Powerful & Flexible Future Launchers in 2- or 3-stage Configuration

Martin Sippel, Sven Stappert, Leonid Bussler, Christian Messe
Martin.Sippel@dlr.de Tel. +49-421-244201145
Space Launcher Systems Analysis (SART), DLR, Bremen, Germany

Semi RLV configurations are investigated with reusability of 1st or booster stages arranged in parallel with an expendable upper compartment. The non-symmetrical architecture consists of a winged RLV-stage and attached ELV-part comprising either one or two stages. The rocket propulsion is mostly cryogenic LOX-LH2 with the option of a storable propellant upper stage.

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 some indicators on the configuration's launch cost efficiency are provided.

Keywords: RLV, TSTO, trajectory, LOX-LH2-propulsion, SpaceLiner, LFBB, 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
AOA	Angle of Attack
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

Europe's Ariane 6 developments are progressing [1] with the inaugural flight of A6 expected in the coming year. 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* [2]. 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 [2].

RLV configurations with partial reusability of 1st or booster stages are also in focus of ongoing system studies at DLR's space launcher system analysis department SART. Several tandem TSTO-launchers with 7.5 t GTO performance are under investigation; not only for one preferred type but for different return and recovery modes, as well as different propellant and engine cycle options [3, 4].

Beyond this medium size TSTO, another class of RLV is preliminarily defined at DLR with parallel stage arrangement serving the payload class segment of Ariane 6 or above. A winged stage is to be connected to an expendable upper stage segment of various size and internal architecture. This powerful launcher option is in focus of this paper.

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 or3 stages is not obvious because on the one hand 3STO become much smaller while on the other hand additional stages add cost.

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) [9]. 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 [9] 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 [9].

Table 1: SpaceLiner Main Engine (SLME) technical data [9] 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 [9].

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 [10]. Currently, Vinci is the most powerful engine of its type worldwide. The good performance data of this engine (Table 2) makes it attractive for powering the upper or kick-stages of the 3STO-concepts described in section 4.

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

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

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” [11] 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 [12]. 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. Figure 1 presents a comparison of the inert mass ratio for generic TSTO-launchers and different return modes of the reusable first stage. All launchers have been sized for 7.5 tons GTO payload with a variation in separation Mach-number of the RLV [8]. As mission and stage number are identical, the inert mass ratio can be presented as function of the total ascent propellant loading. RTLS for GTO results in excessively high stage size and inert mass ratio and has hence been excluded from further studies with GTO-mission.

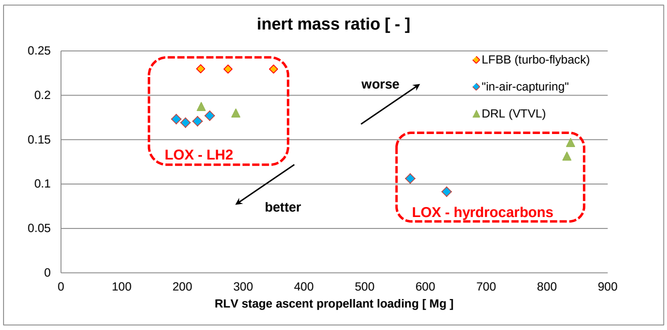

Figure 1: Inert mass ratios of different RLV-return modes (all same GTO mission) [14]

In all of the investigated cases the IAC-mode RLV stages have a performance advantage not only when compared to the LFBB with turbojet flyback but also in comparison to the DRL-mode used by SpaceX for GTO-missions. Thus, “In-air-capturing" is the chosen return mode of the RLV investigated in this paper.

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

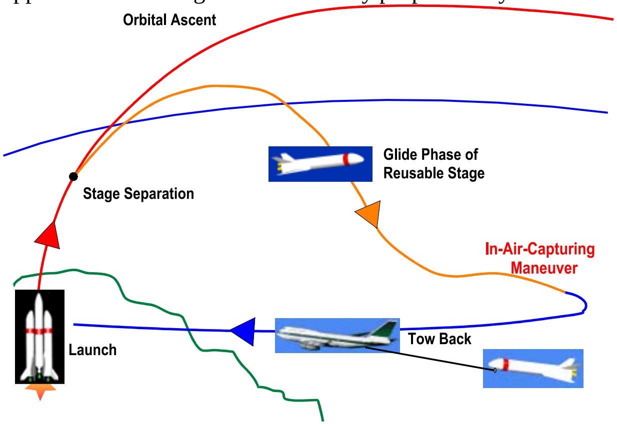

Figure 2: Schematic of the proposed in-air-capturing

Differently, within the in-air-capturing method, the reusable stage is awaited by an adequately equipped large capturing aircraft (most likely fully automatic and 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 [13]. 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. 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 [13].

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 [5, 8]. 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, two 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 (ULA "Vulcan"-launcher) and helicopters as capturing aircraft.

3 TSTO-CONFIGURATIONS

3.1 Type RLVC4-II-A

The RLVC4-II-A is a two-stage to orbit (TSTO) configuration consisting of a winged, reusable first stage and an expendable upper stage arranged in parallel. Such non-symmetrical designs have been used in the past for the Space Shuttle and Energia-Buran and are the reference for the Phantom Express RLV-concept of Boeing, partially funded by DARPA.

A sketch of the architecture is shown in Figure 3. The RLV stage fuselage diameter is chosen at 8.6 m, exactly as the SpaceLiner Booster stage SLB7-3 tank diameter [16, 18]. However, the overall length is reduced from more than 82 m to 60 m for the RLVC4-II-A. As the SpaceLiner concept is defined as a fully reusable TSTO launcher to LEO [18, 19] which requires a heavy lift first stage, the less ambitious, partially expendable RLVC4-II-A can achieve a higher payload ratio and hence, the size of the RLV-stage can be decreased while the number of SLME is reduced from 9 to 7 engines.

