High-Performance, Partially Reusable Launchers for Europe

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RLV configurations with partial reusability of 1st or booster stages are in focus of ongoing system studies. Parallel stage arrangement is preferred with a winged stage connected to an expendable upper segment with potentially various internal architectures. The non-symmetrical architecture consists of a winged RLV-stage and attached ELV-part comprising either one or two stages. The selected rocket propulsion is mostly cryogenic LOX-LH2 (gas generator or staged-combustion cycle) but also the two fuel combinations methane and hydrogen are looked at. An engine similar to the European PROMETHEUS is also considered for main stage propulsion and architecture implications are discussed.

The paper summarizes major results of the preliminary technical design process. The overall shape and aerodynamic configuration, the propulsion and feed system, the architecture and structural lay-out of the stages are described and different technical solutions are compared. The advanced stage recovery by in-air-capturing is explained and the related European Horizon2020 research project FALCon is introduced.

Keywords: RLV, TSTO, LOX-LH2-propulsion, LOX-LCH4-propulsion, 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
ALM	Additive Layer Manufacturing
AOA	Angle of Attack
ATV	Automated Transfer Vehicle (of Ariane 5)
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
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
CoG	center of gravity
cop	center of pressure

1 INTRODUCTION

System studies of future European RLV configurations with partial reusability of 1st or booster stages are ongoing in DLR. Several tandem launchers for different return and recovery modes, as well as propulsion options have been under investigation. These designed as TSTO for a GTO-reference mission turned out to be feasible, however, reaching significant size of up to 80 m length [1, 2].

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. Reference [3] has 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. This powerful launcher option is in focus of this paper.

Europe's Ariane 6 developments are progressing [4]. Meanwhile, a next generation of a partially reusable heavy launcher is under investigation in several system studies. The CNES' Launcher Directorate is evaluating launch system definitions for the next generation of Ariane launchers, so called *Ariane NEXT* [5]. The current reference at CNES is a configuration in different sub-architectures using LOX-LCH4-propulsion in all its stages. The "toss-back" recovery mode (retro-propulsion and vertical landing) is considered by CNES as a baseline for the reusable first stage [5]. 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 [1, 2].

2 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 upper payload range should be in the 12 to 15 tons GTO-class and should include multiple payload deployment capability. Using an adapted, reduced size upper segment, satellites have to be carried to different LEO. 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. One of the key objectives is to find the most cost-efficient design compared with today's ELV. The choice between 2 or 3 stages is not obvious because on the one hand 3STO become much smaller while on the other hand additional stages add cost. The recent studies of CNES [5] and DLR [3] indicate independently an advantage for recurring cost of TSTO.

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.

2.1 Mission assumptions

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

- GTO: 250 km x 35786 km
- Launch site: CSG, Kourou, French Guiana

The vehicles should be capable of performing secondary missions to LEO, MEO or SSO. The design payload target is between 12000 and 15000 kg to GTO beyond the capability of A64 [1].

2.2 Propulsion systems

Staged combustion cycle rocket engines with a moderate 16 MPa chamber pressure are baseline of the 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) [10]. 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 and 2400 kN, 2000 kN respectively for the second 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. Table 1 gives an overview about major SLME engine operation data as obtained by cycle analyses [10] 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.

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. A size comparison of the two variants and overall arrangement of the engine components is published in [10].

Table 1: SpaceLiner Main Engine (SLME) technical data [10] as used by reusable and expendable main stage

	RLV Booster	2nd ELV stage
Mixture ratio [-]	6.5	5.5
Chamber pressure [MPa]	16.9	15.1
Mass flow per engine [kg/s]	555	481
Expansion ratio [-]	33	59
Specific impulse in vacuum [s]	435	451
Specific impulse at sea level [s]	390	*357*
Thrust in vacuum [kN]	2356	2116
Thrust at sea level [kN]	2111	1678

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 [10].

PROMETHEUS is the precursor of a new European large-scale (100-tons class) liquid rocket engine designed for low-cost, flexibility and reusability [11]. This engine is planned to be operated in open gas generator cycle. For the PROMETHEUS a large effort is paid in the development of Additive Manufacturing (AM) processes for the production of low-cost engine components [11]. Maximum capabilities of existing equipment for ALM have been used for the production of large casings, pump impellers and turbines [11].

Baseline propellant combination of the PROMETHEUS-engine is LOX-LCH4. The choice of a hydrocarbon would allow for a mono-shaft turbopump which is technically less attractive for the LOX-LH2 combination. A reduction in complexity of the turbomachinery should realize further cost savings. Currently, the precursor of PROMETHEUS is under development. An eventually operational engine will have somehow different characteristics which are not yet finally frozen. The calculated data in Table 2 have been generated by DLR to make realistic performance of a full-scale engine available for the launcher system design (compare also [1, 2] for similar gas-generator type but slightly different assumptions on nozzle expansion). A different application might require selection of different engine characteristics.

Table 2: Calculated technical data of PROMETHEUS Methane as used by reusable and expendable main stage

	RLV Booster	2nd ELV stage
Mixture ratio [-]	2.68	2.68
Chamber pressure [MPa]	12	12
Mass flow per engine [kg/s]	422.5	422.5
Expansion ratio [-]	20	59
Specific impulse in vacuum [s]	319	337
Specific impulse at sea level [s]	287	*251*
Thrust in vacuum [kN]	1322	1397
Thrust at sea level [kN]	1190	1040

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.

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 3. Many design similarities exist to the methane precursor type with the main architecture change being two separate turbopumps for the LOX and LH2 sides.

Table 3: Calculated technical data of PROMETHEUS H(ydrogen) as used by reusable and expendable main stage

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

The engine masses are estimated at 1750 kg with the large nozzle for the upper stage and at 1385 kg for the booster stage. These values are equivalent to vacuum T/W of 80 and 95.

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 [12]. Currently, Vinci is the most powerful engine of its type worldwide. The good performance data of this engine (Table 4) makes it attractive for powering the upper or kick-stages of the 3STO-concepts described in section 4.

