The idea is also building on a limited number of elements like similar rocket engines (not much different to BB-families) but would instead allow the use of one single launch-pad for all intended missions.

The SpaxeX SuperHeavy&Starship [2] is an example of such a configuration which should serve multiple missions to different Earth orbits with significant range in payload mass. However, SpaceX is intending to be fully reusable with its two stages.

The system studies at DLR's space launcher system analysis department SART have investigated not only one preferred type but different return and recovery modes, as well as different propellant and engine cycle options [3, 12, 13, 14, 15, 16, 17]. Future European RLV configurations with reusability of 1st or booster stages with tandem arrangement of a large expendable upper stage have been preliminarily designed as TSTO for a GTO-reference mission, however, reaching significant size of up to 80 m length [12, 13].

Approaching or even exceeding the payload performance expected for Ariane 6 in GTO or Lunar exploration missions would require extremely tall launcher configurations in case of tandem-staged TSTO with reusable first stage. Therefore, for this class of RLV a parallel stage-arrangement is preferable: a winged stage is connected to an expendable upper segment with potentially various internal architectures. References 14 and 15 have demonstrated that a payload range between 12 to 15 tons GTO-class with multiple payload capability can be achieved by a 3-stage architecture while still remaining at relatively compact size. Less demanding missions to different LEO can be served as TSTO. Beyond the winged VTHL-concepts in focus of this section, similar VTVL options in architecture and size have been studied as a potential alternative [3] and might be reconsidered in future work.

4.1 Assumptions and Design Requirements

The launcher is to be designed for the most suitable combination of high commonality in major components and providing good mission flexibility. The expendable section could be single stage or two-stage, hence the launcher results in a 2-(TSTO) or 3-stage (3STO) to orbit configuration. The design payload target as 3STO is approximately 14000 kg to GTO reaching beyond the capability of A64 [5] and should provide multiple payload deployment capability. Using an adapted, reduced size upper segment, satellites have to be carried to different LEOs.

The TRL of all implemented technologies needs to reach 5 to 6 in 2030 for full-scale development-start enabling operational capability in approximately 2035. The design target for the RLV is 150 missions and between 5 to 10 missions for the engines. A "reusability kit" approach with every other mission flown as ELV is *not* intended for the reusable stages.

4.2 Preliminary architectures of RLVC4-types

The first stage of all investigated RLV-C4 is designed as winged RLV, however, in different sizes and lay-outs depending on the propulsion choice. The expendable stage or stages are attached in parallel configuration on top of the 1st stage. An important design constraint is the requirement of using similar engines in the reusable stage and the large expendable second stage, however, with adapted nozzles. This engine similarity allows for reduced development costs and might permit the reusable engine to be expended after certain number of missions on the RLV.

In case of 3STO systems the fairing covers all of the third stage and the payload and hence connects to the upper part of the interstage as visible in Figure 12 at left. The upper stage for high performance missions, mainly GTO-injection, is selected as H14 for all concepts. An external tank diameter of 5.4 m is no longer suitable for that loading if the stage's dry mass should be attractive. Vinci is the sole engine choice in the 3rd stage.

The 2nd expendable stage is defined as an H150 in case of hydrogen and becomes even more compact than the core stage of the classical Ariane 5G. Note the expendable stage arrangement with the H150 forward skirt or 2-3-interstage adjacent to the RLV intertank ring (Figure 12 at left and center).

The third launcher option investigated uses the same winged RLV first stage but a significantly smaller expendable upper stage to serve smaller payloads in low-energy missions. Figure 12 at right depicts a technical solution with the same attachment point on the RLV and the small expendable upper stage of the 3STO powered by Vinci-engine and significantly reduced size of the payload fairing.