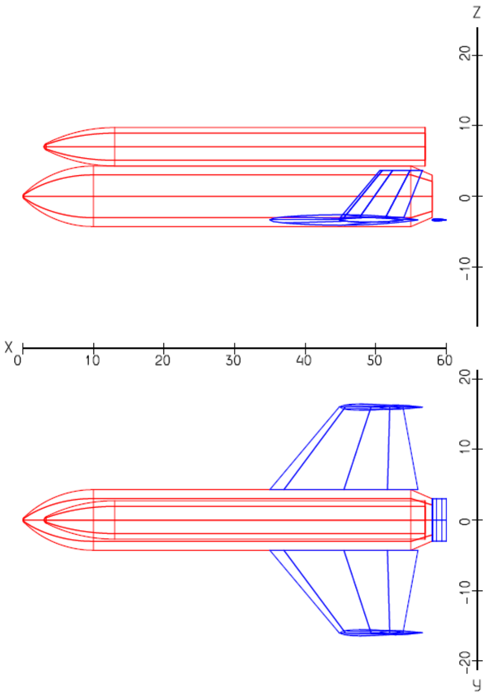

Figure 3: Sketch of RLVC4-II-A (H750 H200) parallel architecture

The smaller vehicle further requires a smaller wing, preliminarily selected in trapezoidal shape with 40° leading edge sweep. Wing flaps and a body flap are used for trimming and flight control. The span is 32 m. Major driver of this less elegant design (Figure 4) compared to the original SLB7-3 is the intention of gaining operational experience with the RLVC4-II-A and to considerably reduce the risk of a following SpaceLiner development. If the stage has already major geometric similarities this should potentially diminish development cost which had been the initial motivation to study this type.

With the same philosophy in mind, the expendable stage diameter is 5.46 m (as Ariane 5 and -6 core) and its length reaches 54 m. The H200 is a lengthened version of the Ariane 5 H170 with common bulkhead structural architecture. This is a deviation from the separate bulkhead structure of the A6 LLPM [1] but is not relevant for assessment of general feasibility and is more related to the A6-manufacturing optimization. The H200 is powered by a single SLME with nozzle expansion of 59. The large 20 m type of Ariane 6 is assumed as payload fairing which allows the dual launch of 2 heavy satellites [1]. The principal dimensions of the H750-H200 configuration are summarized in Table 3.

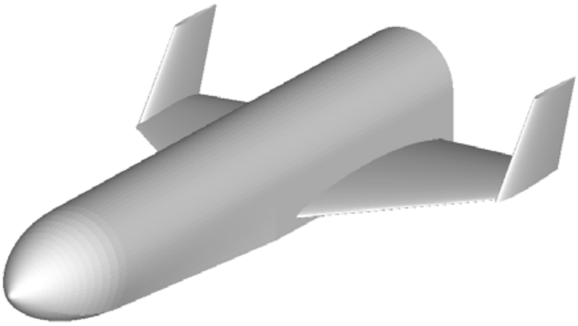

Figure 4: 1st stage RLV-configuration H750 of RLVC4-II-A and of RLVC4-III-A

Table 3: Major stage dimensions TSTO RLVC4-II-A

H750 RLV	
total length	60 m
fuselage diameter	8.6 m
total span	32 m
H200 ELV	
total length (incl. fairing)	54 m
fuselage diameter	5.46 m

The mass breakdown of the H750-H200 configuration is presented in Table 4. The predesigned configuration has an overall gross lift-off weight (GLOW) of 1145.1 Mg of which 893.7 Mg (78%) are attributed to the reusable first stage and 251.4 Mg belong to the expendable upper stage. Most of the reusable first stage component masses are derived from preliminary sizing of the SLB7. The upper stage is relatively heavy for its size but sees significant non-symmetric loads, unusual for vertical launchers and the assumed engine is larger than actually required.

Table 4: Launcher masses by stage TSTO RLVC4-II-A

H750 RLV	
Ascent Propellant	750000 kg
Dry Mass	134700 kg
GLOW	893700 kg
Structural Index incl. Engines	17.80%
H200 ELV	
Ascent Propellant	200000 kg
Deorbit Propellant	2000 kg
Dry Mass	30400 kg
GLOW (incl. P/L)	251400 kg
Structural Index incl. engine and multiple payload adapter w/o fairing	13.2%
separated payload GTO	14200 kg
Total GLOW	**1145100 kg**

The ascent trajectory of the H750-H200 configuration is shown in Figure 5. Reusable first stage separation takes place at an altitude of 79 km, a Mach number of 11.2 and a flight path angle of 13.2°. The choice of the angle γ at separation has a considerable effect on the aerothermodynamic loads experienced by the reusable first stage during atmospheric reentry. The effect of the powerful upper stage engine is visible in the steep ascent. The final altitude of the upper stage at the end of powered ascent is around 250 km and thus close to the desired perigee altitude of the target GTO. The separated payload to GTO is 14.2 Mg.

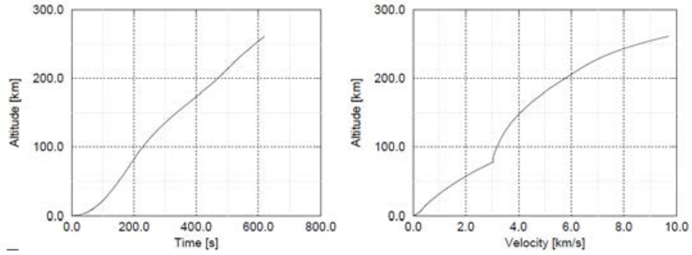

Figure 5: H750 H200 TSTO ascent trajectory to GTO

Following separation from the expendable upper stage, the reusable first stage continues to climb. Shortly after passing its apogee of 110 km, the vehicle is reentering the atmosphere. The angle of attack at the beginning of atmospheric reentry is kept at 45° and after reaching denser layers of the atmosphere it is reduced to limit the normal acceleration to 3.3 g. The altitude profile after separation and the empirically estimated stagnation point heat flux over Mach number are shown in Figure 6. The RLV experiences a moderate peak heatflux at the nose of 326 kW/m² at an altitude of 36 km and corresponding Mach number of 8.0.

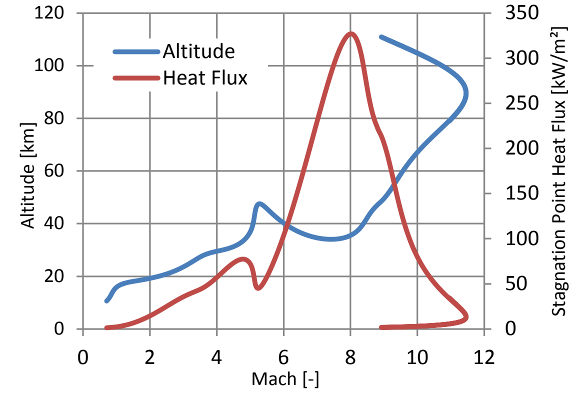

Figure 6: H750 descent trajectory and heatflux

The aerodynamics of the H750 H200 configuration are assessed using fast engineering methods. Special attention is paid to the trimmability of the RLV stage throughout the entire flight regime and its performance at subsonic Mach numbers, i.e. its lift-to-drag ratio. The trimmed, subsonic lift-to-drag ratio of the H750 stage is shown in Figure 7. For Mach numbers of 0.2 to 0.8 the maxima in lift-to-drag are found at angles of attack between 5° and 7°. Maximum, trimmed L/D values are 4.5 for a Mach number of 0.2 and 4.0 for a Mach number of 0.8.