Table 4: 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

2.3 RLV recovery by "in-air-capturing" (IAC)

Techniques of powered return flight like LFBB obligate an additional propulsion system and its fuel, which raises the stage's inert mass. The patented "In-air-capturing" [13] offers a different approach with better performance: The winged reusable stages are to be caught in the air, and towed back to their launch site without any necessity of an own propulsion system [14]. The idea has similarities with the Down-Range Landing (DRL)-mode, however, initially not landing on ground but "landing" in the air. Thus, additional infrastructure is required, a relatively large-size capturing aircraft. Used, refurbished and modified airliners should be sufficient for the task.

From a performance perspective, the IAC mode is highly attractive. In a systematic comparison of different RLV-stage return modii [2, 9, 17] with all launchers generically sized for the same GTO mission, the IAC-mode constantly shows a performance advantage compared to alternate modes. This result was obtained not only when compared to the LFBB with turbojet flyback but also in comparison to the DRL-mode used by SpaceX for GTO-missions. 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 [9]. In combination with the performance advantage the "in-air-capturing"-method based on current analyses seems to be an attractive technology for future RLV.

Thus, "In-air-capturing", revealed as one of the most promising approaches, is the chosen return mode of the RLV investigated in this paper.

A schematic of the reusable stage's full operational circle when implementing IAC is shown in Figure 1. 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.

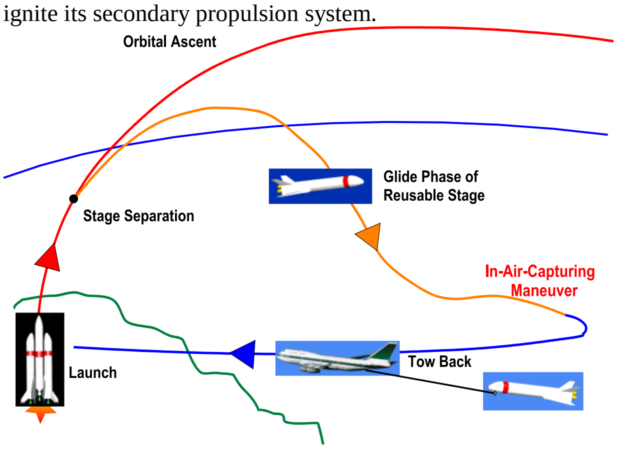

Figure 1: Schematic of the proposed in-air-capturing

Differently, within the in-air-capturing method, the reusable stage is awaited by an adequately equipped large capturing aircraft (most likely fully automatic and 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 unpowered 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 up to about two minutes. The entire maneuver is fully subsonic in an altitude range from around 8000 m to 2000 m [15]. 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.

The selected flight strategy and the applied control algorithms show in simulations a robust behavior of the reusable stage to reach the capturing aircraft. In the nominal case the approach maneuver of both vehicles requires active control only by the gliding stage [15]. Simulations (3DOF) regarding reasonable assumptions in mass and aerodynamic quality proof that a minimum distance below 200 m between RLV and aircraft can be maintained for up to two minutes [15].

In contrast to the previously described capturing strategy, recently a different optimal control approach to the problem of "in-air-capturing" has been attempted by DLR [16], investigating both passive and active (cooperative) RLV and towing aircraft operations. The initial capturing approach and a subsequent second attempt for capture in case of a catching failure have been analyzed. Based on the variety of optimized trajectories, the first in-air-capturing maneuver seems to be feasible from a considerable range of initial relative positions between the launch vehicle and the aircraft. Regarding redundancy and repeatability of the capturing maneuver, additional studies have been performed focusing on a recapturing try in case of miss. This task becomes more challenging than the baseline scenario of both vehicles being in parallel formation flight. A second capturing attempt after an unsuccessful first try is found possible under the condition that the aerodynamic drag of the tow-aircraft being highly adjustable [16]. Further studies in the future will evaluate if the alternative capturing approach might offer advantages compared to the baseline formation flight.

DLR together with European partners is currently preparing for flight testing the “in-air-capturing”-method on a laboratory scale by using two fully autonomous test vehicles. Preliminary results are already available and are published in [6, 9]. The EC funded project FALCon should bring the TRL of the advanced IAC-recovery method beyond 4 in 2022.

After DLR had patented the “in-air-capturing”-method (IAC) for application in future RLVs, similar approaches have been proposed. However, those named *mid-air retrieval* or *mid-air capturing* are relying on parachute or parafoil as lifting devices for the reusable parts and helicopters as capturing aircraft. Rocket Lab is targeting such capture attempt for the *Electron* micro-launcher’s first stage [18].

3 TSTO-CONFIGURATIONS TO GTO

3.1 Type RLVC4-II-A preliminary evaluation

Separated payload mass to GTO of more than 14 tons has been previously shown [3] achievable under conservative assumptions while using an RLV. However, it had to be acknowledged that the payload in this case is only a minor portion (35%) of the orbit injection mass. A large H200 upper stage needs either to be pushed in a graveyard orbit or to be safely deorbited. In any case this process needs significant effort and will be costly. Without any detailed analysis of the upper stage deorbiting process, a mass contingency of 2000 kg has been assumed. This obvious shortcoming justified looking into alternative 3-stage launcher concepts.

3.2 Outlook LEO

Obviously, the TSTO-option remains attractive for Low-Earth-Orbit missions using the smaller 3STO-configurations described in the next paragraph designed for GTO-reference (see example in section 4.3 for the ISS-mission).

4 3STO-CONFIGURATIONS TO GTO

Going for a 3-stage architecture is offering the potential of major performance improvement for the GTO mission because a much smaller inert mass will have to be injected in a high-energy orbit. 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.

A preliminary architecture definition of the 3STO launcher is visible in Figure 2. The expendable segment shown on top resembles the Ariane 5 or 6 core stages. The upper stage has been arranged completely under the fairing.

4.1 Orbit injection strategy GTO

The transfer into GTO with a TSTO is straightforward: the insertion is done directly and following SECO the payload is in the GTO specified. Contrary to that 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, 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.

