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Evaluating launcher options for Europe in a world of Starship

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ABSTRACT

As soon as the SpaceX Starship & SuperHeavy launcher configuration is operationally available it likely will cause a fundamental shift to space transportation. In a first step, the paper provides a thorough technical analysis of Starship's estimated capabilities in its early operational phase, based on independent modeling with openly available data.

The main part of the paper is dedicated to the technical evaluation of European options for serving a roughly similar payload class above 20 Mg up to approaching 100 Mg in single launch to LEO. A launcher system analysis looks into Ariane 6 evolution options and explores the technical limits based on the assumption of expendable stages. A significantly better performance perspective can be achieved through a completely new architecture. In case of these new architecture launchers, all first stages are reusable and exclusively liquid cryogenic propellants are chosen. Fully reusable configurations have been addressed in the small ESA-funded PROTEIN-study for which some complementary concepts are summarized.

The different launcher options show a broad range in payload performance. As these diverse vehicles come with significantly different cost, the NRC and RC are modeled for reasonable European heavy-lift transportation scenarios.

The paper concludes with a comparative evaluation of main technical characteristics of the launch vehicle options and an indication of promising development roadmaps.

Nomenclature

Subscripts, Abbreviations	
3STO	Three-Stage-To-Orbit
AoA	Angle of Attack
CAD	Computer Aided Design
DRL	Down Range Landing
EHLL	European Heavy Lift Launcher
GLOW	Gross Lift-Off Weight (provided as mass)
HLS	Human Landing System
IFT	Integrated Flight Test of Starship
LCH4	Liquid Methane
LEO	Low Earth Orbit
LH2	Liquid Hydrogen
LLPM	Lower Liquid Propulsion Module (of Ariane 6)
LNG	Liquified Natural Gas
LOX	Liquid Oxygen
MECO	Main Engine Cut Off
MR	Mixture Ratio
NRC	Non-Recurring Costs
OTV	Orbital Transfer Vehicle
RC	Recurring Costs
RCS	Reaction Control System
RLV	Reusable Launch Vehicle
RTLS	Return To Launch Site
RVAC	Raptor VACuum variant according to SpaceX nomenclature
SLB	SpaceLiner Booster stage
SLME	SpaceLiner Main Engine
SLO	SpaceLiner Orbiter stage
SLP	SpaceLiner Passenger stage
TPS	Thermal Protection System
TSTO	Two-Stage-To-Orbit
TVC	Thrust Vector Control
ULPM	Upper Liquid Propulsion Module (of Ariane 6)
VTHL	Vertical Take-off and Horizontal Landing
VTVL	Vertical Take-off and Vertical Landing
e.c.	economic conditions

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1. Introduction

The progress made by SpaceX in flight testing its Starship & Super-Heavy launcher configuration is remarkable and could soon trigger a fundamental shift to space transportation. Announced as a fully reusable rocket to LEO with unprecedented payload capacity, the global competitive landscape could be revolutionized with the realization of this super-heavy system.

The main part of the paper is dedicated to the technical evaluation of European options in serving a roughly similar payload class up to above 50 Mg in single launch to LEO. Typical future applications which would require a significant number of missions per year are the deployment of mega-constellations, space solar power or deep-space human and large-scale robotic exploration. The small ESA-funded PROTEIN-study looked into some of the potential launcher design possibilities.

Annual payload delivery of several hundred tons is well-above the design capabilities of Ariane 64. Thus, either a major evolution of A6 or completely new launcher architectures are required. A launcher system analysis looks into such Ariane 6 evolution options and assesses the technical limits based solely on the assumption of expendable stages.

However, a significantly better performance perspective can be achieved through a completely new architecture based on building blocks that already exist or are under development. Reusability with VTVL and VTHL are then viable options with impressive payload capacity in case of expendable cryogenic upper stages. TSTO-launchers are selected as the baseline, but the addition of a third or orbital transfer stage could be attractive for certain high-energy missions. In case of the new architecture launchers, all first stages are designed reusable and exclusively liquid cryogenic propellants are chosen.

2. Understanding SpaceX' Starship launcher

Elon Musk's bold vision of transporting humans in significant numbers to Mars [2] is driving the development of a space transportation system to dimensions and capabilities never seen before. The concept has perceived several technical evolutions in the last 10 years [3,4]. The SpaceX Starship & SuperHeavy is now flying in prototype configuration and is continuously progressing in mastering its ascent and atmospheric reentry mission. Successful capturing of the Super-Heavy first stage could be achieved multiple times after performing a complex RTLS-maneuver. The fully reusable system is still not yet operational and its present payload capability is limited, not being in focus of the current development process.

Once becoming operational, the two-stage Starship launcher could bring a fundamental shift to global space transportation. The circumstances need to be well understood and therefore require, as a first step, a thorough technical analysis of Starship's actual capabilities in its early operational phase, based on openly available data. Data used for DLR's independent analyses are briefly summarized in the following sections. Note, these analyses have been limited to the LEO ascent and return capabilities of Starship & SuperHeavy. SpaceX has been contracted by NASA to develop and operate a Starship derivative as Human Landing System (HLS) on the Moon within the Artemis-program [5], requiring additional features (like tankers for on-orbit refilling) which are not analyzed.

2.1. Architecture

The SpaceX Starship & SuperHeavy is designed to become the first fully-reusable space transportation system to orbit. Defined as TSTO with both stages in tandem arrangement, the vehicle uses the propellant combination LOX-LCH4. Analogous to the operational Falcon 9, the engines of the main propulsion system are similar on both stages with the main difference being the nozzle expansion ratio. The engines have been named Raptor and utilize the hydrocarbon propellant methane, which one day should be produced in-situ on Mars [2]. The medium specific performance of LOX-LCH4 is partially offset by selecting for Raptor the closed cycle at extremely high operating conditions.

The primary purpose of the Starship assembly is to support deep-space human exploration and colonization and thus requires a payload capacity beyond 100 tons in LEO. GLOW is above 5000 tons, roughly twice that of the Saturn V in the Apollo-moon program. (Currently, the GLOW-ratio of the two launchers is around 1.7 but an operational future Starship is probably to reach or exceed the factor of 2.) The total length of the launcher exceeds 120 m with a fuselage and tank diameter of 9 m. A more detailed technical description of SpaceX' Starship & SuperHeavy and how the launcher has been remodeled by DLR is described in Ref. [6].

2.2. Raptor main propulsion system

The SpaceX Raptor operates in Full-Flow Staged-Combustion (FFSC) cycle which is using the complete propellants to drive the turbopumps. Raptor is the first FFSC rocket engine ever flown while the RD-270 (8D420) was already in the Soviet Union in the late 1960s the first ever designed and tested FFSC-engine by Energomash.

Precise engine performance data of the Raptor 2 has not been published. Therefore, an independent DLR-analysis of Raptor 2 has been performed using the rocket cycle calculation tool RPA. The SpaceX announcement of 230 tons (2260 kN) of sea-level thrust [7], the chamber pressure of 30 MPa with assumed mixture ratio of 3.6 and nozzle exit diameter 1.3 m serve as guidelines for the calculation.

The Raptor 2 data based on DLR-calculations while at the same time in overall agreement with references [7,8] are a good working baseline for launch vehicle analyses. Shortly after lift-off the SuperHeavy & Starship-launcher significantly throttles back to reduce loads in the maximum dynamic pressure regime. This change in mass flow might be achieved by adapting mixture ratio (MR), chamber pressure or both. Conditions for the engine with same thrustchamber geometry working at 250 bars are listed in the right column of Table 1 showing a throttling capability in sea-level thrust of almost 18 %. The sensitivity of MR-variation on the Raptor 2 performance is depicted in Ref. [6].

SpaceX announced in 2024 the production start of its latest variant Raptor 3 [9] with capability of operating at very high chamber pressure of 350 bars. This engine should incorporate highly innovative features, like all components regeneratively cooled and at the same time simplifications and significantly increased T/W-ratio. Although, functionality of the features is publicly not yet exactly known, nevertheless, estimation of the engine overall performance is possible assuming same thrustchamber geometries as for Raptor 2 and nominal chamber pressure of 35 MPa.

Calculated thrust levels of Raptor 3 are increasing by 16.6 % for the sea-level variant (Table 2, 2613 kN vs. 2260 kN). Any further increase is not compatible with the geometry constraints and announced chamber pressure. Therefore, the posted thrust [9] of 280 tf (2746.8 kN) is remarkably 4.98 % above the calculated thrust at MR = 3.8. Further approaching stoichiometric conditions does no longer surge thrust as the increase from 3.6 to 3.8 is already pretty small. The photograph provided in Ref. [9] showing all Raptor variants 1, 2, 3 does not indicate changes in the thrustchamber geometry. In any case, a surged-up size of the chamber would be hardly compatible with an integration of 33

Table 1
SpaceX Raptor 2 engine (sea level variant) in DLR-calculated technical data.

Mixture ratio [–]	3.6	
Assumed nozzle area ratio [–]	32	
Chamber pressure [MPa]	30	25
Mass flow engine [kg/s]	663.4	553.9
Thrust at sea level engine [kN]	2123	1747
Thrust in vacuum engine [kN]	2260	1884
Specific impulse at sea level [s]	326.4	321.6
Specific impulse in vacuum [s]	347.4	346.8

Table 2
SpaceX Raptor 3 engine (sea level variant) in DLR-calculated technical data.

Mixture ratio [−]	3.6	3.8
Assumed nozzle area ratio [−]	32	
Chamber pressure [MPa]	35	
Mass flow engine [kg/s]	765.8	774.9
Thrust at sea level engine [kN]	2476	2479
Thrust in vacuum engine [kN]	2613	2617
Specific impulse at sea level [s]	329.6	326.3
Specific impulse in vacuum [s]	347.9	344.3

Raptors in the SuperHeavy base area. However, it is interesting to note that the Raptor 3 operating at 35 MPa with its nozzle assumed at expansion 32 would already be slightly underexpanding at sea-level. This situation is not really advantageous from the performance perspective but is dictated by severe geometry constraints of the SuperHeavy first stage without modification of the base area.

The corresponding engine of Raptor for the upper stage with increased nozzle is called *RVAC* by SpaceX. This engine should have an exit diameter of 2.3 m [8]. Thus, nozzle geometry has been used again as baseline assumption in combination with the reasonable hypothesis that other key-parts of the engine (e.g. turbopumps, injector, throat) are similar to the sea-level variant. An expansion ratio of around 100 is consistent with the stated exit diameter. Reference 5 provides key performance data of *RVAC* calculated by DLR under these constraints.

Table 3 lists the calculated performance data of RVAC assuming the promising upper stage mixture ratio of 3.4. According to SpaceX's website, the thrust should reach 258 t_f (2530 kN) [8], at least approximately 5.7 % higher than 2385 kN computed using RPA at MR of 3.8. An explanation for this deviation is not readily available but turns out to be not critical for the accurate trajectory simulations of reference [6]. Therefore, the values of Table 3 have been used in the system performance assessment.

Despite some deviations of the calculated thrust levels compared to published information on Raptor, the DLR performance estimation is in overall good agreement with the announcements.

2.3. Expected payload performance of Starship V1 through V3

The ambitious interplanetary mission of Starship will require in the future LEO payload capacity well beyond 100 tons. The requirement itself does not assure that the current versions of the launcher actually achieve this target. An independent assessment has been performed by DLR [6] with validated models based on recalculations of the performed Integrated Flight Tests (IFT) of Starship's and Super Heavy from Boca Chica in the US. The models calibrated with telemetry data from IFT#2–4 are described in Ref. [6] and have been used to extrapolate the Starship's payload to orbit performance for fully reusable operations. The Starship's ascent trajectory has been modified from suborbital to orbital and the return trajectory adapted to ensure the SuperHeavy's RTLS-maneuver consuming a massive amount of fuel after separation.