3.2 Preliminary TSTO evaluation

Achieving a separated payload mass to GTO of more than 14 tons under conservative assumptions and using an RLV as first stage is highly impressing. However, it should be acknowledged that the payload is only a minor portion (35%) of the orbit injection mass. The large H200 upper stage needs either to be pushed in a graveyard orbit or to be safely de-orbited. In any case this process needs significant effort and will be costly. Without any detailed analysis of the upper stage deorbitation process, a mass contingency of 2000 kg has been assumed (Table 4). This obvious shortcoming justifies looking into alternative 3-stage launcher concepts.

Further, a maximum trimmed L/D of not more than 4.5 and an RLV-stage dry mass of 135 tons is very demanding for the towing-aircraft of the IAC-recovery mode. Improvements for the stage aerodynamic layout should be considered if investigations on this TSTO-launcher would be continued.

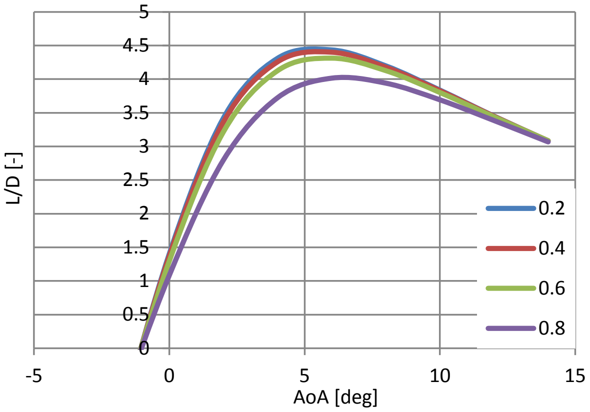

Figure 7: Calculated L/D in subsonic conditions (M= 0.2 – 0.8) for H750 stage (trimmed)

4 3STO-CONFIGURATIONS

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.

After preliminary technical definition, the impact of such 3STO on specific launch costs is to be assessed in a subsequent step.

4.1 Orbit injection strategy GTO

In order to ensure that the uncontrolled descent of the second stage safely occurs in the Pacific Ocean, the ascent phase is split into two steps: First, the second stage is injected into an intermediate orbit with an apogee height of 250 km (RLVC4-III-A) or 600 km (RLVC4-III-B /-C) and a perigee height of 60 km (RLVC4-III-A) or 25 km (RLVC4-III-B /-C). Following its ballistic flight phase after separation, the third stage is ignited so that it reaches the apogee at half burn time, thus performing a Hohmann transfer into the designated GTO with 250 km or 600 km perigee and 35786 km apogee and 5.4° inclination. All stages' major events are plotted in Figure 9.

The ground track of the second stage after its MECO is illustrated in Figure 11.

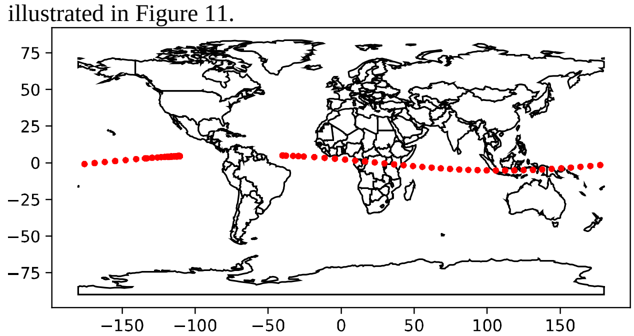

Figure 8: Ground track of uncontrolled second stage descent

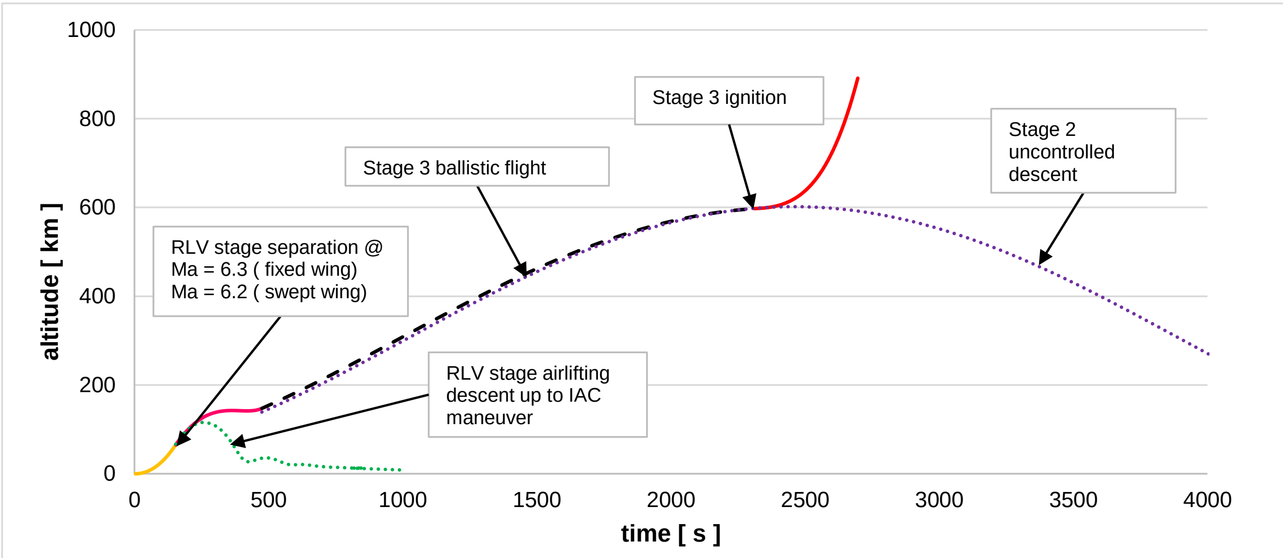

Figure 9: Ascent and descent profiles of different stages for RLVC4-III-B and -C-configurations

4.2 Type RLVC4-III-A

In a first trial the same winged reusable booster H750 of the previous TSTO is analyzed in parallel arrangement to the now 2-stage ELV and its payload compartment with fairing. Essentially, this launcher looks from outside very similar to the type RLVC4-II-A but includes as the main difference two expendable stages. Therefore, the principal dimensions are identical to those already summarized in Table 3.