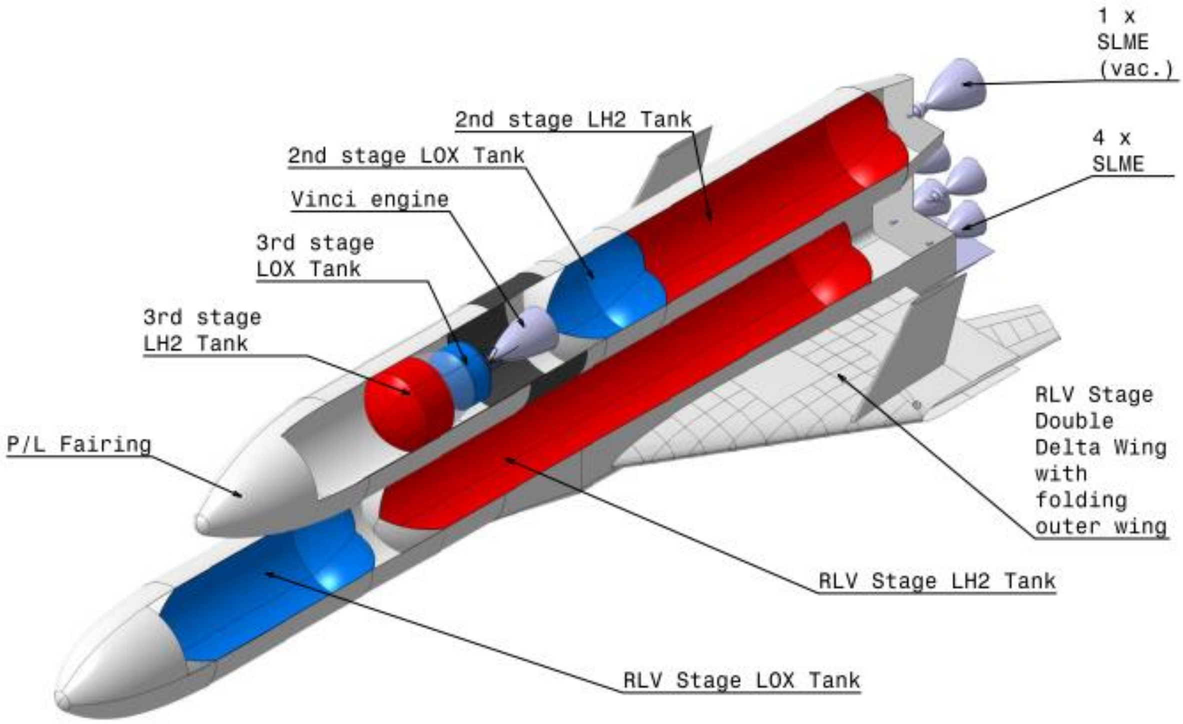

Figure 2: Launcher architecture sketch of 3STO RLVC4-III-B configuration

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 250 km or 600 km perigee and 35786 km apogee and 5.4° inclination. All stages' major events are plotted in Figure 3. The ballistic arc of the main expendable stage and its splash-down in the Pacific w/o active de-orbiting is shown in Figure 4.

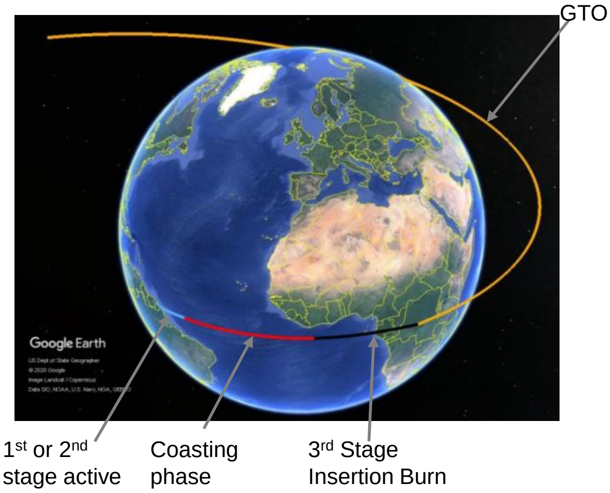

Figure 3: Exemplary ascent profile of 3STO configurations in GTO

Figure 4: Exemplary 2nd stage passive deorbit in Pacific Ocean for GTO-mission

4.2 Type RLVC4-III-A

The 3-stage "HHH-configuration" RLVC4-III-A with SLME and Vinci would deliver almost 25 t separated payload in GTO in a single flight [3]. Even for multiple-satellite launch this performance capability seems to reach beyond current requirements. Several other missions (e.g. for exploration) could be imagined making use of such capacity, however, then potentially requiring an even larger fairing. The very high performance RLVC4-IIIA showed a clear, though theoretical advantage in recurring costs when compared to other 3STO-concepts [3]. However, this advantage in specific cost can only be realized if this launcher is used for most of its missions at close to its maximum capacity which might become challenging with 25 t GTO payload. Research on the RLVC4-III-A has not been continued for this paper.

4.3 Type RLVC4-III-B

Fulfilling the reduced payload requirement of less than 15 t in GTO but maintaining the architecture of three hydrogen stages ("HHH"), necessitates significant reductions in the propellant loading and size of all three stages. The RLV's propellant loading has been reduced in an iterative sizing process by more than 50% compared to the H750. The number of SLME needed for lift-off is no more than four engines. The 2nd stage is only slightly smaller than with concept –III-A in order to remain compatible with the high thrust SLME engine and is defined as an H150. The upper stage also sees a major size and mass reduction (H14) and moves under the fairing as an external diameter of 5.4 m is no longer suitable for such loading if the stage's dry mass should be attractive. Total length of the ELV-segment is considerably reduced.

The RLVC4-III-B overall dimensions are listed in Table 5.