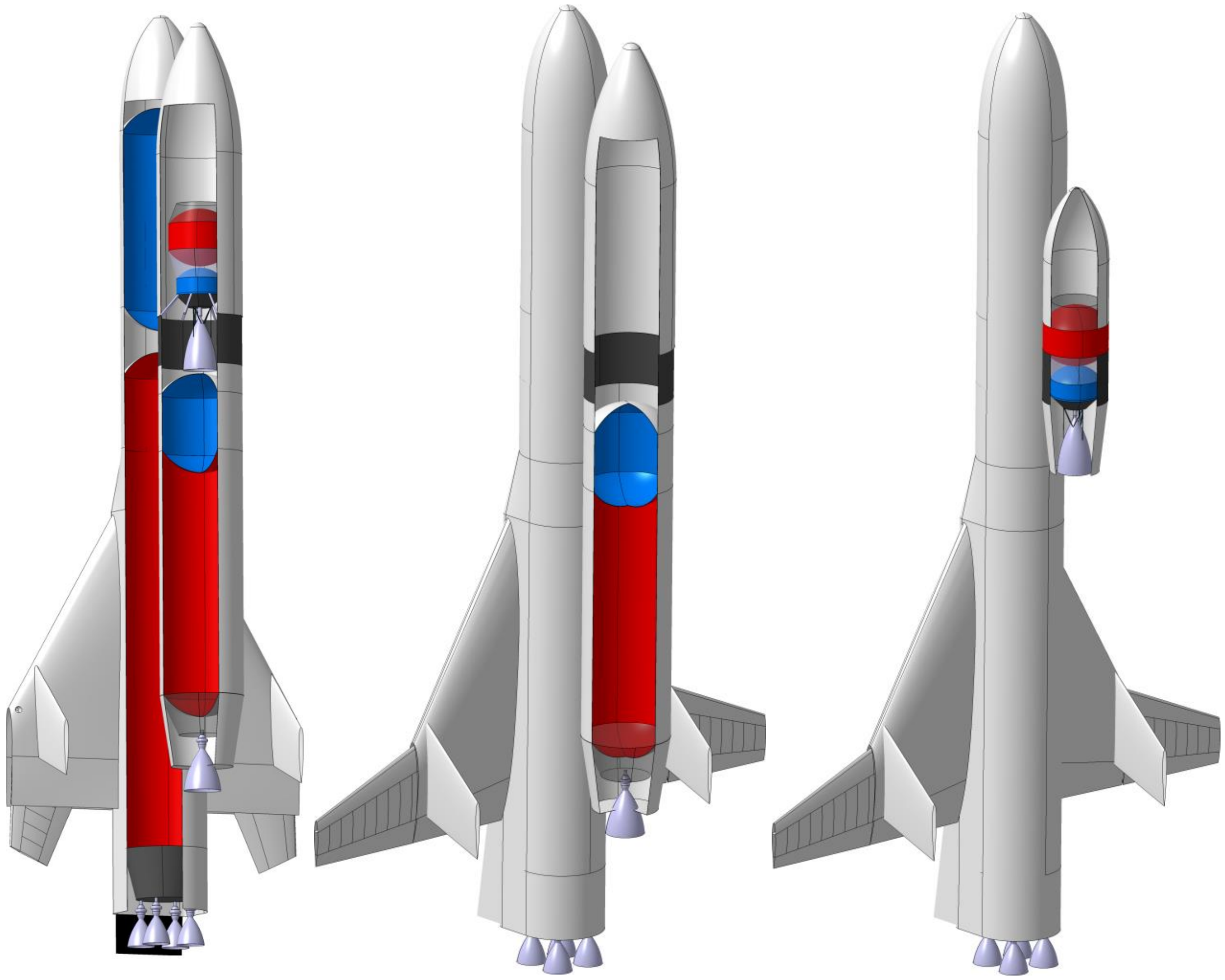


Figure 12: Launcher architecture sketches of RLVC4-B configuration as 3STO (left), TSTO (center) and Mini-TSTO (right)