Two configurations were investigated by DLR: the Starship V1 of the early flights and an enlarged Starship representing the vehicle in an early operational phase. This configuration was announced in April 2024 as V2 [10] with larger propellant loading, a slight increase in length, and the use of the more capable Raptor 3 engines. The configurations' key parameters are listed in Ref. [6]. The maximum possible payload has been estimated for the direct ascent into a 250 km × 300 km orbit with an inclination of 26°, flying directly eastward from the SpaceX's Boca Chica facility. These orbit parameters are close to the reference mission considered for European heavy lift-launchers (see section 3.1).

Meanwhile, SpaceX is following a more incremental approach with an intermediate version as tested in 2025 called now V2 and the larger variant formerly dubbed V2 is now the future V3 [11]. In the absence of more detailed information, this X-post by Elon Musk could be used as semi-official reference to the size and performance of Starship and Super Heavy as expected by SpaceX in 2025. A summary of size and performance data based on independent analyses [6] and announcement [11] is listed in Table 4.

The used Starship model in Ref. [6] does not reserve mass for movable payload bay doors or other payload deployment mechanisms. The actually deployable payload is of more practical interest and is called here net payload and considers mass contingencies for the payload attachment structure and for sufficiently large payload bay doors.

The assumptions of the independent DLR calculations for V1 and V3 published in Ref. [6] remain in overall good agreement with only minor deviations to the more recent statement of [11]. DLR's assessment tends to be a bit more optimistic on payload performance than the latest announcement (Table 4) although it is to be acknowledged that the assumed orbit parameters of reference [11] are unknown. If the Raptor 3 engine performance data at 35 MPa chamber pressure (see Table 2) could be maintained during the major part of the ascent flight, a payload capability of Starship V3 configuration of 100+ Mg might be achievable. With this immense capacity, the fully reusable space transportation system would significantly surpass the largest recent launcher test-flown in 2022, the expendable Block 1 Space Launch System (SLS).

2.4. Transfer scenarios

The super heavy payload capability offered in the future by Starship & SuperHeavy will not necessarily be deployed in single LEO destinations. In the exploration missions the Starship should be refueled in orbit before being reignited for deep-space transfers as intended in the Artemis-program [5].

In other Earth-related scenarios a large number of diverse and usually significantly smaller payloads in the hundreds or thousands could share a single mission to LEO. This might be attractive because of the potential for dramatic launch cost reductions and if offered as service with high availability. These payloads will then require a multitude of transfers using dedicated vehicles for injection into final destination orbits. Currently, such transfer vehicles, or OTV are not yet fully operational in the market. Although, these vehicles likely will play an important role in the future they are not regarded in this paper.

It shall be noted that using multi-payload dispensers and satellite onboard propulsions may respond to the requirement of deploying numerous payloads during one mission. The SpaceX example of such deployment scheme is the Starlink constellation that makes use of electric Hall-effect thrusters. However, such an approach adequate for multiple satellites that operate from similar orbital plane conditions as part of a constellation is less attractive for reaching diverse high energy orbits. Although technically feasible, the required amount of transfer time has a negative impact on-satellites' orbit-lifetime and hence revenue generation.

3. European heavy-lift launcher options

Large European space infrastructures as well as deep space missions would require significantly more performant space transportation in the foreseeable future compared to what exists today. ESA is starting to define and evaluate a "hub and spoke" space logistics network to reach the final orbits (e.g. constellations phasing, exploration missions …) and

Table 3
SpaceX Raptor 2 RVAC engine (vacuum variant) calculated technical data at 30 MPa and MR = 3.4

mass flow	kg/s	655.77
sea level thrust	kN	1943.05
vacuum thrust	kN	2369.64
sea level specific impulse	s	302.14
vacuum specific impulse	s	368.48

Table 4
Payload and launcher mass of the V1, V2 and V3 Starship & SuperHeavy in Ref. [11] and DLR modeling [6].

Starship variant [11]	V1		V2 [11]	V3	
	DLR modeling [6]	[11]		DLR modeling [6]	[11]
Propellant mass	4500 t	4450 t	4750 t	5150 t	5250 t
Lift-off mass (without payload)	4931 t	no information	no information	5596 t	no information
Lift-off thrust	70 MN	69.65 MN	69.65 MN	81.8 MN	80.83 MN
Net estimated payload LEO	**45 t**	**~15 t**	**~35 t**	**100 t**	**100 + t**

provide transportation support for in-orbit servicing (see e.g. Ref. [12]).

Together with exploration ambitions to the Moon or interplanetary missions, this infrastructure will require efficient means of transportation from Earth to LEO. The technical concepts of future European heavy-lift orbital launch capabilities are linked to the following development targets: short-term (up to 2030), medium-term (after 2035 up to 2040) and longer term (around 2050). This classification will later also be used in the discussion of a potential development roadmap.

3.1. Reference mission

All configurations in this section assume similar key mission requirements.

- 250 km × 300 km with an inclination of 25°
- Launch site: CSG, Kourou, French Guiana

The orbit represents a suitable staging orbit for a translunar trajectory but could also be representative for large LEO-satellite constellations. The vehicles should be capable of performing secondary missions which were not investigated in the context of this paper. All upper stages are to be actively deorbited at the end of their Earth orbital missions to reduce the buildup of additional space debris. A contingency of fuel mass is reserved for this final part of the mission.

3.2. New generation launcher main propulsion

Appropriate options of new main liquid stages' propulsion are all based on cryogenic fuels: either liquid methane or liquid hydrogen. Baseline here is a DLR-performed systematic assessment of future European engine concepts suitable for RLV-applications [13]. This investigation considers hydrocarbon propellants as well as hydrogen under similar conditions and preliminary high-thrust engine designs of 2200 kN vacuum thrust level in gas-generator and staged-combustion cycle. On purpose, the assumed main combustion chamber pressures are less ambitious than those of SpaceX Raptor (see section 2.2). The Ariane 6 derived launcher concepts also make use of the existing cryogenic engines Vulcain and Vinci and solid propellant rocket strap-on boosters which are not described in this section.

3.2.1. Open gas generator cycle engine PROMETHEUS

PROMETHEUS is the precursor of a new European large-scale (100-tons class) liquid rocket engine using methane as fuel. Currently, the precursor of PROMETHEUS is under development. The calculated data in Table 5 have been generated by DLR to provide realistic performance of a full-scale engine for the launcher system design. 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.

Table 5
Calculated technical data of gas generator Methane engine as assumed for potential Ariane 6 liquid boosters.

Mixture ratio [–]	2.67
Chamber pressure [MPa]	11.77
Mass flow per engine [kg/s]	421
Expansion ratio [–]	16.4
Specific impulse in vacuum [s]	316
Specific impulse at sea level [s]	288
Thrust in vacuum [kN]	1305
Thrust at sea level [kN]	1200

3.2.2. Staged combustion cycle engine SLME

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 is the preferred design solution for the SpaceLiner Main Engine (SLME). It is interesting to note that the ambitious full-flow cycle is currently developed by SpaceX for its Starship & SuperHeavy with the Raptor-engine [4]. The Swiss company SoftInway and DLR jointly completed in 2024 a de-risk study for ESA on the SLME-type rocket engine [1] (Figs. 1 and 2).

The expansion ratios of the SpaceLiner booster and passenger stage/orbiter SLME engines are adapted to their respective optimums; while the mass flow, turbo-machinery, and combustion chamber 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 and 2400 kN, 2000 kN respectively for the passenger stage. All these values are given at a mixture ratio of 6.5 with a nominal operational MR-range requirement from 6.5 to 5.5. The full pre-defined operational domain of the SLME is shown in Ref. [1] including extreme operating points. Table 6 gives an overview about major SLME engine operation data for the nominal MR-range as obtained by cycle analyses [1]. Performance data are presented for two different nozzle expansion ratios: 33 and 59.

The lay-out of the SLME V7 is relatively conventional, like the Space Shuttle Main Engine (SSME), and a more advanced version with ox-rich powerhead mounted in-line with the main combustion chamber as on Raptor is under investigation for the SLME. The engine masses of V7 are estimated at 3500 kg with the large nozzle for the upper stage and at 3218 kg for the booster stage [1].

3.2.3. Staged combustion cycle engine derivatives

The calculated characteristics of the staged-combustion engines used by DLR for the fully reusable PROTEIN-related configurations are presented in Tables 7 and 8 and have been derived of SLME and the engine definitions in Ref. [13], however, with increased chamber pressure for the sea-level variant.

3.3. Expendable: Ariane 6 derived

Europe's Ariane 6 has been under development since 2014 in two configurations: A62 with two solid strap-on boosters and A64 with four solid strap-on boosters. Ariane 6 has performed its inaugural flight on July 9, 2024 from Kourou to medium inclined LEO, releasing several CubeSats [14]. This launcher has a central core consisting of the Lower Liquid Propulsion Module (LLPM) equipped with Vulcain 2.1 engine and providing space for about 154 Mg LOX/LH2-propellants. On top of this stage is the upper stage (Upper Liquid Propulsion Module or ULPM) which is propelled by the re-ignitable Vinci engine [15]. On the side of the LLPM solid boosters provide additional acceleration to the launcher. These P120C boosters house about 142 Mg solid propellant. The maiden launch was performed with the Ariane 62 version, using two solid boosters while the more powerful Ariane 64 using four P120C is

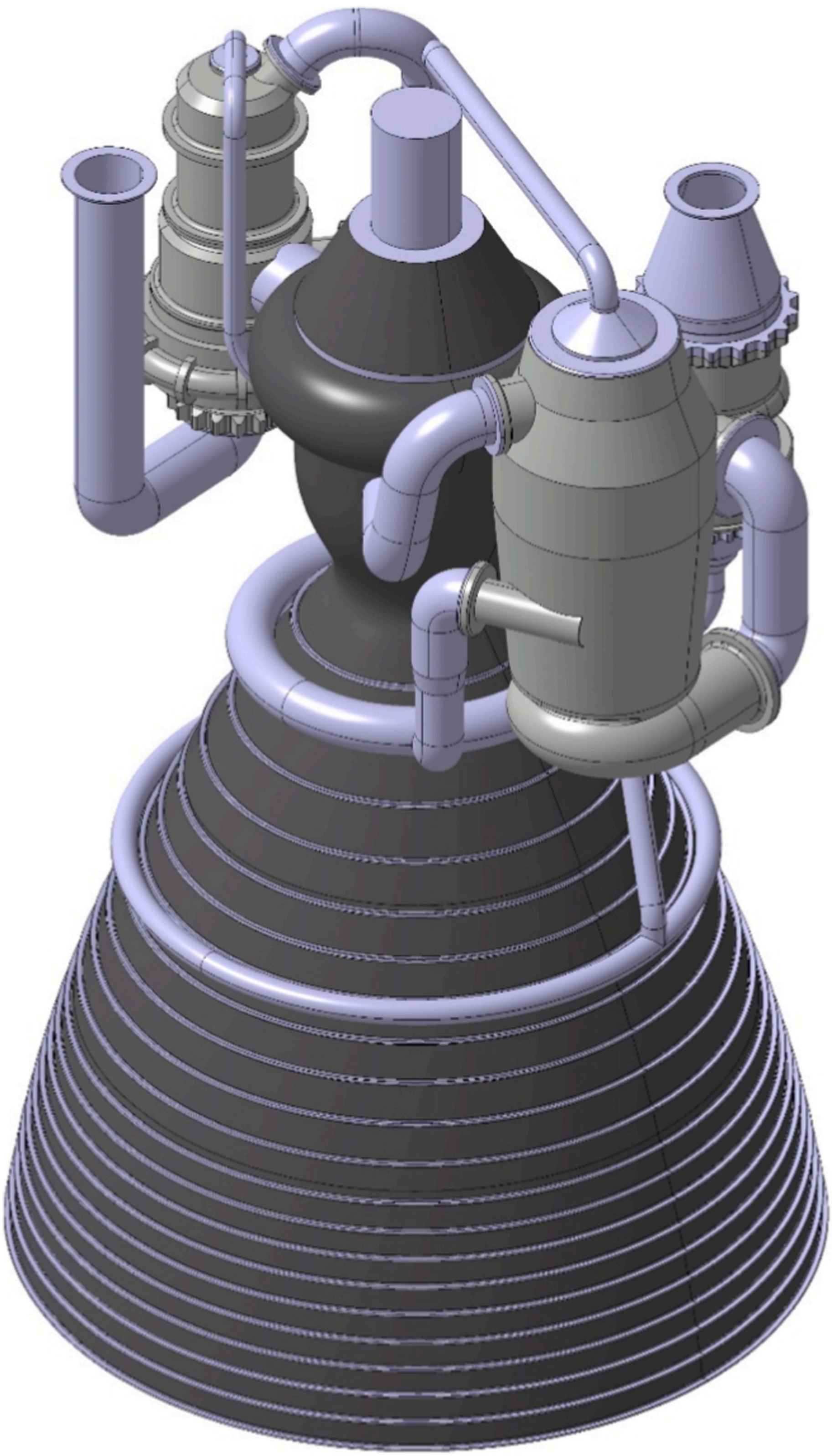

Fig. 1. CAD-drawing of SLME V7 with nozzle expansion ratio 33 [1].

expected to have its debut in the coming year.