In order to fit two stages in the available volume of the ELV-segment, the propellant loading has to be adapted. The cryogenic main stage is now reduced to 170 tons loading and therefore the layout becomes very similar with today's operational Ariane 5 ECA H170 with common bulkhead structural architecture, however, improved by a different, more powerful main engine with altered mixture ratio. An additional 3rd stage is to be added which could be either storable or cryogenic. The latter will be equipped with the existing and qualified Vinci engine (see Table 2) and a propellant loading of approximately 26 tons is foreseen. Similarities to the planned Ariane 6 upper stage (ULPM) [1] are limited. However, some upgrade ideas already discussed for the A6 might be useful for implementation in the H26 proposed here. The 20 m length fairing is selected as for the RLVC4-II. Total lift-off weight is growing by approximately 10 tons (Table 5) compared to the RLVC4-II-A, mainly due to the increased payload mass.

Table 5: Launcher masses by stage 3STO RLVC4-III-A

H750 RLV as in Table 4	893700 kg
H170 ELV	
Ascent Propellant	170000 kg
Deorbit Propellant	0 kg
Dry Mass	24950 kg
GLOW including fairing	202000 kg
Structural Index incl. Engine w/o fairing	14.3%
H26 ELV	
Ascent Propellant	25700 kg
Deorbit Propellant	500 kg
Dry Mass	6500 kg
GLOW (incl. P/L)	59375 kg
Structural Index incl. Engine and multiple payload adapter	23.2%
separated payload GTO	24800 kg
Total GLOW	**1155965 kg**

The 3-stage "HHH-configuration" RLVC4-III-A would deliver almost 25 t separated payload in GTO in a single flight. 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 alternative option for the upper stage is a variant with storable propellants. Before any definition of the propulsion system, a hypothetical engine of 47 kN and 324 s Isp has been assumed. Table 6 summarizes the vehicles masses and shows that more than 15 tons payload can be delivered in GTO.

Table 6: Launcher masses by stage 3STO RLVC4-III-A with storable propellant upper stage

H750 RLV as in Table 4	893700 kg
H170 ELV as in Table 5	202000 kg
L38 ELV	
Ascent Propellant	37570 kg
Deorbit Propellant	500 kg
Dry Mass	4400 kg
GLOW (incl. P/L)	59375 kg
Structural Index incl. Engine and multiple payload adapter	11.1%
separated payload GTO	15100 kg
Total GLOW	**1155965 kg**

Separation conditions of the reusable first stage are very similar to the previously described RLVC4-II-A, only slightly below Mach number of 11. Thus, the reentry conditions are also similar to those presented in section 3.1.

The impressive payload capabilities, however, reaching considerably beyond the intended target and the challenges of the H750 stage design mentioned in section 3.2 motivate to explore options for RLV-size reduction aiming for a maximum of less than 15 t in GTO. Such designs should be more in line with expected future European payload requirements and are described in the next sections.

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 sees also a major size and mass reduction (H16) 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 7.

Table 7: Major stage dimensions 3STO RLVC4-III-B

H340 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
H16 ELV (under fairing)	
total length	4.0 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 [4, 16, 17]. 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 [20]. Recently, a new investigation on potential updates of the SpaceLiner Booster has been furthermore looking into swept-wing design options [16].

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 10. 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 7). Figure 10 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.

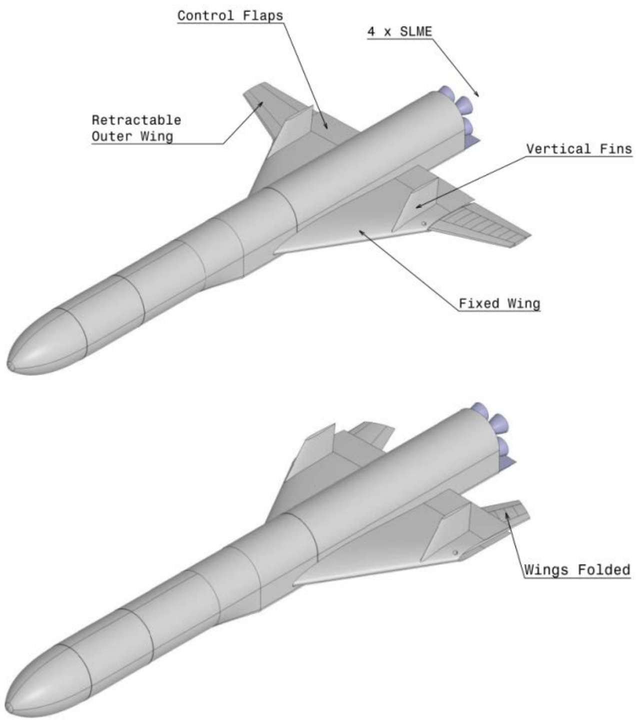

Figure 10: Conceptual design of variable wing RLV stage RLVC4-III-B with wings extended (top) and wings stored (bottom)

An internal view of the preliminary wing design including structures layout of both wing parts and the flaps is shown in Figure 11. 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 11.