Table 6: RLVC4 launcher characteristics for different engine types and missions

		Version SC **LH_2-SC**		**Version G** **LH_2-GG**	**Version MH** **Hybrid-GG**	**Version MM** **LCH_4-GG**
RLV Stage		**H370**		**H450**	**C620**	**C800**
Dry mass		78.8 t	92.2 t	95.1 t	84.5 t	105.4 t
Total Propellant		378.2 t		456.8 t	628.2 t	810.2 t
Structural index		20.8 %	24.3%	20.8 %	13.5 %	13.0 %
Engines		4 x SLME		10 x Prometheus-H	11 x Prometheus-M	15 x Prometheus-M
Fuselage Diameter		5.4 m		6.0 m	5.4 m	6.0 m
Length		59.1 m		64.5 m	54.5 m	56.3 m
Mission	GTO	LEO-ISS	SSO	GTO	GTO	GTO
2nd stage ELV	**H150**	**H150**	**H14**	**H152**	**H152**	**C230**
3rd stage ELV	**H14**	**-**	**-**	**H14**	**H14**	**H14**
GLOM	665.0 t	663.2 t	496.86 t	760.1 t	920.8 t	1203.2 t
Payload	13.9 t	> 21 t	5.5 t	14.1 t	14.0 t	14.0 t
Payload ratio	2%	3.16%	1.1%	1.85%	1.5%	1.16%

The baseline version of the RLV-C4 concept to be equipped with staged combustion rocket engines is called variant B featuring the LOX/LH2 propelled SLME engines (Table 1 and Figure 1). Figure 12 shows the RLVC4-B configuration in two sub-variants. on the left a swept-wing concept described in more detail in [14, 15] and a more recent fixed wing design in the center and right part with updated aerodynamic control features (see following section 4.5). Characteristics of the RLV-stage like tank size and arrangement are identical for the sub-variants and all are equipped with 4 SLMEs. Currently, no decision has been taken on the preferred wing solution for this RLV. The more compact swept-wing variant is obviously beneficial during launcher ascent and in reentry avoiding interactions between the nose shock and the wing's leading-edge shock [13, 26] but on the downside is coupled with increased complexity and potentially weight.

Since development of advanced closed cycle engines like the SLME has not yet started in Europe, it is also of interest to understand how an RLV powered by a modern gas-generator engine is performing. The G-type (for gas generator) utilizes a recently proposed variant of the PROMETHEUS engine with LOX-LH2 propellant combination and the characteristics listed in Table 5. In order to limit the length of the RLV-stage and increase the lift-capabilities during re-entry, the fuselage diameter is set to 6 m (Table 6).

The (M)ethane proposal for the RLVC4 consists of two different versions: The first ("MH") uses LOX/LCH4 in the RLV-stage and LOX/LH2 in the second stage while the second ("MM") utilizes LOX/LCH4 in both the RLV and second stage. The propulsion is based on the PROMETHEUS engine currently in development as well as the drafted PROMETHEUS-H in the MH-variant. While the second stage in the MH-version is identical to the G-version, the MM variant requires a complete redesign. With 230 tons of propellant in the second stage, it is significantly heavier. Therefore, this stage requires two PROMETHEUS engines to inject the third stage and payload into the transfer orbit. This design change decreases the compatibility with the current Ariane 6 LLPM significantly and is therefore not considered favorably.

The main overall dimensions of the launcher and RLV-stages are listed in Table 6 and some additional information has been provided in [3, 17].

4.3 Comparison of RLVC4-types in GTO-mission

All launcher types are dimensioned for a similar payload mass of 14000 kg to GTO. The gross lift-off mass almost doubles from LOX/LH2 with staged combustion (B) to LOX/LCH4 with a gas generator cycle (MM) in RLV and 2nd stage (Figure 13). The SLME-powered B-version achieves the lowest GLOW of 665 tons in the 3STO GTO-mission (Figure 13) with propellant loading for the RLV limited to less than 380 t [3, 17].

Due to the lower Isp-performance of the PROMETHEUS-H engine compared to SLME, the propellant mass of the RLV-stage is to be increased to 450 tons, which is an addition of 80 tons compared to the III-B version. 10 PROMETHEUS-H engines are used for the ascent until stage separation which needs to be increased due to the almost similar size of the 2nd H150 stage. but 20 s lower Isp.