For all the Ariane 6-derived configurations of this section a fairing mass of 2.6 Mg has been assumed.

3.3.1. *Ariane 6 "Block2"*

A first upgrade of the Ariane 6 called "Block2" is already under development and related work focuses on increasing the performance by increasing size and loading of the solid booster, now named P160.

Compared to the P120C, 14 Mg additional solid propellants are added to each booster and the casing is increased in length by 1 m while maintaining the same outer diameter. This improved performance version of Ariane 6 is of particular interest for the heavy lift role. The

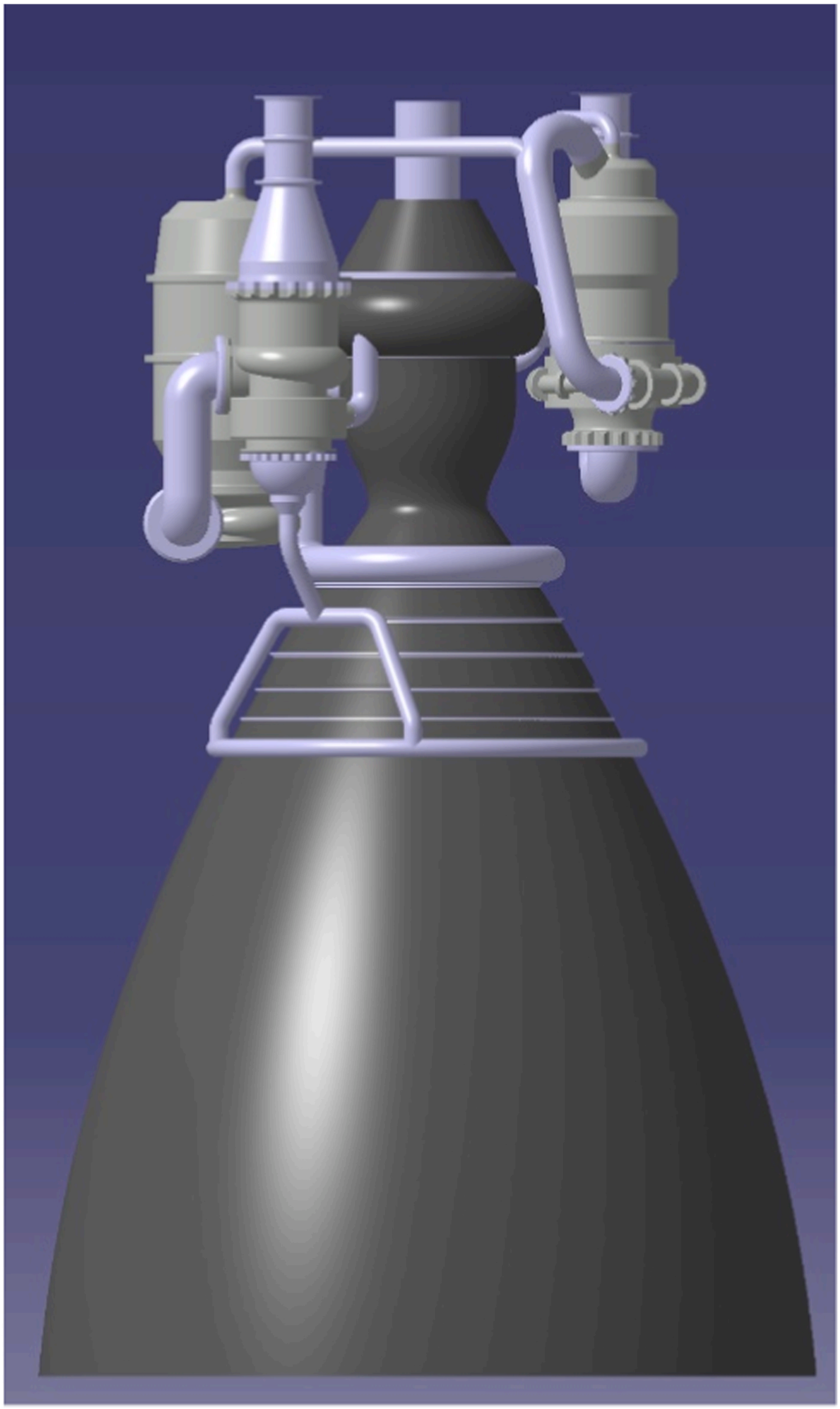

Fig. 2. CAD-drawing of SLME V7 with nozzle expansion ratio 59.

thrust profile of the P160 used in the ascent flight optimization of DLR follows the law as presented in Ref. [16].

This leads to a calculated payload of approximately 22.5 Mg into the reference LEO-mission, a gain of 3 Mg compared to the Block1 Ariane 64 version in that orbit. Assuming the feasibility of an underloading of 7 Mg fuel on the ULPM, a maximum gain of about one additional metric ton seems possible.

3.3.2. Ariane 6 evolution with C130

Another option of future Ariane 6 evolution (not yet decided) could be the replacement of the P120C/P160 SRM with potential new liquid booster "strap-ons" using LOX/LCH4 or LNG with PROMETHEUS engine. Such a launcher concept might operate the side boosters in expendable mode for high-performance missions (as investigated here) or in RLV modes DRL or RTLS in medium/low performance missions described in Ref. [15].

Table 6
SpaceLiner Main Engine (SLME) technical data from numerical cycle analysis [1].

Operation point	O1	O1	O2	O2	O3	O3
Mixture ratio [−]	6		6.5		5.5	
Chamber pressure [MPa]	16		16.95		15.1	
Mass flow rate in MCC [kg/s]	513.5		555		477.65	
Expansion ratio [−]	**33**	**59**	**33**	**59**	**33**	**59**
Specific impulse in vacuum [s]	436.9	448.95	433.39	445.97	439	450.56
Specific impulse at sea level [s]	385.9	357.77	386.13	361.5	384.2	352.6
Thrust in vacuum per engine [kN]	2200	2260.68	2358.8	2427.28	2056.7	2110.49
Thrust at sea level per engine [kN]	1943	1801.55	2101.6	1967.32	1800	1651.56

Table 7
PROTEIN-study sea-level staged-combustion engines calculated technical data.

Fuel type	LCH4	LH2
Nozzle area ratio [−]	36	
Chamber pressure [MPa]	20	
Mass flow engine [kg/s]	642.6	516.7
Thrust at sea level engine [kN]	1976	1972
Thrust in vacuum engine [kN]	2200	
Specific impulse at sea level [s]	313.4	389
Specific impulse in vacuum [s]	349	434

Table 8
PROTEIN-study vacuum staged-combustion engines calculated technical data.

Fuel type	LCH4	LH2
Nozzle area ratio [−]	120	
Chamber pressure [MPa]	16	
Mass flow engine [kg/s]	610.2	488.6
Thrust in vacuum engine [kN]	2200	
Specific impulse in vacuum [s]	367.5	459

The architecture intends to keep the existing liquid propulsion main stages LLPM and the ULPM including the interstage structure untouched as far as possible. The P120C (or P160) side booster of the Ariane 6, however, are replaced by new liquid boosters. This requires the same axial position for the booster attachment points at the H150's intertank and aft-skirt structure as for the Ariane 6 (see Fig. 3 at right). The feasibility assessment presented in Ref. [15] targets the maximum LOX/LCH4 propellant loading under tight geometry constraints. The liquid booster diameter is slightly increased from 3.4 m of P120C to 3.6 m instead. Both propellants are stored aft of the forward attachment ring in an integral tank with common bulkhead. Three PROMETHEUS engines are placed in linear arrangement in the booster's base. This design enables to put one of the engines in a center position simplifying the vertical landing with single operational engine in case of RLV-mode. The approximate nozzle exit diameter of 1.1 m is still compatible with this placement and the booster diameter [15].

The preliminary stage architecture shows that each booster can carry up to 130 Mg of propellant in total (C-130) and stage lift-off mass is estimated at less than 150 Mg.

For the envisaged heavy LEO-mission, the C130 boosters are operated in expendable mode, that is without recovery in order to maximize the performance. This configuration would enable a performance of above 23.9 Mg into the considered orbit, 1.4 Mg more than the "Block 2" variant. This is possible despite the significantly lower GLOW due to the significantly better specific impulse of the LCH4-engines compared to solid propellants.

3.3.3. Ariane 6 derived with closed-cycle engine on LLPM

Another potential A6-evolution could be the replacement of the Vulcain 2.1 engine on the central core by a high-thrust, better performing liquid engine. The installation of a LOX-LH2-staged combustion engine as the main propulsion system on LLPM could be straight forward. A suitable engine under investigation by DLR is the SLME which is required to be operated at mixture ratios between 5.5 and 6.5 [1] and Table 6.

Operating the SLME at a mixture ratio of about 6 allows maintaining the existing tank setup of LLPM since the Vulcain 2.1 operates at MR of nearly 6 as well. Feed system and thrust structure will have to be adapted on A6 to the higher mass flow of SLME. This impact on stage dry mass was not investigated here, but the increased mass of the SLME was considered.

Implementing the SLME on the Ariane 64 "Block 2" would boost the performance to a staggering 27.4 Mg that is a substantial gain of 4.8 metric tons. A reduced mixture ratio of 5.5 would offer slightly better specific impulse but would imply a smaller LLPM propellant loading and a moderate stage re-design. The obtained performance of a lower mixture ratio is marginal at best, not justifying the re-design effort of the tank system.

A delayed ignition in flight of the SLME at approximately 51 s after lift-off in an altitude of 9.6 km allows a remarkable increase of the performance to 29.9 Mg. The ground impact point of an the LLPM would be shifted slightly to the East but still very far from any coastal areas. Obviously, the boosters need the capability to provide full attitude control during the initial ascent with an inactive core propulsion. Additionally, it might prove necessary to re-assess the structural strength of the booster attachments and possibly of the core stage itself. Due to the significant thrust increase by replacing the Vulcain 2.1 engine by the SLME, the loads on the structure increase as well compared to the "Block 2", both in terms of dynamic pressure (~20 %) and axial load factor (~10 %). The potential requirement of strengthening the involved structures could slightly reduce the calculated payload gain. Another point to be considered is related to safety. Igniting the main engine on ground allows performing safety checks before solid booster ignition. This is particularly relevant in human spaceflight missions not regarded here. Such safety checks are omitted when igniting the SLME in flight.