Table 5: Major stage dimensions 3STO RLVC4-III-B

H335 RLV stage	
total length (incl. bodyflap)	59.5 m
fuselage diameter	5.4 m
total span (deployed wing)	35.5 m
H150 ELV	
total length (incl. fairing)	46.5 m
fuselage diameter	5.4 m
H14 ELV (under fairing)	
total length w/o Vinci	6.5 m
fuselage diameter	4.0 m

A stage design with a variable wing offers some advantages over a fixed wing design. First, the bow shock of the fuselage might impinge on the wing structure of substantial span and interact with the respective leading-edge shock which leads to extensive heat loads at the affected wing parts which in turn demands for a reinforced TPS. This phenomena was observed in several DLR studies and was identified as being more critical, the higher the re-entry velocity [2, 20, 21]. Hence, with retractable wings the effective span during re-entry could be limited to make sure that the wings are not lying within the shock-shock interaction. When transitioning to subsonic speed, the wing could be extended to allow for a higher L/D; if adequately designed even higher than with a fixed-wing configuration.

Variable geometry wings in aeronautics have been under investigation at least since the mid of the 20th century and numerous concepts and operational aircraft have been studied and realized. RLV first stages with variable wings have been considered in the USSR in the context of Energia Buran evolution and later also in DLR [24]. Recently, a new investigation on potential updates of the SpaceLiner Booster has been furthermore looking into swept-wing design options [20].

The wing geometry parameters and the wing position with respect to the fuselage are offering several degrees of freedom to the design. Moreover, the impact of parameter variation on the different disciplines is strongly coupled. E.g. wing geometry is affecting mass and vehicle CoG-position while both impact flight dynamic behavior and trimming.

A favorable swept-wing configuration was found by comparing a vast range of different possible wing configurations in a partially automatic variation of parameters to allow for a design that fulfils all requirements:

- High L/D of at least 6 allowing for adequate gliding path angles during In-Air-Capturing
- Small span in hypersonics to avoid shock-shock interaction
- Landing Speed of ≤ 105 m/s
- Trimmable to high AoAs in hypersonics to generate lift and consequently minimize heat flux

The convergent preliminary design of the variable-wing first stage is shown in Figure 5. The wing span of the inner fixed part of the wing is around 20.2 m which enables a total span with wings extended of 35.5 m (see Table 5). Figure 5 also shows the difference between the re-entry configuration with the movable part of the wings retracted (bottom) and the transonic and subsonic flight configuration with wings extended (top). The swiveling wing is stored inside the fixed wing during re-entry and is connected via a forward outboard pivot-point mounting to the wing structure. It is visible that the outer wing in stored position extends rearward over the chord length of the inner wing so that the wingtip parts extend outside. This makes it necessary for the inner wing to be open at its trailing edge to accommodate the protruding part of the outer movable wing.

An internal view of the preliminary wing design including structures layout of both wing parts and the flaps is shown in Figure 6. The upper and lower rear parts of the fixed wing can be deployed as spoilers and thus adopt the role of non-existing trailing edge flaps. The inner rib and spar structure has to leave out space to accommodate for the outer wing. The landing gear box is positioned to consider sufficient distance to the CoG while allowing AoAs of 12° during landing. Any detailed landing gear design is not yet performed which might require modifications to the structural layout presented in Figure 6.

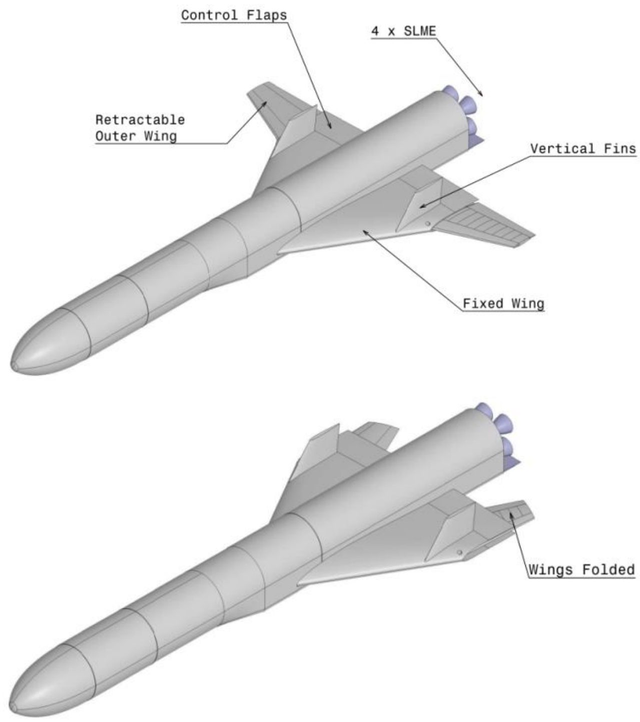

Figure 5: Conceptual design of variable wing RLV stage RLVC4-III-B with wings extended (top) and wings stored (bottom)