Given the even lower specific impulse of the LOX/LCH4 propellant combination, these versions require significantly more propellant than their hydrogen counterparts. Because of the LOX/LH2 second stage, the MH version demand is still relatively modest with 620 tons while the MM RLV-stage propellant mass is already to be increased to above 800 tons, more than twice the SC-variant with SLME.

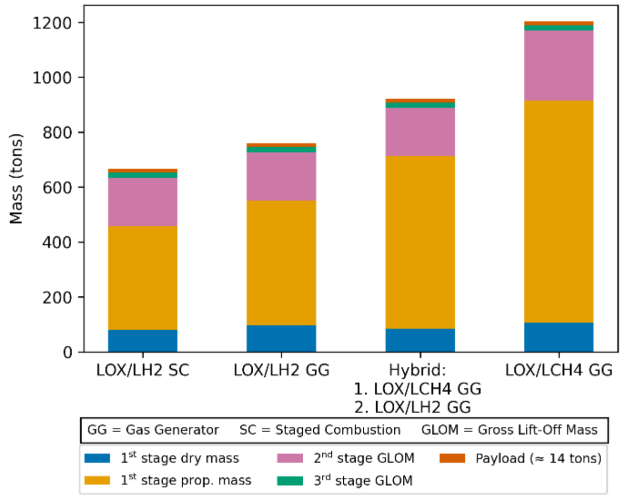


Figure 13: Mass comparison of RLVC4-types, GTO-mission

Consequently, 11 (MH) and 15 (MM) PROMETHEUS-M engines are used on the RLV-stages respectively. A concern with regards to the MM version is the number of engines that necessarily have to be installed in the rear skirt. A preliminary study of possible engine arrangement for the RLVC4-MM first stage with 6 m diameter and 15 PROMETHEUS-M engines is shown in Figure 14.

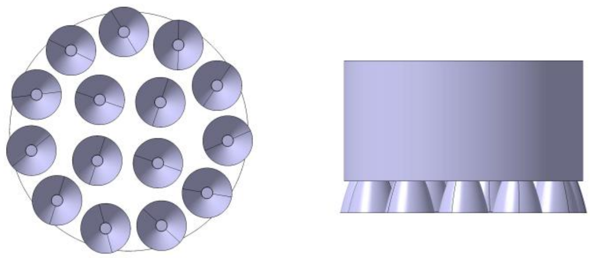


Figure 14: Possible engine arrangement for MM first stage with 15 PROMETHEUS-M engine

Here, the outer engines are installed without gimballing capability whereas the center engines can gimbal about 7° each independently. However, feasibility on ascent control is still to be validated. Another potential problem is the fact that the outer engines overlap the rear skirt by a small extent. A similar situation exists with the SuperHeavy CH4-booster of SpaceX (see e.g. [2]). Increasing the diameter of the rear skirt only towards the engine bay could allow for a more convenient arrangement, however, reducing L/D in subsonic gliding, detrimental to achieving the "in-air-capturing"-target.

Increased bulk density of the propellant combination LOX-/LCH4 compared to LOX/LH2 helps in limiting the growth of the methane RLV-stages' size and dry weight. Nevertheless, the estimated dry weight of the RLVC4-MM is more than 30% above the -B variant's with staged combustion engines. The increased complexity when moving from 4 main engines on the RLV-booster stage to 15 would also have a negative impact on production and refurbishment costs.

4.4 Orbit injection to GTO, LEO-ISS, SSO

The transfer into GTO with a TSTO is straightforward: the insertion is done directly and following SECO the payload is in the specified GTO. Opting for a 3-stage architecture is mainly attractive for the GTO mission (or beyond) because a much smaller inert mass will have to be injected in a high-energy orbit. However, the insertion with a 3STO calls for additional measures in order to ensure that the uncontrolled descent of the expendable second stage safely occurs in the Pacific Ocean.