3.3.4. Ariane 6 Derivations synthesis

Fig. 4 gives a depiction of the optimized ascent trajectories for the Ariane 6 evolution options in the heavy-lift LEO-mission. Tracks show some coherence with the achieved payload performance (Table 9). While the A64 "Block 2" and A6 Evolution with C130 pursue a very similar trajectory with the latter being slightly more performant during booster ascent, the trajectories using the SLME show a significant increase in acceleration, most noticeably during the flight of the LLPM after booster separation, increasing MECO-condition by several 100 m/s while separation altitude drops (Table 10 and Fig. 4).

The calculated gross payload masses in Table 9 include also the mass of the payload adapter. Hence, the net performance might vary depending on the actual adapter mass to be foreseen and could be at least several hundred kg lower. The small reduction of lift-off mass for the Ariane 6 Evolution with an air start of the LLPM is due to the fact that the solid boosters will consume slightly more fuel until the thrust is

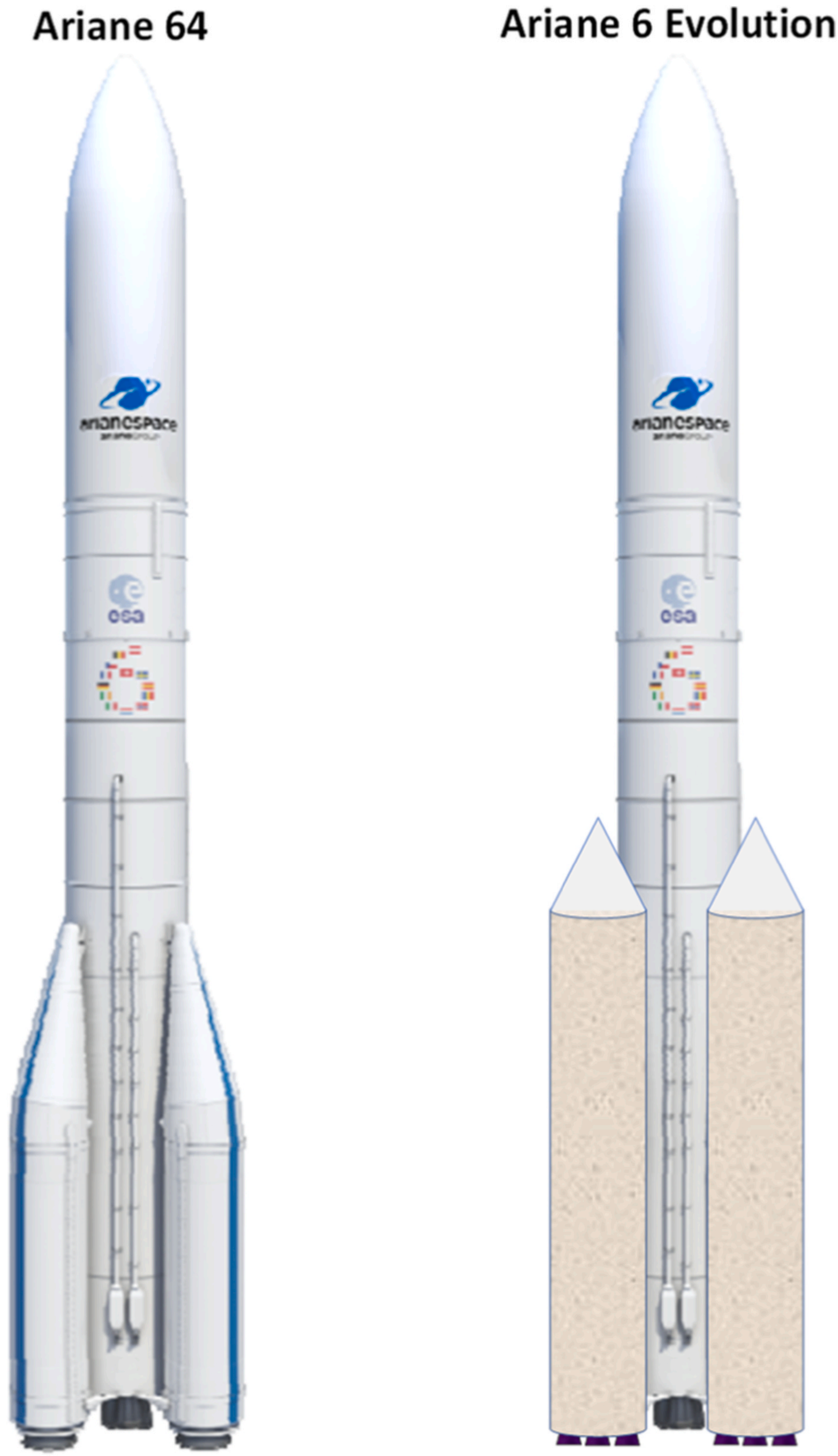

Fig. 3. Sketch of Ariane 6 Evolution with four liquid strap-on boosters compared to A64 (left) [15].

sufficient for release lacking the thrust of the central stage in this moment.

The version with inflight ignition of the SLME is notably different from the other trajectories and shows a more evenly and consistent altitude-velocity increase during the LLPM flight whereas the other variants perform a steeper ascent but then achieve lower MECO-velocity. The higher the initial velocity of ULPM, the lower its Δ-v-requirement and hence the more payload mass can be lifted to orbit. For higher separation velocities the ULPM net gravity losses slightly increase while thrust vector losses will decrease significantly (Table 10).

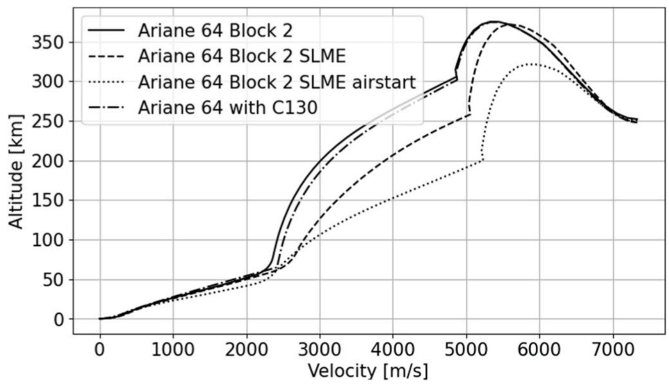

Fig. 4. Ascent trajectories for Ariane 6 Evolution options in reference LEO.

Table 9
Payload and total lift-off mass of options for Ariane 64 Block 2 and Ariane 6 Evolution.

	Block 2	C130 booster	SLME on LLPM	SLME-LLPM, air start
Lift-off mass (without payload)	916 Mg	806 Mg	917.6 Mg	917.4 Mg
Gross payload 250 km × 300 km x 25°	22.57 Mg	23.98 Mg	27.4 Mg	29.9 Mg

Table 10
Separation conditions of LLPM and calculated accumulated losses during ULPM flight.

	Block 2	C130 booster	SLME on LLPM	SLME-LLPM, air start
Separation velocity	4869.5 m/s	4875.3 m/s	5062.6 m/s	5233.1 m/s
Separation altitude	306.2 km	303.5 km	256.5 km	199.9 km
ULPM gravity loss	−66.6 m/s	−62.5 m/s	4.8 m/s	85.6 m/s
ULPM thrust vector loss	524.1 m/s	524.0 m/s	362.0 m/s	320.4 m/s

3.4. Partially reusable VTHL: RLVC-4

Investigations of semi-reusable heavy launchers with the internal project name RLV-C4 [15,17,18] have been carried-out by systematic variation of design options on propellant choice or aerodynamic configuration. One concept with the SLME as the main engine has served as RLV-reference in the FALCon-project [19] and its architecture has some similarities to the SLB8V3 (section 3.5), however, with significantly reduced propellant loading (380 Mg) and only four SLME [17,18].

The RLV-C4 could form Europe's first step to reusable space transportation with a payload performance equivalent or even in excess of an expendable Ariane 6 evolution as described in the previous section 3.3. 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 [15,17]. Beyond the winged VTHL-concepts in focus of this section, similar VTVL options in architecture, size and payload performance have been studied as a potential alternative [15] and might be considered in future work. If smaller gas-generator type engines in the PROMETHEUS-class are implemented instead of the SLME, convergent designs have been found. However, in case of methane fuel the GLOW and hence the number of engines will dramatically grow [18].

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. A 14 tons GTO-class with multiple payload capability can be achieved by a 3-stage architecture while still remaining at relatively compact size [15,18].

The TSTO-concept with large expendable 2nd stage (Fig. 5, right) was initially defined as an H150, even more compact than the core stage of the classical Ariane 5G. With the heavy-lift LEO mission in mind, the expendable upper stage's propellant loading has been optimized keeping the choice of single SLME untouched. The reusable RLVC-4 stage remains also unchanged.

A small increase in propellant loading to 160 Mg delivers the optimum performance with roughly 27.9 Mg separated payload mass. Fig. 6 shows the ascent profile of this launcher with RLV-stage separation occurring at moderate speed, below 2 km/s. The 2nd stage would grow slightly in length compared to what is shown in Fig. 5. The disadvantage of bringing this stage into LEO is the requirement of its controlled deorbitation consuming a significant amount of fuel. Therefore, the interest of using a 3STO instead has also been studied for the same mission which allows the large cryogenic stage to remain suborbital with its splashdown occurring in the Pacific.

A hypothetical storable kick-stage has been defined for raising the orbit from second stage MECO of 30 km × 250 km–250 km × 300 km. A separated payload of around 29350 kg could be reached, an improvement of approximately 1445 kg compared to the TSTO. If the final destination of the mission is similar to the reference LEO a fully cryogenic 3STO with H14 as 3rd stage (as visible in Fig. 5 at left) is of limited interest being too heavy and too expensive. However, in case of more demanding missions the picture changes and such a configuration could become highly attractive for e.g. translunar injection.

3.5. Partially reusable VTHL: RLVC-5 with SpaceLiner SLB8

A semi-reusable launcher based on the SpaceLiner 8 booster design [20–22] and a side-mounted large expendable upper stage has been defined under the designation RLVC-5 (Fig. 7). The configuration's architecture is quite similar to the RLVC-4 TSTO (section 3.4) but sized as a significantly larger winged RLV-stage with 10 SLME (see section 3.2.2) instead of merely 4.

The principal architecture of the expendable stage is even more similar to the RLVC-4 TSTO's second stage with LOX-LH2 stored in a common bulkhead tank and powered by a single SLME in large expansion ratio variant. Faring and stage diameter have been increased to 6.5 m. A huge 24.2 m long fairing, that provides 700 m^3 of internal volume, is assumed for the super-heavy lift transport with its mass conservatively estimated at 6400 kg.

At lift-off the ten engines on the RLV-booster stage SLB8 are ignited and accelerate to stage separation at high altitudes at the edge of the atmosphere. This maneuver could be relatively relaxed in terms of timing and allow for some delay in the ignition of the upper stage if necessary or advantageous. While the SLB8 is kept in the configuration for its primary application of SpaceLiner, the expendable stage's propellant loading has been varied to find the maximum achievable payload mass.

The investigations reveal that the maximum payload to the reference LEO is found at 80 Mg with an upper stage ascent propellant of approximately 160 Mg. Thus, the expendable part of this heavy launcher remains relatively compact in size (length 18.8 m without faring) as visible in Fig. 7. Achieving maximum performance, would require some off-loading on the RLV-stage to realize adequate initial acceleration levels. Though this choice does not result in the optimum launcher for this particular application, the approach makes sense if the SLB8 designed for the SpaceLiner missions (e.g. Ref. [20]) is used for secondary tasks and thus demonstrates its operational robustness.