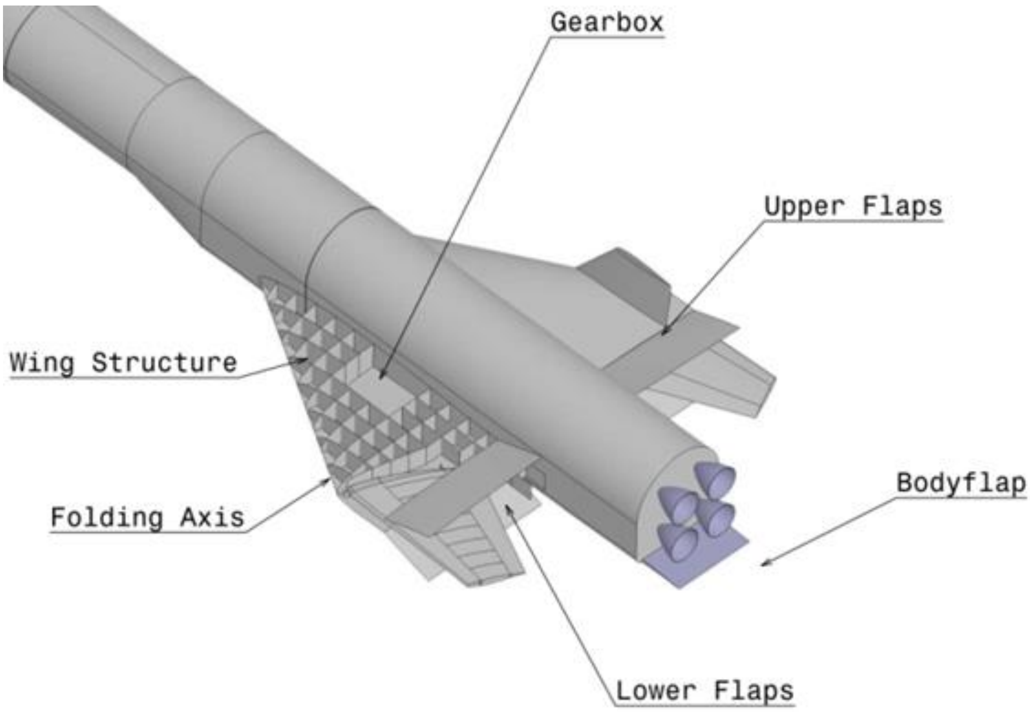

Figure 6: Preliminary structural layout of the variable wing stage RLVC4-III-B

The achievable lift-to-drag ratio of the variable wing configuration is shown in Figure 7. A subsonic L/D of around 6 for trimmed conditions is found in a relatively broad AoA-range. This is an acceptable value although slightly reduced compared to the configuration's previous analysis [3]. This allows the unpowered stage a flight path angle of -9.5° during the In-Air-Capturing formation maneuver. The maximum trimmed L/D in hypersonics is 2 at an AoA close to 15°. However, during re-entry the AoA is ought to be as high as safely controllable to produce sufficient lift to keep the maximum heat flux within boundaries and increase drag to decelerate the vehicle. Hence, the actually flown L/D at re-entry conditions with AoAs of around 40°-50° is around 1.

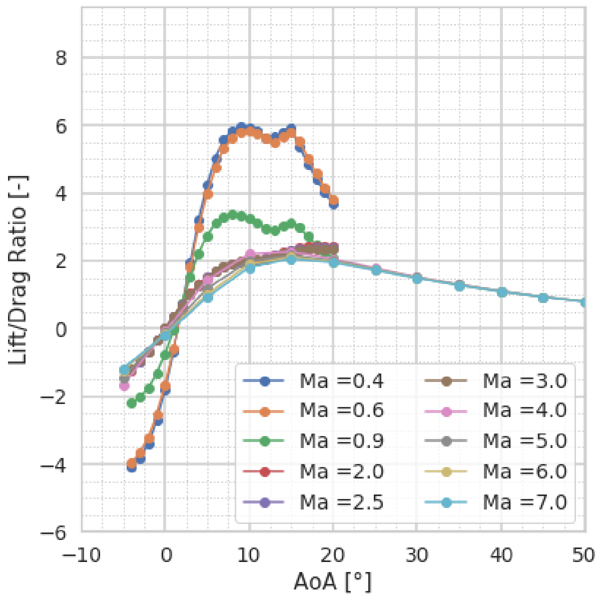

Figure 7: L/D of the variable wing RLVC4-III-B configuration in trimmed flight

The mass breakdown of the complete RLVC4-III-B launcher with the variable-wing first stage is listed in Table 6. The reusable first stage dry mass reaches 71 tons. A structural index of 20.7% obtained for a simplified component mass breakdown is probably realistic. However, it is important to note that the variable wing design is related to a certain amount of weight uncertainty. The cut-out in the fixed wing part to accommodate the stored outer wing as shown in Figure 6 has the disadvantage of a less efficient structural design and hence increased weight. Further, the sweep-wing's pivot point sees a major load concentration and some kind of mechanism for wing deployment is to be added. All these factors generate additional mass and need closer analyses in the future. Compared to some large military aircraft of similar size which are employing sweep wings, the RLVC4-III-B sees lower mechanical loads in operation (less dynamic pressure) and only requires one single wing deployment. Continuous adaptation of the sweep angle to changing flight conditions is not necessary for the RLV-mission, allowing for some mass savings compared to such military aircraft.

The expendable cryogenic main stage loading is around 150 tons and therefore the layout is close to the early Ariane 5 "G" EPC-stage with common tank bulkhead. The 3rd stage is assumed to use cryogenic LOX-LH2 and to be equipped with the existing and qualified Vinci engine (see Table 4). The propellant loading is approximately 14 tons. The 20 m length fairing is selected. Total lift-off weight is approximately 622 tons, significantly below that of the Ariane 5 and 6 ELVs despite considerably increased payload mass and reusability of the first stage. The explanation is related to the more efficient, fully cryogenic propulsion system

Table 6: Launcher masses by stage 3STO RLVC4-III-B, GTO mission

1st stage	**H 335**
Ascent Propellant	335.0 t
Dry Mass	71.0 t
GLOW	414.2 t
Structural Index incl. Engines	20.7%
2nd stage	**H150 ELV**
Ascent Propellant	150.0 t
Deorbit Propellant	-
Dry Mass (w/o fairing)	17.31 t
GLOW (incl. fairing)	174.2 t
Structural Index incl. engine and 3rd stage adapter w/o fairing	11.3%
3rd stage	**H14 ELV**
Ascent Propellant	14.2 t
Deorbit Propellant	0.5 t
Dry Mass	4.2 t
GLOW (incl. P/L)	33.3 t
Structural Index incl. engine and SYLDA	26.8%
Separated Payload GTO	**13620 kg**
Total GLOM	**621.7 t**

Figure 8 shows the ascent trajectory into the low transfer orbit (blue line) and subsequently into GTO (orange line). The ascent burns of 1st and 2nd stage propel the 3rd stage and payload to a low altitude of roughly 134 km. RLV stage separation occurs at slightly less than 2 km/s or Mach 6.02 and an altitude of 60.4 km (Figure 8) related to a dynamic pressure well below 1 kPa, allowing a safe separation maneuver of the RLV and ELV stages in parallel arrangement. Further, the 2nd stage ignition is delayed by several seconds that the RLV has sufficient time for distancing. Full thrust of the single SLME on the H150 is assumed to be reached 8 s after separation when the upper segment is already in more than 66 km altitude (Figure 8). After approximately another 5 minutes of acceleration the MECO-conditions of the transfer LEO are achieved. The 3rd stage coasts along the ballistic trajectory until reaching the equator [3] where it ignites its engine to provide the final Δv required to reach GTO.