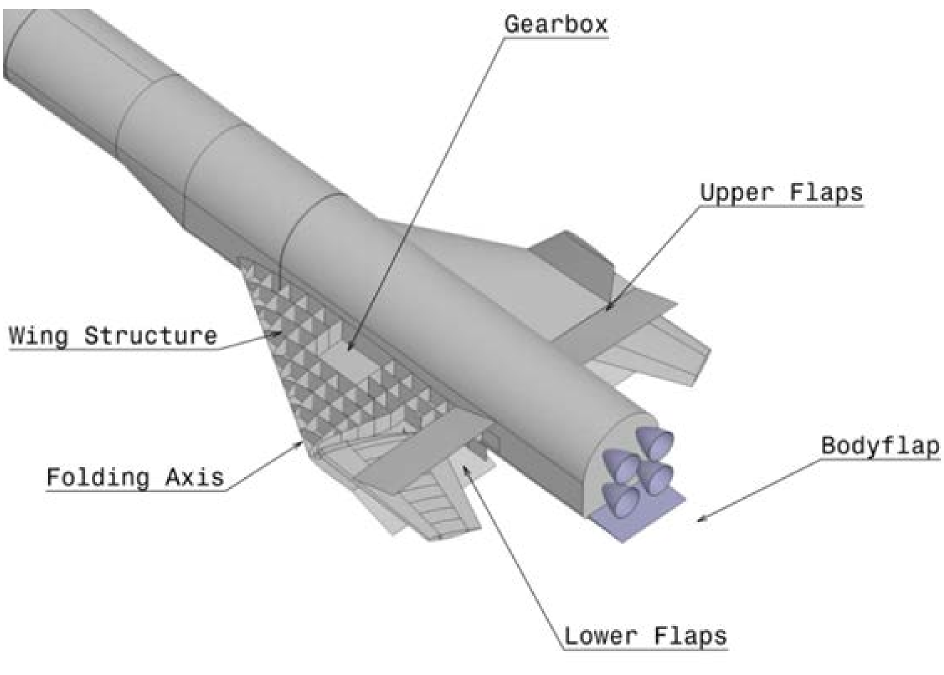

Figure 11: Preliminary structural layout of the variable wing stage RLVC4-III-B

The achievable lift-to-drag ratio of the variable wing configuration is shown in Figure 12. An L/D of slightly above 6.5 for trimmed conditions is a suitable value since it leads to a flight path angle of -8.75° during the In-Air-Capturing maneuver. The maximum trimmed L/D in hypersonics is 2 to 2.5 at an AoA close to 20°. 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 drag to decelerate the vehicle. Hence, the actually flown L/D at re-entry conditions with AoAs of around 40°-50° is around 1.

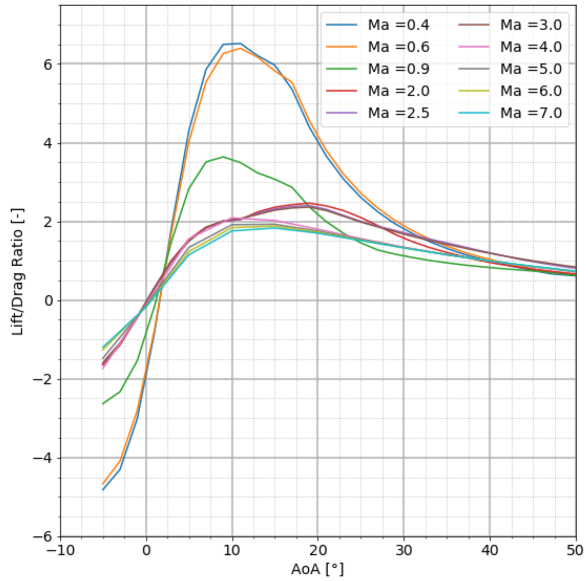

Figure 12: L/D of the variable wing RLVC4-III-B configuration in trimmed flight

In any case trimmability and controllability of the vehicle in all flight conditions is to be assured. Figure 13 shows the pitching moment coefficients for the folding wing stage from Mach 2 upwards. A stable trim point requires cm=0 and $\partial cm/\partial \alpha \leq 0$. Hence, stable trim points above Mach 3 can be found at high AoAs with positive (downward) spoiler deflections between 10° and 20°. Prior to more detailed analyses such deflections are assessed as technically feasible.

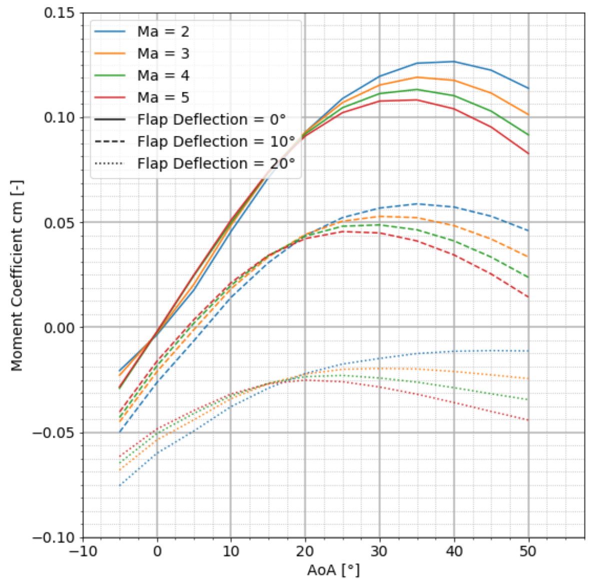

Figure 13: Pitch moment coefficient cm in supersonics for variable wing RLVC4-III-B stage

The mass breakdown of the complete RLVC4-III-B launcher with the variable-wing first stage is listed in Table 8. The reusable first stage dry mass reaches almost 70 tons. A structural index of 20% 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 11 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 ELV-segment is much smaller than the one of the RLVC4-III-A (compare Table 5). The cryogenic main stage loading is further reduced to 150 tons and therefore the layout is close to the early Ariane 5 "G" EPC-stage with common tank bulkhead. Note the lower structural index of the smaller H150 compared to the H170 due to reduced mechanical loads and not fully identical assumptions for the weight estimation. The 3rd stage is assumed to use cryogenic LOX-LH2 and to be equipped with the existing and qualified Vinci engine (see Table 2). The propellant loading is approximately 16 tons. The 20 m length fairing is selected as for the RLVC4-II. Total lift-off weight is approximately 625 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 8: Launcher masses by stage 3STO RLVC4-III-B

H340 (RLV stage)	
Ascent Propellant	340000 kg
Dry Mass	69670 kg
GLOW	417800 kg
Structural Index incl. Engines	20.0%
H150 ELV	
Ascent Propellant	150000 kg
Deorbit Propellant	-
Dry Mass	17310 kg
GLOW including fairing	174210 kg
Structural Index incl. Engine w/o fairing	11.3%
H16 ELV	
Ascent Propellant	15300 kg
Deorbit Propellant	500 kg
Dry Mass	4225 kg
GLOW (incl. P/L)	32800 kg
Structural Index incl. Engine and multiple payload adapter	25.5%
separated payload GTO	12000 kg
Total GLOW	**624825 kg**

Figure 14 shows the ascent trajectory of the complete vehicle into the low transfer orbit. The ascent burns of 1st and 2nd stage propel the 3rd stage and payload to an altitude of roughly 140 km. The 3rd stage coasts along the ballistic trajectory (see section 4.1!) until reaching the equator where it ignites its engine to provide the final Δv required to reach GTO (not shown in Figure 14). RLV stage separation occurs at slightly less than 2 km/s (Mach 6.3) and an altitude of 64.3 km resulting in 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 70 km altitude (Figure 14). After approximately another 5 minutes of acceleration the MECO-conditions of the transfer LEO are achieved.