In Fig. 8 the RLVC-5's direct injection into the 250 km × 300 km

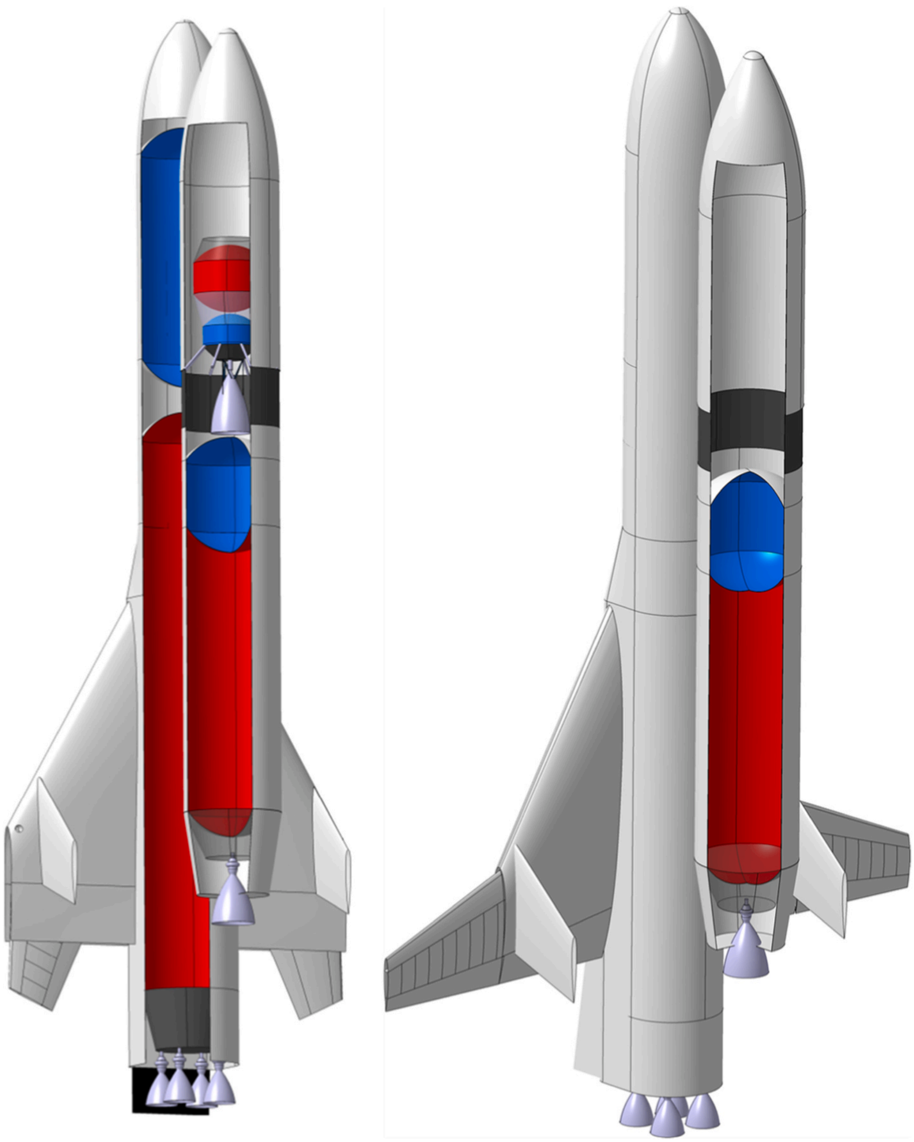

Fig. 5. Launcher architecture sketches of RLVC-4-B configuration as 3STO (left), TSTO (right) [18].

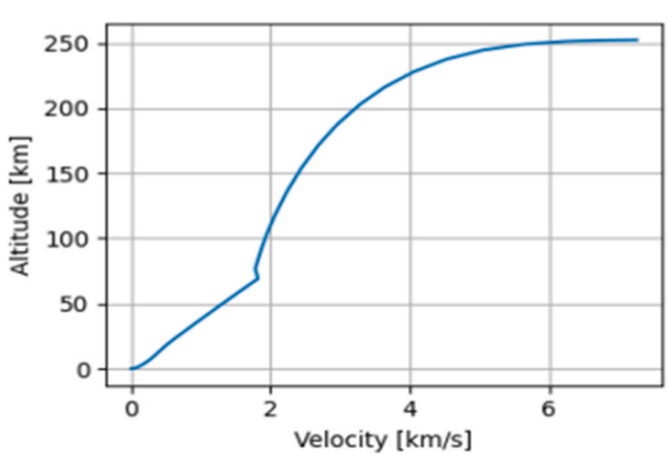

Fig. 6. RLVC-4 ascent with expendable upper stage H160 in reference LEO.

reference orbit is shown. The RLVC-5 stage separation occurs at roughly twice the velocity of the much smaller RLVC-4. Thus, the expendable second stage of similar size can accelerate a considerably higher payload mass.

Return to the launch site of the SLB is assumed to make use of the patented "in-air-capturing"-method which likely provides the best performance [19]. The study for the next SpaceLiner 8 booster design is ongoing, however, a consolidated configuration is not yet defined. Nevertheless, the SLB8V3 as summarized above, serves in this paper as the large, reusable booster stage of the RLVC-5 future European heavy-lift launcher.

3.6. Reusable VTHL TSTO SpaceLiner

The SpaceLiner TSTO, while initially investigated as a point-to-point passenger transport system, has also been defined as a fully reusable space transportation system to LEO with payload performance in the A6-class. The parallel arrangement of the two SpaceLiner stages of variant 7, the reusable booster and the orbiter or passenger stage, at lift-off and its main dimensions are presented in Ref. [20]. The version 7 is powered by 9 + 2 SLME (see section 3.2.2) at lift-off.

The SpaceLiner7 passenger stage's internal design has been adapted for its secondary role as an unmanned satellite launcher. The passenger cabin is not needed for this variant and is instead replaced by a large internal payload bay [23] as shown in Fig. 9. Key geometrical

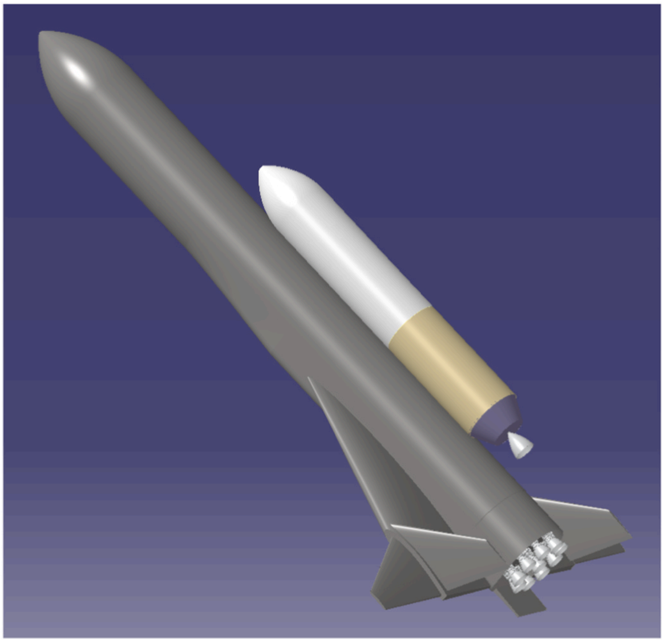

Fig. 7. RLVC-5 as CAD geometry.

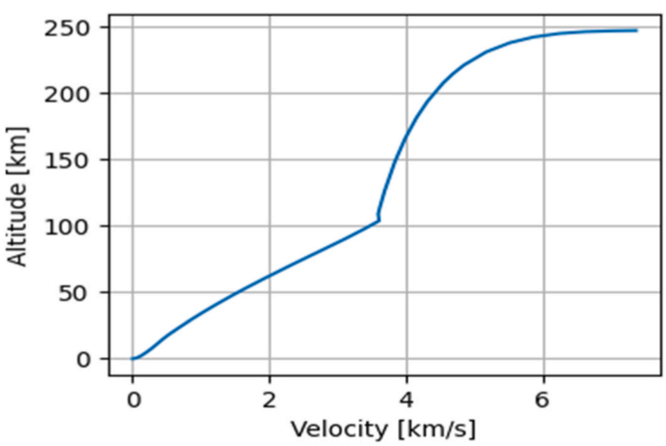

Fig. 8. RLVC-5 ascent with expendable upper stage H160 in reference LEO.

constraints and requirements are set so that the SpaceLiner 7 passenger stage's outer mold line and aerodynamic configuration including all flaps are be kept unchanged. The internal arrangement of the vehicle could be adapted; however, maximum commonality of internal components (e.g. structure, tanks, gear position, propulsion and feed system) to the passenger version is preferred in order to realize development cost savings.

Further, the payload bay should provide sufficient volume for the accommodation of a large satellite and – if required - its orbital transfer stage. For this purpose, the SLO's propellant loading has been reduced by 24 Mg–190 Mg compared to SLP with a smaller LOX-tank to allow for a payload bay length of 12.1 m and at least 4.75 m diameter [23]. These dimensions are close to the Space Shuttle (18.3 m × 5.18 m x 3.96 m) and should accommodate even super-heavy GTO satellites of more than 8 m in length and their respective storable upper stage [23]. Large doors open on the upper side to enable easy and fast release of the satellite payload in orbit.

The SpaceLiner 7-3's GLOW as TSTO without payload is at 1783 Mg (Table 11) and reaches including the payload between 1800 and 1810 Mg. These values are below those of the partially reusable Space Shuttle STS of more than 2000 Mg while the SpaceLiner delivers higher payload. This better launch efficiency results out of the fully cryogenic system and the SLO being unmanned, saving the crew cabin and life-support systems of the Space Shuttle.

Launch of the SpaceLiner 7 TSTO orbital launcher has been simulated from the Kourou space center for various missions. In case of satellites transported to GTO, the injection of SLO occurs into a low 30 km × 250 km transfer orbit allowing the reusable orbiter stage becoming a once-around-Earth-vehicle capable of reaching its own launch site after a single circle around the planet. Subsequently, an orbital transfer is necessary from LEO to GTO using an expendable upper stage with storable propellants [20].

The SpaceLiner 7 TSTO has been newly assessed for the reference LEO-Mission using a similar bi-boost strategy of the SLO as previously applied to the ISS-mission [23]. The initial ascent goes into a 70 km × 300 km LEO before the perigee is raised to 250 km by a second SLME burn. Fig. 10 presents the initial phase of the orbital ascent profile.

Table 11
Mass data of SpaceLiner 7-3 TSTO fully reusable launch configuration.

GLOW without payload [Mg]	1783
Payload SLO 250 km × 300 km, 25° [Mg]	19.85
Payload by kick stage 250 km × 300 km, 25° [Mg]	24.25

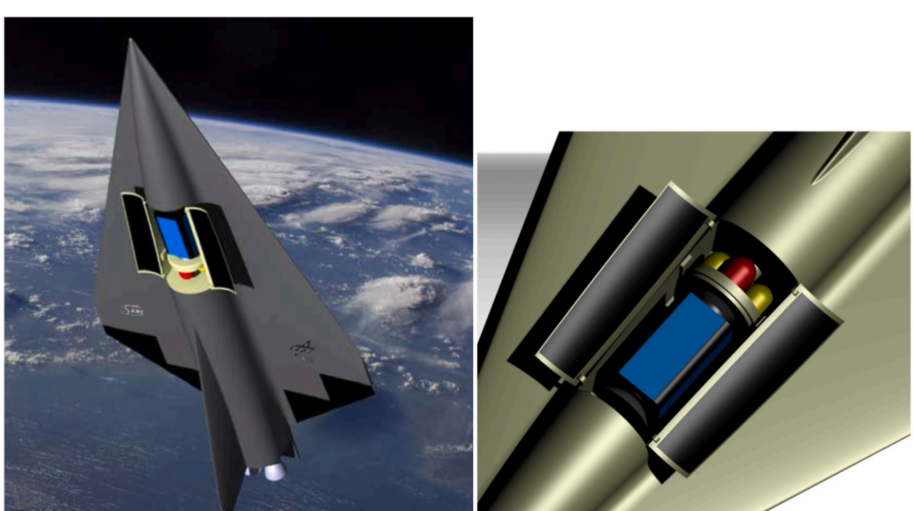

Fig. 9. SpaceLiner 7 orbital stage (SLO) in CAD renderings with open cargo bay and payload with kick-stage.