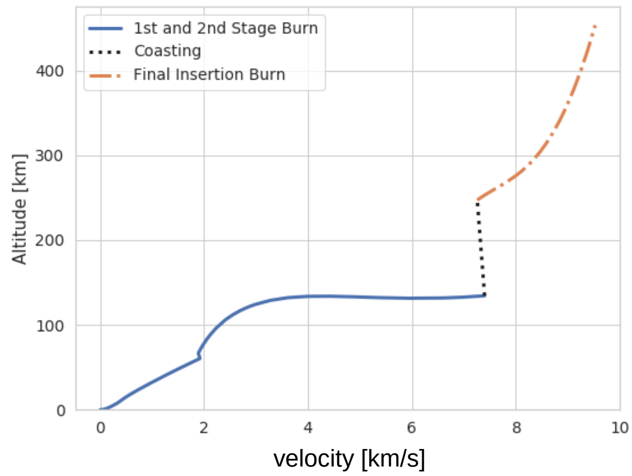

Figure 8: Ascent trajectory of 3STO RLVC4-III-B in 250 km x 35786 km GTO

Due to the comparably low re-entry velocity of around Mach 6 the heat flux and temperatures are moderate with the highest local temperature not exceeding 1200 K. The estimated TPS mass is merely around 2.5 tons for the RLVC4-III-B. Note that some flaps might require additional protection if subjected to increased loads when deflected or when seeing flow re-attachment. Due to the specific design of the RLVC4-III-B aerodynamic configuration a critical bow-shock-generated shock interaction is not to be expected. An external metallic cover-sheet on the TPS could be attractive for operational reasons, however, would increase the system mass.

The descent maneuver of the RLV is performed in four steps. After separation and ascending ballistically to an apogee of 99 km, the vehicle reenters the atmosphere with the outer wing still in stored position and constant AoA of 45° until the n_z load factor is approaching 3.5 g. During this second phase of rapidly increasing dynamic pressure, the angle of attack is controlled to limit n_z to the specified maximum of 3.5 g. In the third phase, a banking maneuver is introduced to turn the azimuth about 180°, thus heading back to the Kourou launch site. A bank angle of 50° is chosen for approximately 200 s. The movable outer wing is preliminarily assumed to be deployed at supersonic Mach number of 3 at an altitude of around 25 km. These conditions might be slightly adapted in future work to perform the transition maneuver at minimum dynamic pressure.

Afterwards in the fourth phase, when the banking maneuver is complete, the RLV is in transonic flight at an altitude around 20 km. The stage is now in gliding flight close to its maximum subsonic L/D. This is the optimum condition for approaching the rendezvous area for performing the in-air-capturing maneuver. The flight path angle during this part of subsonic descent is stabilized around -10°. The calculated reentry trajectory of the RLV in GTO-mission up to capturing is shown in Figure 9. Note the smooth reentry gliding conditions.

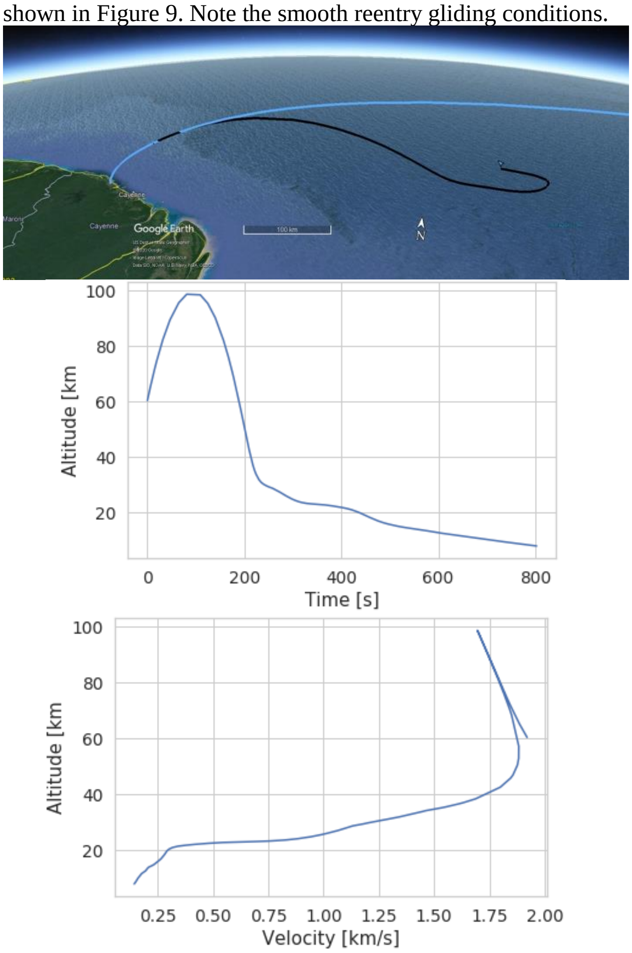

Figure 9: 3STO RLVC4-III-B H335 descent trajectory GTO-mission

The 3STO-concept for high-energy GTO missions is evaluated for its performance in LEO as TSTO with the upper stage removed and only the RLV-stage H335 and expendable H150 remaining. The mission to the ISS is targeting a 350 km x 400 km, 51.6° intermediate orbit. Final approach to the ISS is performed by an orbital transfer module or space-tug like the former ATV which is part of the payload mass and not regarded here as a launcher stage. The orbital injection conditions of the expendable stage are requiring an active deorbiting of the H150. Again, the stage's splashdown is foreseen in the Pacific Ocean in the vast remote areas east of New Zealand. A rough estimation shows that 1.2 tons of propellant should be sufficient for the deceleration burn including contingencies for engine chill-down and start-up fuels.