Thus, the ascent phase is split into two steps: first, the second stage plus third stage and payload are injected into an intermediate orbit with an apogee height of 400 km and a perigee height of 35 km [14, 18]. The large expendable cryogenic 2nd stage should be designed not to reach a stable orbit but to splash into the Pacific safely off the American West coast. Following separation of the third stage from the second stage the third stage coasts along a ballistic trajectory. Slightly before crossing the equator the third stage is ignited to insert the payload into a GTO with 350 km perigee and 35786 km apogee and approximately 6° inclination.

In case of LEO-missions the launcher can best be operated as TSTO. The orbital injection conditions of the expendable 2nd stage will require an active deorbiting of the H150. In case of an ISS-resupply mission the stage's splashdown is foreseen in the Pacific Ocean in the vast remote areas east of New Zealand [15]. The latest assessment as listed in Table 6 considers an astronautic mission assuming in modeling the addition of a launch escape system instead of conventional fairing. The crew compartment assumptions are very similar to the earlier assessment in [3] but the more powerful upper stage of the RLV-based TSTO reaches roughly 3 tons better payload performance than A64 with its ULPM. Thus, a more robust system is enabled which could have the capability of larger, deep-space missions.

The so-called "Mini-TSTO" is a new launcher variant best representing the "one-[RLV]size-fits-all"-philosophy. Usually, payload mass requirements in SSO are modest, not exceeding 5 t. Therefore, the H14-upper stage with Vinci has been assessed as the only second stage attached to the RLV. As visible in Figure 15 the stage easily fulfills this role with a separated payload in circular 700 km orbit around 5.5 t. The separation Mach-number of the RLV-stage would in this case increase to around 16, well beyond the GTO- and ISS-missions for which heavier upper stages are to be accelerated.

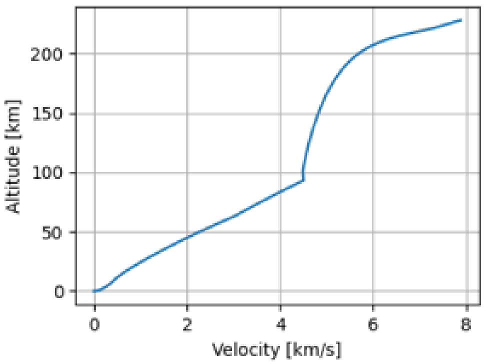


Figure 15: Ascent profile of RLVC4-III-B configuration in 100 km x 500 km, 97.4° transfer orbit for small SSO-mission

This condition will require an adapted, likely heavier TPS on the RLV for safely performing its reentry. Such preliminary design has not yet been carried-out. However, an assumed additional mass contingency for the RLV has been considered. Alternatively, a lower separation Mach-number could result in reduced payload mass, still fully sufficient for SSO-missions.

4.5 Feasibility of RLV flight control during reentry

Designing an aerodynamically controlled vehicle reentering the atmosphere at hypersonic velocity, subsequently slowing down to subsonic velocity and finally reaching equilibrium gliding flight conditions, is a very challenging task. The stage covers a vast range of flight conditions at which it has to be controllable to allow a safe reentry while also fulfilling the gliding flight requirement necessary for executing a successful In-Air-Capturing maneuver [22]. Understanding the flight dynamics of a winged vehicle reentry already in the early design phase is necessary to identify challenges with regards to controlling and actively steering such a high-performance vehicle in order to arrive at a feasible and robust design that does not fail to converge in later design iterations.

Therefore, the LOX-LH2 version with staged combustion engines, concept RLVC4-III-B, was subjected to a thorough analysis of reentry aerodynamics and its effect on flight dynamics. This includes studying the impact of design changes to the initial aerodynamic configuration (still visible in Figure 12 at left), investigating the dynamic motion and stability of the stage and, finally, studying control possibilities and simulating 6-DOF flight maneuvers [17, 18].