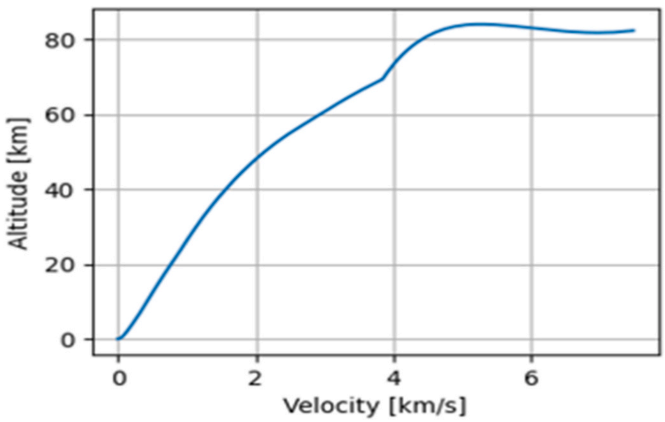

Fig. 10. Calculated ascent trajectory of SpaceLiner 7-3 TSTO in LEO mission plotted up to SLO's 1st MECO.

Almost 20 tons separated payload could be delivered if the reusable upper stage itself is orbited (Table 11). The relatively large dry weight of the SLO makes it attractive, even in case of the LEO-mission, to consider keeping the orbiter in suborbital conditions and attaching a smaller expendable kick-stage for in-orbit injection (Δv≈ 66 m/s) of the payload. As a consequence, the achievable payload mass increases to more than 24 tons (Table 11).

The SpaceLiner Orbiter reentry has been simulated for both injection options. The aerodynamic trimming of the satellite transport stage with the trailing edge flaps and its bodyflap identical to the passenger variant has been preliminarily checked in numerical simulation under hypersonic flow conditions of atmospheric reentry and is found feasible within the constraints of the 7-3 lay-out [23]. In case of SLO injection one full orbit is to be performed before deorbiting. Reaching the destination CSG in Kourou is without problem for the orbiter, either from stable LEO or the simpler once-around mission due to its still very good hypersonic L/D. The vehicle crosses Central America at high altitude and turns to the South over the Caribbean Sea reaching CSG from the Atlantic. The maximum heatloads remain slightly lower than for the reference SL7-passenger concept because of a different AoA-profile and lower vehicle mass. The preliminary assumption of a common TPS with the passenger stage is once again confirmed by the new reentry simulations for LEO.

3.7. Fully reusable VTVL: PROTEIN-related study

The ESA-initiated PROTEIN-study called for the creation of a concept capable of carrying up to 10000 tons of LEO-cargo per year using a new European Heavy Lift Launcher (EHLL) not exceeding a recurring payload cost target of 280 €/kg [24]. Both targets are independently assessed by the analyses described in this paper.

DLR joined Rocket Factory Augsburg (RFA) in PROTEIN for the definition of a fully reusable launch vehicle optimized for missions to LEO coupled with in-space transportation to final destinations [25]. However, this paper *does not* summarize results out of the ESA-funded PROTEIN-study but additional configurations investigated later by DLR under similar but not necessarily identical boundary conditions.

In order to reach the ambitious recurring cost-objective, the EHLL is expected to be a fully reusable TSTO-vehicle much larger than any other heavy launcher currently envisaged in Europe. The basic launcher architecture is inspired by the SpaceX Starship & SuperHeavy as VTVL TSTO with stages in tandem arrangement. Two variants have been considered and preliminarily sized in iterations: one based purely on methane fuel and the other storing methane in the 1st stage while switching to hydrogen fuel in the 2nd stage. The main propulsion system is based on closed cycle engines with chamber pressures up to 20 MPa as listed in section 3.2.3.

Fig. 11 shows CAD-based sketches with LOX-tanks in blue, LH2-tank in red, and LCH4-tanks in green. Note, aerodynamic characteristics are reused from DLR's 2022 analyses of Starship [4], however, any dedicated assessment and flyability evaluation has not been performed for the PROTEIN-related types. Therefore, these devices are also not shown while a mass contingency is considered. It is interesting to see in Fig. 11 that the "hybrid" configuration with both fuels methane and hydrogen becomes significantly more compact than the pure LOX-LCH4-launcher. The hybrid configuration is about 98 m in length while the purely methane fueled configuration reaches about 108 m with assumed massive 1st stage diameters of 11 m.

Fig. 12 gives an impression of possible engine arrangements in the base areas of the stages. As is already well-known from the SpaceX SuperHeavy, a purely methane-fuel reusable TSTO-launcher requires an extremely dense packaging of the engines on the first stage. In total 35 methane staged-combustion engines would be needed instead of merely 24 of the same engines on the first stage of the hybrid configuration. A skirt had to be added to accommodate all 35 engines in the base (compare Fig. 11 at bottom right). Engines colored in red have a dedicated role for vertical landing or hovering. Actual feasibility of the engine arrangement as presented in Fig. 12 under TVC requirements is not investigated and thus could require a further increase of the base diameter of the methane type.

All launcher configurations have been sized by trajectory optimizations for ascent and descent or return of the reusable stages. The launchers were all predesigned for high but not necessarily identical payload performance. Note in Fig. 13 left, the remarkably lower separation velocity of the hybrid launcher compared to the pure methane variant in case of first stage return to the launch site. It is advantageous to shift a larger portion of total Δ-v to the more efficient LOX-LH2-2nd stage.

Despite the more compact layout, the calculated performance of the hybrid launcher with hydrogen in the upper stage has a significantly better payload performance (Table 12). This was already the case for the PROTEIN-related design mission of 450 km and inclination of 6° with +115 % in case of RTLS [26]. The mission with higher reference inclination of 25° in combination with the RTLS maneuver shows similar improvement.

The rather moderate performance of a fully reusable methane-based super heavy TSTO with GLOW of more than 4200 Mg is on the one hand due to the limited performance of this propellant combination. Further, the staging conditions turn-out to be not optimal and a shift of propellants from the first to the second stage would somehow improve the situation. A systematic assessment by DLR with variation of generic assumptions for fully reusable TSTO gives directions for future designs [27].

4. Cost estimation and comparison

4.1. Comparing dry mass fractions

The size and hence the lift-off weight of the different investigated launchers spans a substantial range, making any direct comparison difficult. The reentry and return methods of the reusable stages are quite diverse. Depending on the size of the stages also the dry masses show massive differences.

However, the stage dry masses can be associated to functional classes, either in ELV or in RLV. Such functional classes are directly used in the cost estimation process of the following section. The simplest distribution is defined for ELV with only two classes.

- (integral) tanks and primary structures, subsystems
- main engine propulsion

In case of RLV the classes become more complex because additional hardware is required allowing the stages to be recovered after a mission.

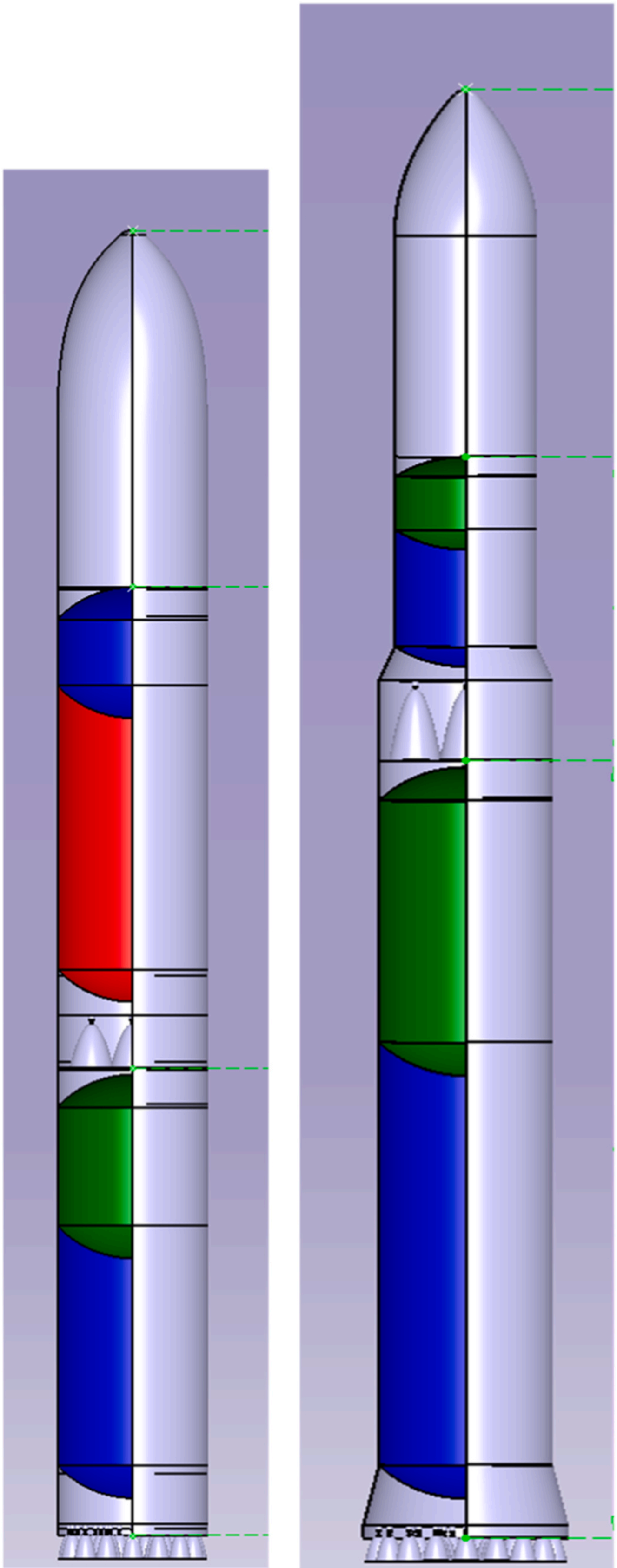

Fig. 11. Architecture definition of EHLL "hybrid" (H_2 and CH_4) type (left) and only methane type (right) [26].

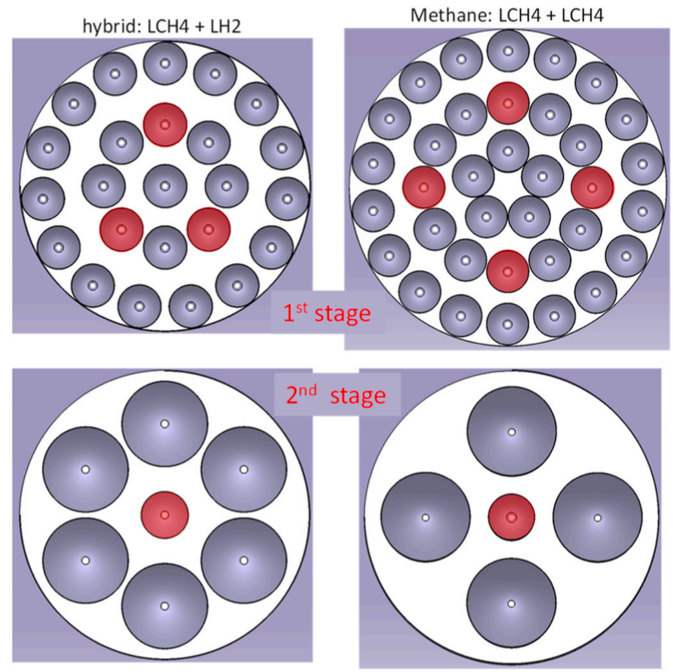

Fig. 12. Potential engine arrangement of EHLL hybrid type (left) and methane type (right) [26].

The following four are used.