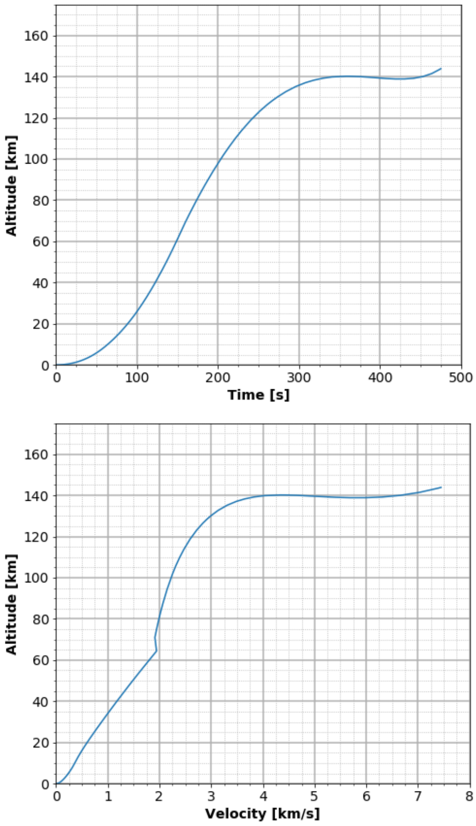

Figure 14: Ascent trajectory of 3STO RLVC4-III-B in transfer orbit 30 km x 600 km

After its MECO and stage separation the winged RLV stage ascents in ballistic flight to an apogee slightly above 100 km (Figure 15). Around 200 seconds after stage separation during the stage's descent a rapid increase in aerodynamic forces and loads can be observed. The re-entry AoA is kept at 45° in the beginning of the atmospheric flight phase before α is rapidly reduced to less than 20° to limit the n_z load factor to a maximum of 3.5 g. The vehicle is controlled in a smooth reentry corridor without extensive skipping by adapting AoA and by banking which also initiates its heading change towards the launch site. The movable outer wing is preliminary 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. After transitioning to subsonic velocity, the stage enters a steady gliding flight with an AoA that provides it the maximum trimmable L/D and flight path angle of around -8° favorable to starting the "in-air-capturing" maneuver approximately 12 minutes after stage separation.

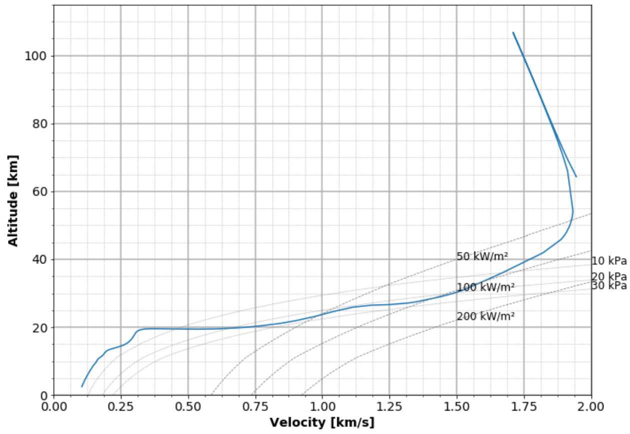

Figure 15: Descent trajectory (altitude vs. flight speed) of winged RLVC4-III-B first stage

The TPS of the system was preliminarily defined according to the calculated thermal loads experienced during re-entry of the GTO-mission. Figure 16 presents a distribution of windward side surface areas distinguished by the maximum external temperature reached during the mission. These areas help in selecting the most suitable TPS-type and usually each of them is designed with a constant insulation thickness in its sector. Depending on the expected temperature, the respective areas are covered with FRSI (Felt Reusable Surface Insulation) in lower temperature zones from 400 K - 600 K maximum surface temperature, AFRSI (advanced flexible reusable surface insulation) for 600 K – 900 K surface temperature, and TABI (tailored advanced blanket insulation) for temperatures from 900 K – 1200 K. The one-dimensional TPS sizing analyses performed for the complete vehicle along the full reentry trajectory intend to provide mission-dependent TPS mass, but not a preliminary functional architecture of this subsystem.

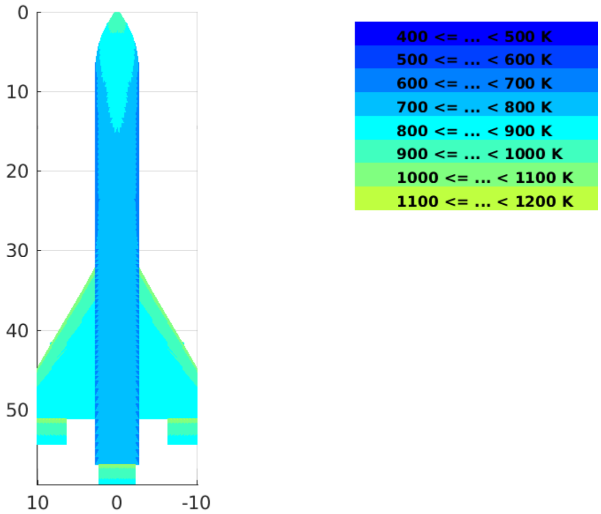

Figure 16: Areas of maximum temperature reached on the windward side of RLVC4-III-B first stage during re-entry

Due to the comparably low re-entry velocity of around Mach 6.7 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 shock-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.

4.4 Type RLVC4-III-C

A similar configuration as the RLVC4-III-B in size, however, with a different, fixed double delta wing geometry has been defined as a potential alternative. Its overall dimensions are listed in Table 9 which are slightly below those of the swept-wing stage. The first stage is equipped with double delta wings with a leading angle of 70° on the inner panel, and 40° on the outboard panel. The transonic airfoil RAE2822 was chosen for both the inner and the outer wing section. The span is selected to achieve sufficient L/D-ratio at IAC-flight Mach number of 0.4. A preliminary sketch of the parallel launcher configuration is presented in Figure 17.