The wings feature two vertical fins of increased size to improve directional stability. Furthermore, this RLV version features a rather large bodyflap extending over some part of the lower fuselage in order to improve pitch trim characteristics. The bodyflap (deflected in Figure 16) will be extended only during hypersonic reentry and at high AoA. Adding sidewalls to the flap further helps reducing the vehicle's yaw instability. Further, the vertical stabilizers are extended significantly below the wing because in hypersonic reentry with high AoA-flight, those portions located on top of the wing show limited efficiency due to shading effects. A preliminary assessment of the aerodynamic coefficients indicates that pitch maneuvering is stable in almost the complete reentry flight while yaw movement with respect to sideslip has become stable in subsonics but remains unstable in the hypersonic regime [17, 18].

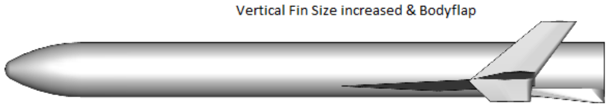


Figure 16: Side view of updated RLVC4-III-B configuration in hypersonic reentry mode with large deflected bodyflap

A simplified approach to determining flight dynamics at certain flight points is the linearization of the equations of motion. This method allows to derive linear, time-invariant equations that are valid in a range of small disturbances around an equilibrium point of the vehicle (usually trim points) and are suitable for describing the flight dynamics in this area [17, 18]. By checking the real and complex parts of the eigenvalues at specific points in the trajectory one can determine if the vehicle is stable or unstable throughout the flight regime.

References 17 and 18 show that the longitudinal motion is mostly stable throughout the trajectory. Only in the region of Mach 5 to Mach 3 at high to medium AoA and in the region of maximum dynamic pressure active control is crucial to keep the commanded AoA profile. Contrary to that, the lateral motion is unstable throughout most of the flight. In fact, even down to Mach 0.5 there is at least almost always one unstable eigenvalue.

The analysis of dynamic stability in [17, 18] has revealed the need for an active and fast control in order to stabilize the vehicle throughout reentry flight. For an early check on feasibility, an active control loop has been simulated with full state feedback and infinitely fast actuators. This simplification was deemed suitable at the early design state. Including actuator and sensor models could be focus of future work. Furthermore, no wind or atmospheric disturbance was assumed yet.

For pitch control, the inner trailing edge flaps are used, for roll control the outer flaps are used, and for yaw control the vertical fins had been assumed to be deflectable entirely. Additionally, an RCS system in the nose (similar to the Space Shuttle) is required for exo-atmospheric control. For each axis, 4 RCS engines with a thrust of 650 N each are foreseen. The preliminary, simplified 6-DOF analysis presented in [17, 18] using RCS and aerodynamic control surfaces to steer the vehicle, shows that the reference profile in AoA and bank angle can be followed and, thus, the updated configuration is feasible in principle. However, a more detailed simulation including wind, actuator models and realistic sensor models will increase the insight and might trigger further design improvements.

5 EVALUATIONS AND COST ESTIMATIONS

5.1 Overall performance and mass comparison

This paper has not the intention of comparing different launcher concepts in a generic way with exactly the same modelling assumptions for engine- or structural efficiency and identical performance requirements. Nevertheless, the stage and engine building blocks and mission constraints have overall sufficiently good similarities that a comparison of the launcher types makes sense.

Maximum payload performance, such as that required for GTO missions or manned ISS flights, requires the use of the XXL-type BB-VTVL-launchers with both core and upper stages expendable. Despite all the differences in architecture, this configuration's performance is pretty close to the one of VTHL 3STO. On the other end of the spectrum, the SSO-mission can be served by the M- or L-versions of the BB-TSTO or by the Mini-TSTO of the VTHL. Achievable payload ratios of the VTVL are in general below the VTHL using IAC with the exception of the LOX-LH2 M-size BB reaching 1.5% above 1.14% for the Mini-TSTO (see Table 6). This is the consequence of using a large RLV-stage also for smaller missions.

While the relevant missions can be served by all investigated (family) concepts, the range in necessary lift-off weight could be vast. In the GTO-example almost the same separated payload could be lifted by RLV-C4-IIIB with LOX-LH2-staged combustion propulsion at GLOW 665 t or by VTVL with methane gas-generator type in the XXL-configuration and reusable side-boosters at 1842 t (+177%). The launcher with reusable first stage and lowest lift-off weight reaching still meaningful payload is the LOX-LH2-M-size BB with slightly below 300 tons.