Table 7 lists a different dry mass of the H150 compared to the same stage's mass in the GTO mission (compare Table 6) because the complete avionics bay of the upper stage is to be added to the 2nd stage. The payload mass of more than 21 tons is slightly above the proven Ariane 5 ES performance in a similar mission.

Table 7: Launcher masses by stage TSTO RLVC4-III-B, LEO (ISS) mission

1st stage	**H 335**
Ascent Propellant	335.0 t
Dry Mass	71.0 t
GLOW	414.2 t
Structural Index incl. Engines	20.7%
2nd stage	**H150 ELV**
Ascent Propellant	150.0 t
Deorbit Propellant	1.2 t
Dry Mass (w/o fairing)	18.4 t
GLOW (incl. fairing)	176.5 t
Structural Index incl. engine and payload adapter w/o fairing	11.3%
Separated Payload LEO	**21270 kg**
Total GLOM	**619.9 t**

4.4 Type RLVC4-III-C

A similar configuration as the RLVC4-III-B in size and using the same SLME main rocket propulsion, however, with a different, fixed double delta wing geometry has been defined as a potential alternative. This concept described in more detail in reference [3] has not been updated for this paper but serves with its geometry as the baseline for the following RLV-concepts with gas-generator cycle rocket engines.

4.5 Type RLVC4-III-G

The previously described -A/-B/-C-versions are based on the development of a new, advanced closed cycle engine, the SLME. Although offering major launcher system advantages, such development has not yet started in Europe. Therefore, 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 3.

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 435 tons, which is an addition of 100 tons compared to the III-B version. Nine engines are used on the RLV-stage to perform the lift-off with sufficient thrust-to-weight ratio. The 2nd stage is marginally larger with a propellant mass of 152 tons and is limited in its size due to available engine thrust. The 3rd stage for the GTO-insertion is implemented very similar to III-B as H15 with the cryogenic LOX-LH2 Vinci engine.

In order to limit the length of the RLV-stage and increase the lift capabilities during re-entry, the fuselage diameter is increased to 6 m. Additionally, it enlarges the separation space between the nine engines for a more simplified integration. These are arranged with one center engine and the other eight placed on an octagon around it (Figure 10).

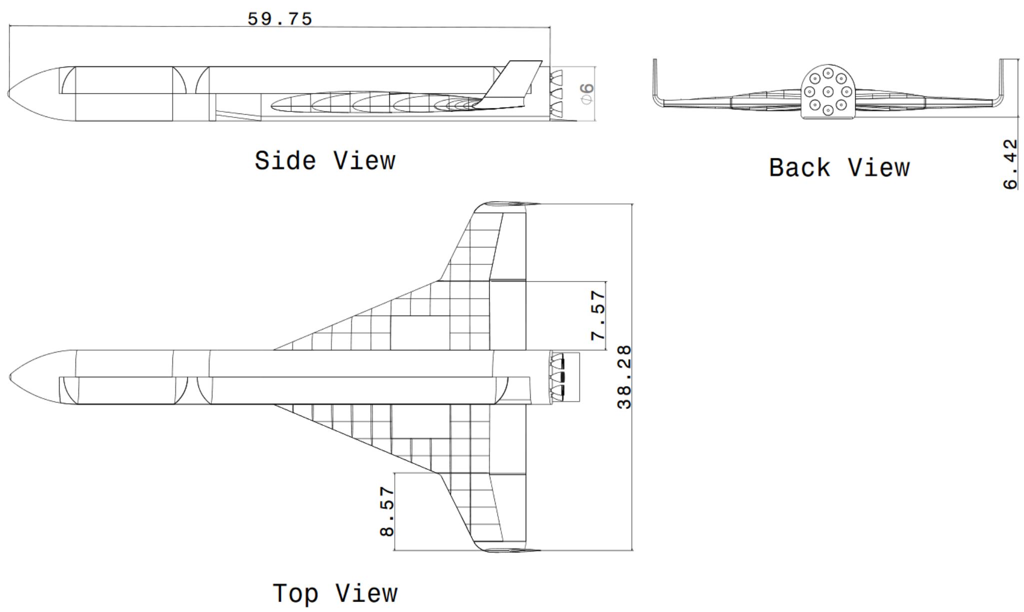

Figure 10: Preliminary structural layout of the fixed wing stage RLVC4-III-G

In comparison with the III-C variant [3], the fixed wing is slightly changed to increased wing aspect ratio which enables improved L/D-values in the subsonic regime while still having longitudinally stable aerodynamic behavior during the hypersonic re-entry. The maximum trimmed L/D at Mach 0.4 is calculated to reach 6.

The overall dimensions of the launcher are listed in Table 8. The RLV type G is only slightly longer than the -B variant due to its increased tank diameter.

Table 8: Major stage dimensions 3STO RLVC4-III-G

H435 RLV stage	
total length (incl. bodyflap)	62.9 m
fuselage diameter	6.0 m
total span	36.8 m
H152 ELV	
total length (incl. fairing)	46.5 m
fuselage diameter	5.4 m
H15 ELV (under fairing)	
total length w/o Vinci	6.5 m
fuselage diameter	4.0 m

The ascent profile is similar compared to the -B/-C-variants, with a few key distinctions. Because of the lower performance engine in combination with a similarly sized 2nd stage, the RLV-stage has to accelerate to a higher separation velocity of 2.33 km/s (Mach 7.45). The 2nd stage injects the stack, consisting of the 3rd stage and payload, into the desired 35 km x 400 km transfer orbit at a separation altitude of 155 km. GTO insertion is again performed by the 3rd stage after a coasting phase and subsequent ignition close to the equator.

Due to the higher separation velocity of the RLV-stage, the re-entry trajectory leads to an amplified heat flux and accordingly higher temperatures on the TPS, peaking at 1370 K. Consequently, the TPS dimensioned to withstand these temperatures requires 3.3 tons of material (800 kg more than for the B-version). The calculated re-entry trajectory is shown in Figure 11.