Table 9: Major stage dimensions 3STO RLVC4-III-C

H340 RLV stage	
total length (incl. bodyflap)	56.6 m
fuselage diameter	5.4 m
total span	30.8 m
H150 ELV	as in Table 7
H16 ELV (under fairing)	as in Table 7

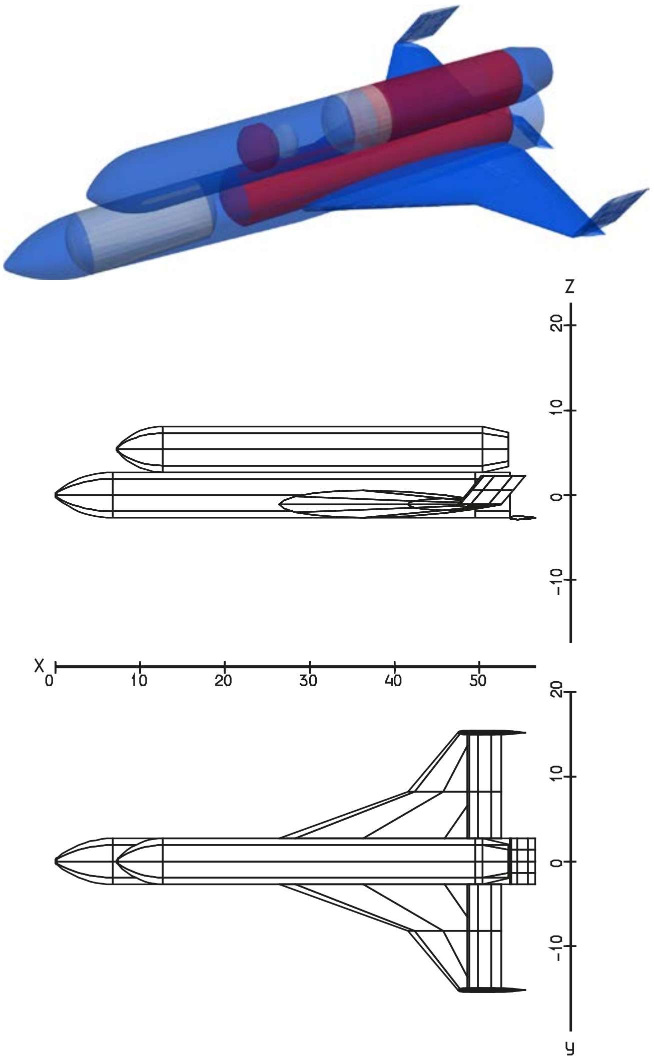

Figure 17: Sketches of RLVC4-III-C reusable first stage (top) and parallel launch configuration (bottom)

The RLVC4-III-C-configuration might achieve L/D-ratios around 6 in subsonic flight, however, requiring wing and body flaps of significant size as visible in Figure 17. The actual feasibility of such devices is to be critically assessed in the next investigation steps. The dry mass of the RLV is estimated at 66.7 Mg below that of the RLVC4-III-B with swiveling wings (Table 10). The 2nd and 3rd stage masses are the same as those from RLVC4-III-B.

Table 10: Launcher masses by stage 3STO RLVC4-III-C

H340 (RLV stage)	
Ascent Propellant	340000 kg
Dry Mass	66675 kg
GLOW	413500 kg
Structural Index incl. Engines	19.2%
H150 ELV	174210 kg
H16 ELV	
Ascent Propellant	15225 kg
Deorbit Propellant	500 kg
Dry Mass	4220 kg
GLOW (incl. P/L)	32650 kg
Structural Index incl. Engine and multiple payload adapter	25.6%
separated payload GTO	11940 kg
Total GLOW	**620385 kg**

The descent maneuver of the RLV is performed in four steps. After separation and ascending ballistically to an apogee of 120 km, the vehicle reenters the atmosphere with 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. 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 stagnation point heatflux and altitude over Mach number during reentry are shown in Figure 18. The RLVC4-III-C experiences a moderate peak heatflux at the nose of 170 kW/m² at an altitude of 30 km and corresponding Mach number of 5. These data are significantly below those shown in Figure 6 for the much larger H750 with higher staging Mach number.

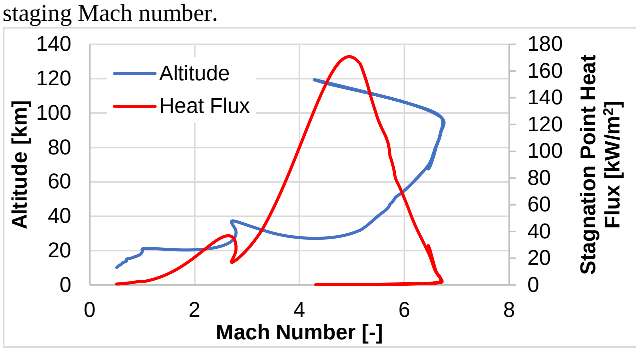

Figure 18: H340 descent trajectory and heatflux

5 DATA COMPARISON AND COST ASSESSMENT

The heavy lift, partially reusable launch configurations in parallel arrangement are able to deliver significant payload mass in the high-energy GTO. The calculated range spans from 11.9 to 24.8 tons separated satellite mass. This corresponds to payload ratios between 1.24% for the TSTO and 2.15% for the largest 3STO as shown in Figure 19. The 3-stage "HHH"-launchers improve payload ratio between 55% and 73% compared to the "HH"-TSTO which in case of –IIIB and -IIIC is equivalent to a much smaller launcher GLOW for similar payload class.

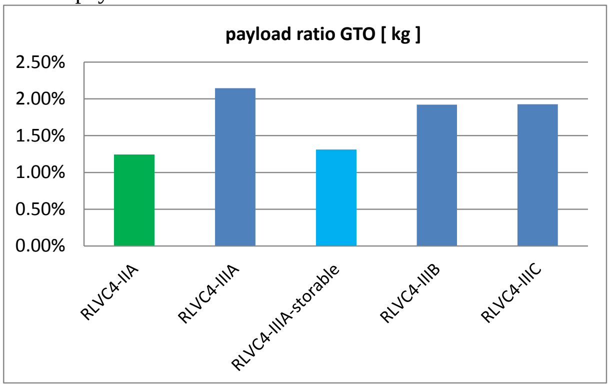

Figure 19: Payload ratios for GTO mission of the different RLVC4-configurations

A significantly smaller launcher GLOW is related to smaller dry weight and might offer lower launch costs. However, a 3STO requires an additional upper stage adding complexity and cost. Therefore, a comparison of payload ratios alone is not sufficient in finding the most attractive configuration.

Both, the development (NRC) and operational costs (RC) have been preliminarily estimated for different RLVC4-configurations using TRANSCOST-derived relationships for stages and engines. The vehicle development costs are driven mainly by the RLV, the reusable SLME, and the expendable stages. A relative comparison of the NRC is presented in Figure 20. The largest and most powerful configuration RLVC4-IIIA is used as reference and set to 100%. The overall similar TSTO RLVC4-IIA without additional cryogenic upper stage saves about 4% in NRC while the smaller RLVC4-IIIB with H16 upper stage saves almost 12%. The potential reduction in development costs by the smaller or less complex systems is less substantial than might be expected on a first look. However, it is to be acknowledged that the main propulsion system SLME is identical in all configurations and costs growth has a regressive correlation with stage size.