- (integral) tanks and primary structures, subsystems
- main engine propulsion
- aerothermodynamic devices (wings, fins, actuators, TPS)
- landing devices

Fig. 14 shows a relative comparison of the different dry mass fractions by functional classes. In case of the first stages (top) the ELVs' structural fraction reaches 80 %–90 % with around 10 % for the main propulsion system. The RLV-stages show a significant shift: only about 50 % remain for conventional rocket structural components and between 20 % and 30 % for dedicated hardware to support reusability. Further, a notable difference between VTHL and VTVL is reported. The horizontal landing requires wings of significant size and weight which are to be protected by TPS resulting in a relatively large fraction of at least 25 %. The vertical landing 1st stages have merely a minor fraction for these components but require instead an increased fractional (and absolute mass) for propulsion. As the reusable VTVL-stages also need wings and TPS for their high-speed reentry flight, the difference to VTHL mass fractions shrinks.

4.2. Cost estimation

While cost is considered one of the most important metrics in comparing space transportation systems, the estimation of development and recurring costs remains challenging. This is especially true for the early concept phase, where only limited data is available or transportation scenarios of the future are uncertain [28]. For a broad comparison of the European heavy-lift launchers options presented herein, the parametric cost model described in Ref. [29] is employed. It is a parametric cost model based on the widely used TRANSCOST [30] relations, but augmented with additional data points, where possible, and additional component categories for reentry and landing hardware as well as GNC development effort.

Where applicable, the reusable second stages are treated the same as reusable first stage, including the refurbishment factor and the number of reuses. Thus, reusable second stages considered in this study are

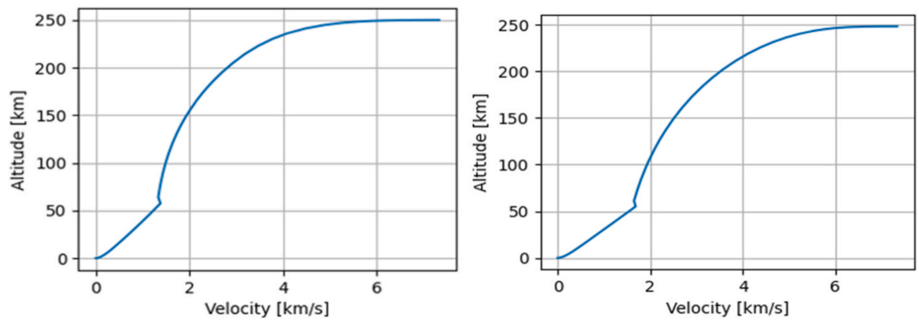

Fig. 13. Calculated ascent in 250 km × 300 km reference LEO considering RTLS with "hybrid" (H_2 and CH_4) type (left) and only methane type (right).

Table 12
PROTEIN-related EHLL calculated payload performances in reference LEO for 1st stage RTLS return mode.

"hybrid" CH4+ H2	100 Mg
purely CH4	47 Mg

assumed to achieve 50 reuses, consistent with the assumptions for winged first stages. Rocket engines on horizontally landing stages are reused 25 times, while engines on vertically landing stages are reused 20 times. The reusable second stages specific costs per kg are higher, as they reach increased reentry component masses due to the more challenging flight environments. Kick stages are modeled as "propulsion modules" (see Ref. [30]) with regard to non-recurring cost and as liquid stages and separate propulsion system with regard to the recurring costs.

The following simplifications are applied to the cost estimations: The investments in new, modified or extended ground and production

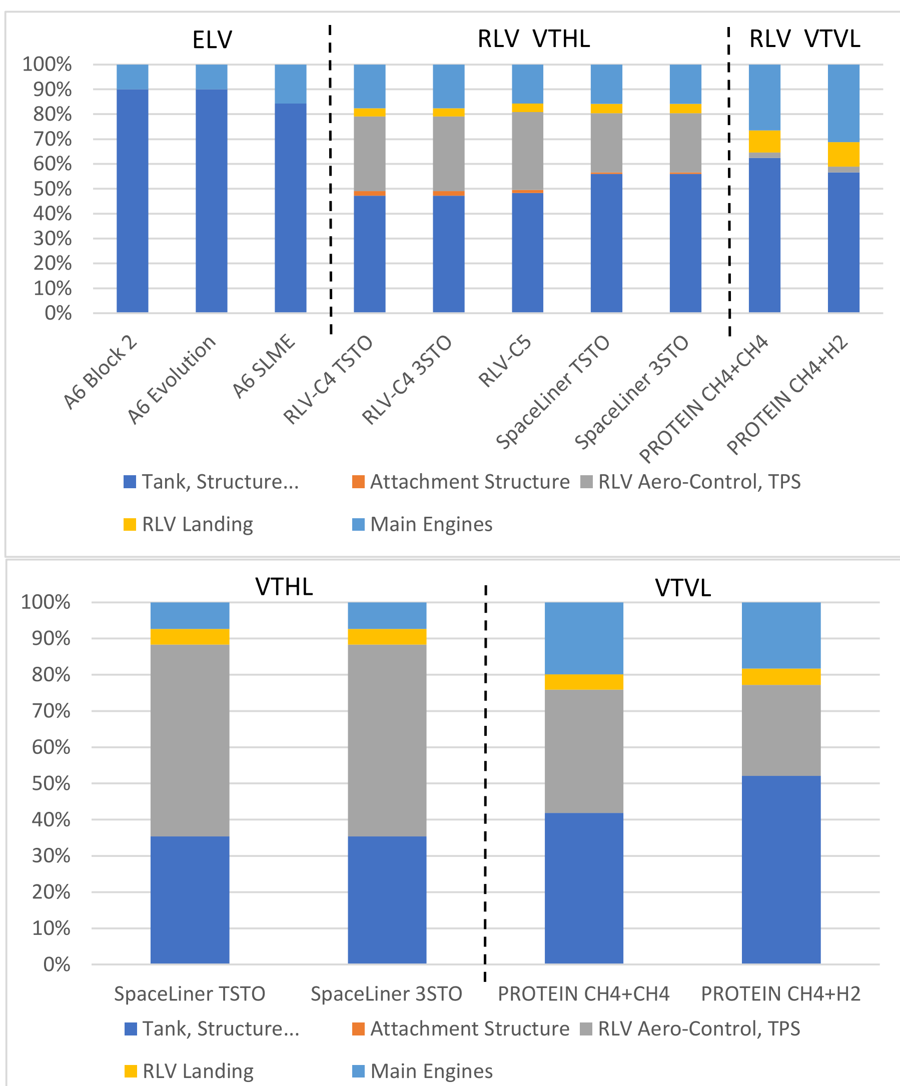

Fig. 14. Relative dry mass fractions of main components (1st stages top, reusable 2nd stages bottom).

infrastructure are not calculated. Such an assessment would be highly complex for the diverse launcher configurations and in most cases being based on extrapolations well beyond any European experience in the field. Further, estimated recurring costs are always assuming full operation in each year. This is neglecting the uncertainties of a ramp-up phase which might be different for the regarded configurations.

4.2.1. Limitations

The cost model employed herein reduces a complex technical and organizational question to a handful of parametric equations to estimate the cost of space transportation systems. In case of ELV some validation of the estimated costs is possible although access to actual cost data is restricted in many cases. The fact that most of the assessed reusable systems have never been implemented in Europe (or anywhere in the world) adds additional uncertainty. With these caveats in mind, the absolute results should be interpreted with care. The goal here is a relative comparison of the various options. Comparability between different vehicle types is achieved by separating the main stage into different categories, as described in section 4.1, so that structural masses of lower specific cost are not lumped together with more expensive categories (per kg) such as propulsion or the dedicated reentry hardware.

The estimated costs do allow at least a relative comparison of the various concepts. While small relative differences should not be over-interpreted as fundamental, the general trends are deemed sufficiently robust.

4.2.2. Comparison of results

Fig. 15 shows the estimated development cost (without the launch site infrastructure) of the various (Super) Heavy Lift options. The estimation includes also the Ariane 6 with all its major components although this European launcher is now already operational in its Block 1 configuration. Part of the development costs could disappear since components already exist. However, as will become clear below, this hardly influences the final comparison.

In general, the development costs are of a significant, albeit realistic order of magnitude. Actual Ariane 6 Block 1 development costs including the P120C solid boosters were less than 10 B€, however, two liquid rocket engines (Vulcain 2 and Vinci) had already been developed previously. The estimated costs in Fig. 15 instead include all major components. Further taking into account today's economic conditions, the presented estimation for an Ariane 6 hypothetically to be developed with all elements in 2025 is not unrealistic. The increased NRC of A6 Evolution is due to additional main propulsion development for a methane engine and more complex and hence costly liquid boosters compared to solid ones. The Planetary Society estimates the cost of the Space Shuttle development to have been 50 B$ (2020 e.c.) [31]. While the Space Shuttle was a manned system, it was actually not fully reuseable, so it appears reasonable that the fully reuseable unmanned systems with increased payload capacity assessed herein exhibit a similar development cost.

All A6-derived expendable launchers show development efforts below 20 B€ while the fully reusable TSTO of significantly larger scale reach out beyond 40 B€. The partially reusable launchers are located between these boundaries. Although the development expenses of the A6-type are less than a third of the heavy full-RLV-concepts, this turns out to be rather insignificant when the launch costs along operational lifetime are regarded.

The recurring costs for the various concepts have been assessed for launching a fixed total payload mass to the reference orbit each year over the span of 20 years. The annual delivered payload mass has been varied from 1000 Mg to 10000 Mg to LEO. As ramp-up phases are not considered, the yearly payload requirement is applied immediately and uniformly. With the parametric cost models used herein, the total recurring cost depends on the total number of launches, not their temporal distribution. Effects that a ramp-up phase or non-uniform launch schedule might have on the business-case are relevant for more detailed future studies but are not considered in the current model. The number of necessary launches per year for each option depends on the annual payload mass as well as the launcher payload performance and ranges from about 10 (1000 t-scenario served by most powerful 100 t launcher) to more than 500 (10000 t-scenario with A6-Block 2). The latter is ridicule-high with almost two launches per day. Although calculated, the resulting excessive total costs above 400 B€ for the medium size ELVs are not completely shown in Fig. 16. Any such investment is unrealistic, the more since there are more affordable options.

The lifetime cost including NRC and RC over 20 years are shown for the different payload market scenarios in Fig. 16. The numbers are pretty high but as these are stretched over at least 30 years including the development phase, the yearly investment is not out of reach. The larger the annual mass for orbit, the larger the saving by introducing reusability at least to the first stage and the benefit of really super heavy size vehicles.

The associated specific recurring launch costs of the different concepts are shown in Fig. 17. It can be seen that the Ariane 6 derivatives are always the costliest options and only for the smallest considered payload scenario are they at a similar magnitude to the other concepts. For sure, Ariane 6 was never intended to bring 1000 t per year to orbit. With the investment in ground and manufacturing infrastructure not considered in this comparison, the unavoidable costs of building several new launch pads even for Ariane and exploding current manufacturing size by a factor up to 50 would further deteriorate the relative position of

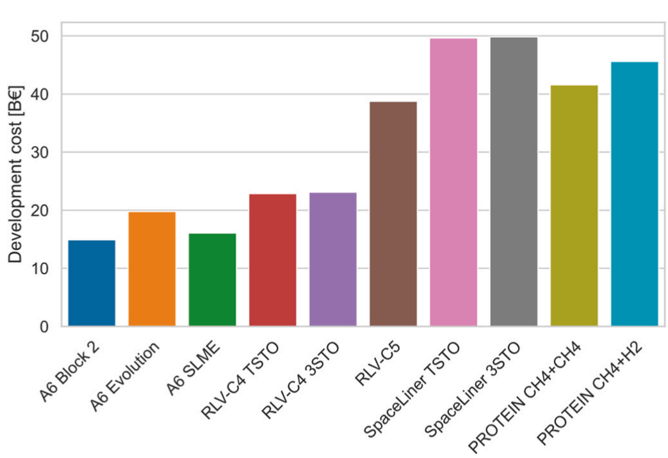

Fig. 15. Estimated development cost of heavy lift launcher concepts (e.c. 2024).