5.2 Preliminary cost comparison

5.2.1 Non-Recurring Cost (NRC)

Launch System NRC have been estimated using DLR cost estimation methods. In case of BB-launcher families it has been assumed that each of the building blocks need to be developed and qualified only once as RLV- or ELV-stages. For any additional member of the family only system-engineering- and potentially launch site costs are considered. Obviously, this is a simplification, however, suitable in this early approach when elements are not yet well defined.

It is interesting to note that the quite diverse launcher concepts come relatively close in the estimated development costs. Building-Block launcher families including all potential systems from S- to XXL-class would be expensive. If, however, the BB-families are restricted to its most promising members, only 3 to 4 launchers would remain and NRC becoming much more affordable.

With the development costs of the advanced concept RLV-C4-IIIB set as reference (=100%), the range of all studied concepts is found relatively close between 80% to 110% (Figure 17). Configurations using methane propellant are slightly more expensive because of the complex and heavy multi-engine bay, increased development risks from European perspective and assuming Mira still to be developed while Vinci already fully qualified [6]. For similar reasons the relatively large staged-combustion cycle engine SLME is pushing costs of the reference concept somehow upward.

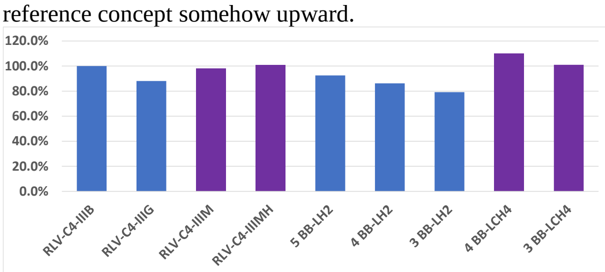


Figure 17: Relative comparison of total launch system NRC

The differences in NRC are plausible and also not negligible. However, it is to be kept in mind that such estimations are a prognosis of future activities which by its very nature has an uncertainty of at least 10%. Thus, it is far too early to establish an NRC-ranking.

5.2.2 Recurring Cost (RC)

The costs per flight or launcher recurring costs are relatively difficult to compare. Several uncertainties are to be considered: the number of launches per year, the distribution of missions and payload classes for the launcher families and in case of the RLV-stages, the expected recovery and refurbishment costs and the number of reuses per stage and engine.

The launch market situation in 20 or 30 years is unknown and the related uncertainty is probably best to be addressed by running several scenarios. This, however, is beyond the scope of this paper and should be addressed in future work. Most of the partially reusable launchers investigated reach specific transportation costs between 50% and 60% lower than the current generation of European space transportation. An adequate usage of payload capacity remains essential for achieving low specific launch costs. A dedicated mission with a smaller element from the BB-families might become possible and even affordable. However, its specific launch cost could easily exceed that of multi-satellite deployments.

6 CONCLUSION

Different options for the next generation of European RLV-launchers have been investigated. Option 1 regarding "Building Block" families with 1st stage as VTVL-RLV are found technically feasible when assuming 3-stage and 2-engine BB-elements. In case of LOX-LH2 five different launchers are identified while for LOX-LCH4 only four different sizes are feasible. The GLOW of LOX-LCH4 is always found roughly 80% above LOX-LH2, although the payload capacity of the methane concepts as RLV is constantly significantly lower.

The second option with a "Big-Size" VTHL-RLV and side-mounted expendable upper stages is also confirmed to be technically feasible as 3STO to GTO, TSTO for heavy payload to LEO-ISS and as an innovative Mini-TSTO for smaller SSO-missions.

The preliminary launcher system sizing approach revealed that seemingly contrarious options have many characteristics in common if the number of launcher configurations in the BB-family are limited to a maximum of 3 different variants. Further refinements of the models are recommended for future work.

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