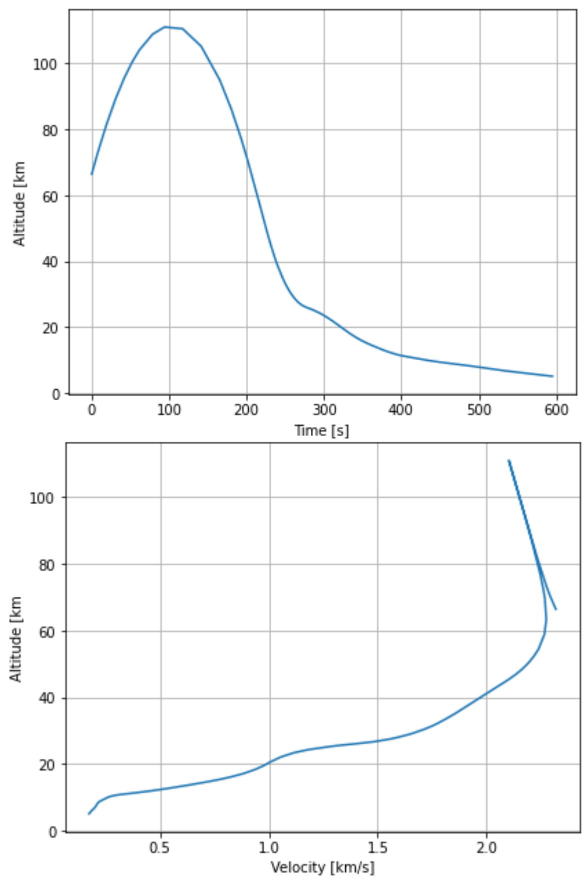

Figure 11: 3STO RLVC4-III-G H435 descent trajectory GTO-mission

Table 9 lists the overall mass breakdown of the gas-generator variant. The dry mass of the RLV-stage is about 85.4 tons which results in a structural index of 19.3%. The dry mass is increasing by more than 14 t (+ 20%) compared to III-B while the structural index of the significantly larger stage is slightly lower. For the 2nd stage, a reduction in engine weight, due to the change from the larger SLME to the PROMETHEUS-H engine, leads to an overall decrease in 2nd stage dry mass while maintaining the same overall architecture. 14.5 tons of propellant are required for the GTO-insertion burn of the 3rd stage, slightly more than for the -B-variant. The partially reusable launcher with gas generator cycle engines achieves a separated GTO payload mass of almost 14 tons. The gross lift-off mass of 735.9 tons is still significantly below Ariane 5 and 6 and the payload mass fraction into GTO is still reaching an impressive 1.90%.

Table 9: Launcher masses by stage 3STO RLVC4-III-G, GTO mission

1st stage	**H 435**
Ascent Propellant	435.0 t
Dry Mass	85.4 t
GLOW	527.7 t
Structural Index incl. Engines	19.3%
2nd stage	**H152 ELV**
Ascent Propellant	152.0 t
Deorbit Propellant	-
Dry Mass (w/o fairing)	15.83 t
GLOW (incl. fairing)	174.7 t
Structural Index incl. engine and 3rd stage adapter w/o fairing	10.2%
3rd stage	**H15 ELV**
Ascent Propellant	14.5 t
Deorbit Propellant	0.5 t
Dry Mass	4.0 t
GLOW (incl. P/L)	33.5 t
Structural Index incl. engine and SYLDA	25.5%
Separated Payload GTO	**13967 kg**
Total GLOM	**735.9 t**

4.6 Type RLVC4-III-M

After studying launcher configurations with hydrogen fuel in all of their stages, the impact of switching to hydrocarbons should be investigated. Table 2 shows that the Isp of the PROMETHEUS engine with LOX-LCH4 propellants is approximately 90 s less than the H-variant with similar operating conditions. The increased bulk density of the oxygen-methane propellant combination should limit the size of the stages. Even the M-type (for methane) is defined with the 3rd stage using hydrogen and the Vinci engine similar to all other RLVC4-III variants.

The lower segment with RLV and ELV stages will be investigated in two sub-versions: a full methane powered version using this propellant in the winged RLV and the 2nd stage ELV and a hybrid configuration with methane used in the RLV booster but swapping to hydrogen in the 2nd stage similar to the H152 of the G-type.

Iterative sizing of the RLVC4-III-M-versions is not finished yet and results will be presented in a future paper.

5 CONCLUSION

Different partially reusable launcher concepts have been investigated in 3-stage to orbit (3STO)-configurations for heavy-lift GTO-missions to be launched from Kourou's CSG. All concepts are of asymmetric architecture with a winged RLV booster in parallel arrangement to expendable stages. Baseline for RLV-recovery is the "in-air-capturing" method showing superior performance to all alternative options. Preferred propellant choice is the combination of hydrogen with LOX.

The multi-disciplinary preliminary sizing process demonstrates that heavy payload performance of more than 13.5 tons is achievable in GTO when using a parallel arrangement of RLV and ELV stages. The 3-stage concepts with all stages implementing hydrogen achieve attractive payload ratios of up to 2.2% in GTO. The less demanding LEO-mission is better served by switching to TSTO and could be realized simply by removing the upper stage. An ISS-resupply mission has been analyzed as a typical case and the III-B-variant can deliver more than 21 tons separated payload to the medium inclination LEO. All investigated partially reusable launchers are capable of delivering multiple payloads with masses significantly beyond the capabilities of A5 and A64 into GTO while GLOW remains considerably lower.

A comparison between closed and open cycle LOX-LH2-rocket engines on launcher performance shows that GLOW increases by 30% for the less efficient gas-generator cycle and RLV dry mass is increasing by approximately 20%. Thus, cost savings by simplified engines are to be compensated by a larger and hence more expensive stage. The impact assessment on actual operating costs requires a dedicated study considering several different annual launch scenarios.

The investigations of promising next generation European launcher concepts are to be continued and refined.

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