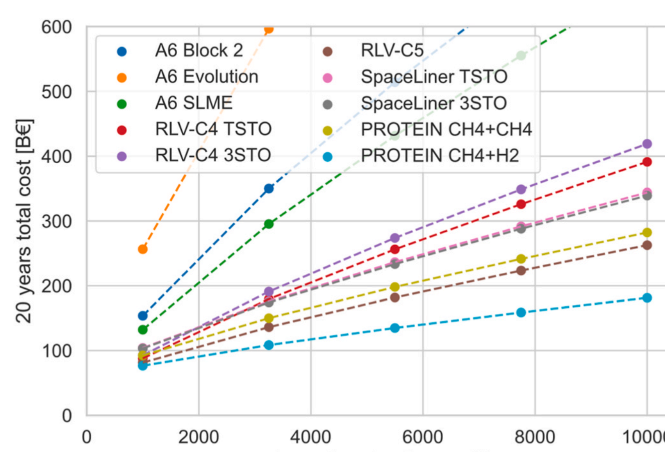

Fig. 16. Total cost for variation of annual payload delivery (e.c. 2024).

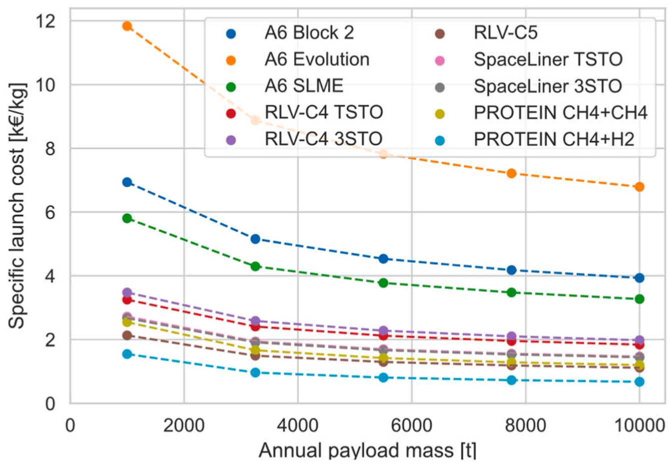

Fig. 17. Specific launch cost for variation of annual payload delivery (e. c. 2024).

the pure ELV.

The RLVC-4 semi reusable concept and the SpaceLiner variants show total cost in a similar range while dramatically improved compared to the ELV. These aforementioned types all have similar payload capacities and though the SpaceLiner variants are much larger and heavier (hence estimated at higher development and production cost per item), their full reusability appears to offset this disadvantage that they reach an edge in total and specific cost for the investigated scenarios.

In Fig. 18 the segment with specific transportation costs at or below 2000 €/kg is visible. The relatively tight spacing of the different cost lines is better resolved. The most cost-efficient design options, especially for the larger payload market scenarios, are provided by the three true Super Heavy Lift Launchers (SHLL): The two PROTEIN-related VTVL's and the RLVC-5 concept. All of them are able to launch at least 50 Mg of payload into the reference LEO in a single flight. While the RLVC-5 concept is semi reusable, the comparatively small cryogenic upper stage, combined with a substantial payload performance, results in relatively low recurring launch costs bringing it in second best place. The PROTEIN-related launcher with methane first stage and hydrogen upper stage achieves with some margin the lowest cost for all payload scenarios. The combination of being fully reusable and also sporting the highest payload performance of all herein considered concepts supports this result.

DLR's assessment identifies some heavy launcher concepts with attractive specific transportation costs approaching 1000 €/kg down to

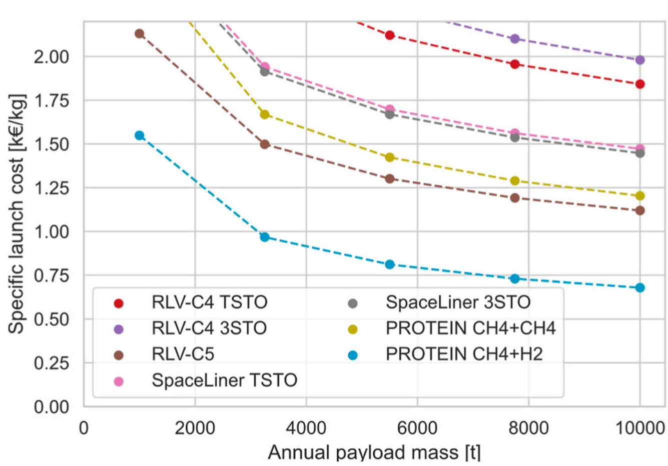

Fig. 18. Specific launch cost for variation of annual payload delivery, Fig. 17 zoomed in RLV segment (e.c. 2024).

roughly 700 €/kg. This does not exactly confirm feasibility of ESA's payload cost target of 280 €/kg [24] but is found at the same order of magnitude. If new design and manufacturing approaches (not reflected in the current cost estimation) would prove in the future to be significantly more efficient, the specific transportation costs could further drop.

5. Indication of development roadmap

The European heavy-lift launcher options previously described, remain to be sorted in a logical way by the time-scale in which they might be realized. The potential roadmap in Fig. 19 puts all the concepts on a time scale for the next 25 years. Obviously, not all of these large configurations should actually be realized, some are better understood as alternative options.

Within the next roughly 10 years only expendable European heavy-lift launchers are to be expected. The A6 Evolution with potentially reusable liquid strap-on boosters [15] can serve the payload class above 20 Mg to LEO only in fully expendable mode. Partially reusable systems might be realized with the cryogenic RLVC-4 and -5 starting from the second half of the 2030s, potentially achieving after 2040 significant payload mass of up to 80 Mg. Following a sober assessment, a European fully reusable TSTO bringing more than 20 tons to LEO is not to be expected before end of the 2040s. Any such vehicle in the class of 100 Mg comparable to Starship & SuperHeavy will realistically require at least another 25 years before becoming operational in Europe.

The answer to the question, which of the launchers presented in Fig. 19 are to be realized, strongly depends on the overall operational scenario of European space transportation. If performance should approach 30 tons relatively soon, a high thrust closed-cycle engine to be integrated in the A6 core stage is attractive. An engine such as the SLME could be matured first in expendable operations before being attached to reusable first stages. Even if the engine might seem oversized for its initial application, the elevated thrust-level in the 2200 kN class will pay off in all future heavy-lift launchers. In case a broad range of missions should be served a semi-reusable option as RLVC-5 carrying heavy-payloads and the same RLV-booster accelerating also the fully reusable upper stage of SL TSTO for missions with lower payload demand could turn-out to be most attractive. Such combined, building-block-like launchers are not yet reflected in the cost estimations of the previous section.

6. Conclusion

Once becoming operational, the two-stage, fully reusable super heavy-lift launcher Starship & SuperHeavy of the US-company SpaceX could bring a fundamental shift to global space transportation. The actual operational payload performance of Starship is unknown. An independent DLR assessment is compared with updated announcements from SpaceX. While the current test-flight variants' performance is limited, the payload capability into 250 km × 300 km LEO of future enlarged Starships with more than 5500 Mg lift-off mass could realistically reach an impressive 100 Mg net payload mass.

The main part of the paper is dedicated to the technical evaluation of European options in serving a roughly similar payload class in single launch to LEO.

A launcher system analysis first looks into Ariane 6 evolution options and explores the technical limits based on the assumption of expendable stages. The goal of 50 tons can't be reached by any of the investigated evolutionary A6 concepts. Nevertheless, LEO-performance could be pushed up to values between 22.5 Mg and more than 29 Mg. The maximum gain is achieved when the LLPM propulsion of Vulcain 2.1 is replaced by a high-thrust staged-combustion engine. It is acknowledged that results on modified existing launchers are to be taken with care because potential constraints on controllability and structure are difficult to assess without deep knowledge of the existing design.

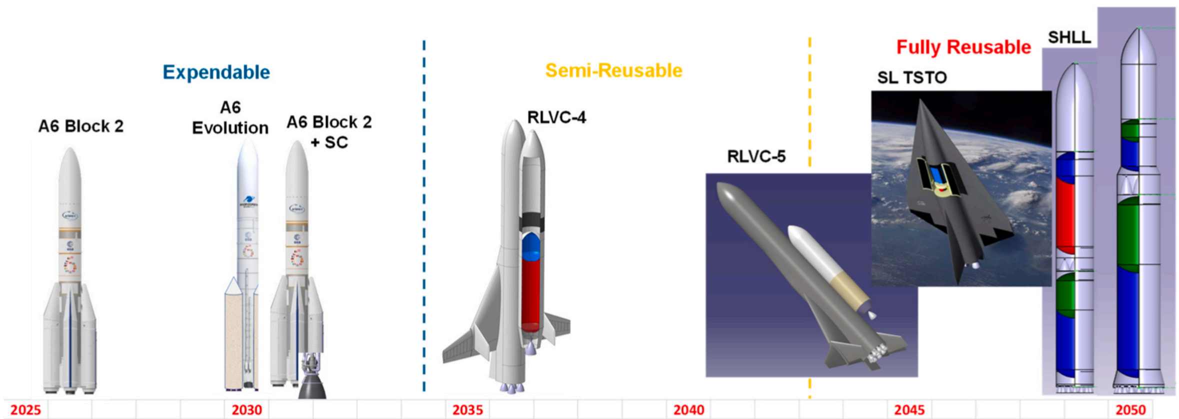

Fig. 19. Potential European roadmap for heavy-lift launchers in the next 25 years.

A significantly better performance perspective can be achieved through a completely new architecture based on building blocks that could start soon the development. In case of the new architecture launchers, all first stages are designed reusable and exclusively liquid cryogenic propellants are chosen. The RLVC-4-concept of DLR might reach similar performance as the A6-derived variants, however, with a major part being reusable. While the fully reusable TSTO SpaceLiner orbiter with internal payload bay based on previous studies is limited to the payload range below 25 tons, using the latest investigated variant of the reusable SpaceLiner booster in combination with an expendable cryogenic upper stage would allow Europe entering the super-heavy class with 80 Mg payload.

Fully reusable configurations have been addressed in the small PROTEIN-study of ESA for which some complementary DLR-concepts are presented. Completely new designs of considerable dimensions will be required if payload mass to LEO should approach or even exceed 100 tons. The combination of a methane lower stage and a hydrogen upper stage is of great interest as the payload delivered would be significantly superior even with a more compact lay-out compared to the pure methane concept.

An extensive launch cost assessment reveals that expendable European launcher modifications or derivatives are not attractive for heavy-lift launch scenarios with annual capacity of 1000 Mg or more. The specific transportation costs of such medium size ELV are four to six times higher than those of dedicated heavy-lift systems. The launch cost difference between fully reusable or semi-reusable heavy systems is limited as long as the reusable part of the space transportation system remains dominant. The launch costs estimations are generated with relationships based on historical launchers or respective design studies. If new design and manufacturing approaches would prove to be significantly more efficient, the specific transportation costs of RLV could drop well below 1000 €/kg.

CRediT authorship contribution statement

Martin Sippel: Writing – review & editing, Supervision, Methodology, Investigation, Data curation, Conceptualization. **Jascha Wilken:** Writing – original draft, Visualization, Methodology, Investigation, Formal analysis, Data curation. **Ingrid Dietlein:** Writing – original draft, Investigation, Formal analysis. **Moritz Herberhold:** Writing – original draft, Investigation, Formal analysis, Data curation. **Kevin Bergmann:** Visualization, Investigation, Formal analysis. **Leonid Bussler:** Investigation, Formal analysis.

Declaration of competing interest

The authors declare that they have no known competing financial interests or personal relationships that could have appeared to influence the work reported in this paper.

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