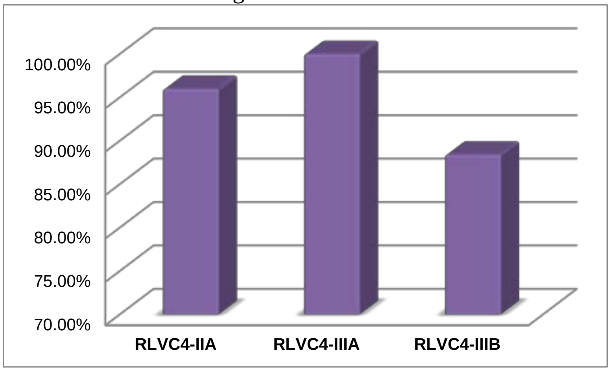

Figure 20: Estimated relative development costs of different RLVC4-configurations

Achieving a major reduction in today's launch costs to orbit is the key motivation of introducing reusability in space transportation. A comparison of the recurring launch costs (RC) is probably even more important than the evaluation of the different development costs. As a similar engine is used on the RLV and on the main ELV-stage, it has been reasonably assumed that each SLME is used 7 times (4 in case of –IIIB) on the RLV-booster before being refurbished with a larger expansion nozzle and flown on its final expendable mission. Further, a yearly rate of 15 missions has been presumed. A relative comparison of total launch costs is shown in Figure 21, again normalized to the data of the RLVC4-IIIA. It is interesting to see that the TSTO RLVC4-IIA promises slightly lower costs than the much smaller RLVC4-IIIB. This result is not to be expected for an ELV, but in case of an RLV-booster the larger number of engines is linked to their increased reusability rate and the relinquishment of the 3rd stage has non-negligible effect on launch cost savings.

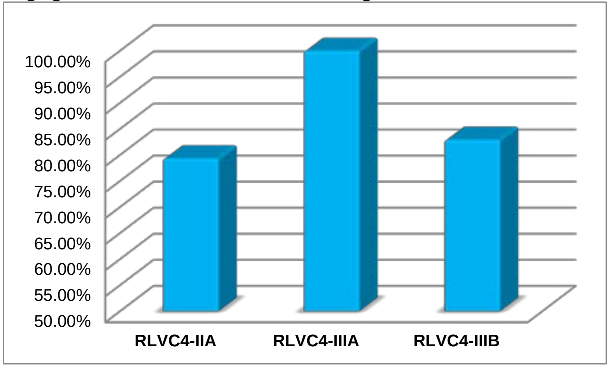

Figure 21: Estimated relative launch costs of different RLVC4-configurations

Finally, an assessment in specific transportation cost (e.g. €/kg) to GTO is depicted in Figure 22. All three concepts presented are capable of achieving major reduction in specific launch cost compared to the latest generation of expendable vehicles. The very high performance RLVC4-IIIA shows a clear (theoretical) advantage and its reference costs are exceeded by more than 70% for the smaller 3STO with H340 RLV. 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.

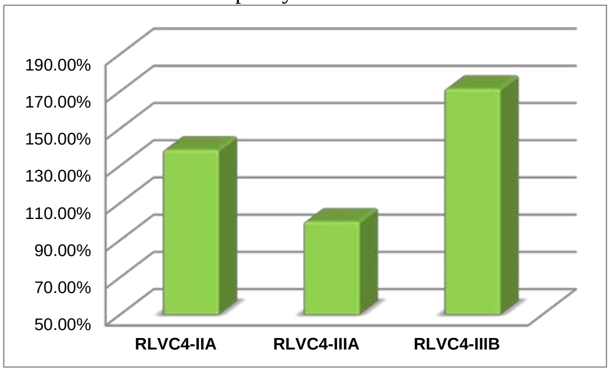

Figure 22: Estimated relative specific launch costs of different RLVC4-configurations

6 CONCLUSION

Several partially reusable launcher concepts have been investigated in TSTO- and 3STO-configurations for heavy-lift GTO-missions to be launched from Kourou's CSG. 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 between 11.9 tons and 24.8 tons is achievable in GTO when using a parallel arrangement of RLV and ELV stages. The 3-stage concepts bring attractive payload ratios of up to >2% with all stages implementing hydrogen. Even the reduced-size RLV H340 delivers multiple payloads with masses significantly beyond the capabilities of A64 into GTO while GLOW remains considerably lower.

The attractive technical design of the studied concepts has been subjected to a first row of launch cost estimations. A reduction of >50% in specific RC (wrt. expected A6 RC) seems to be feasible. The performed estimations show a relative benefit of TSTO in RC compared to 3STO in the reference GTO-mission. The impact on alternative missions is to be assessed in the future.

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

7 ACKNOWLEDGEMENTS

Part of this work (booster definition of RLVC4-III-B) was performed within the project '**F**ormation flight for in-**A**ir **L**auncher 1st stage **C**apturing dem**on**stration' (**FALCon**) addressing development and testing of the "in-air-capturing" technology. FALCon, coordinated by DLR-SART, is supported by the EU within the Horizon2020 Programme 5.iii. *Leadership in Enabling and Industrial Technologies – Space* with EC grant 821953. Further information on FALCon can be found at http://www.FALCon-iac.eu

8 REFERENCES

1. Chaize, M.; Collange, G.; Chavy, S.; Debas, G.; Bonguet, P.: ARIANE 6 LAUNCHER SYSTEM DEVELOP-MENT STATUS, IAC-18-D2.1.2, 69th International Astronautical Congress (IAC), Bremen, Germany, 1-5 October 2018
2. Patureau de Mirand, A.; Bahu, J.M.; Louaas, E.: Ariane Next, a vision for a reusable cost efficient European rocket, 8TH EUROPEAN CONFERENCE FOR AERONAUTICS AND SPACE SCIENCES (EUCASS), Madrid 2019
3. Wilken, J.; Stappert, S.; Bussler, L.; Sippel, M.; Dumont, E.: Future European Reusable Booster Stages: Evaluation of VTHL and VTVL Return Methods, IAC-18-D2.4.1, 69th International Astronautical Congress, Bremen 2018
4. Stappert, S.; Wilken, J.; Bussler, L.; Sippel, M; Karl, S.; Klevanski, J.; Hantz, C.; Briese, L.E.; Schnepper, K.: European Next Reusable Ariane (ENTRAIN): A Multidisciplinary Study on a VTVL and a VTHL Booster Stage, IAC-19-D2.4.2, 70th International Astronautical Congress (IAC), Washington D.C., United States, 21-25 October 2019
5. Sippel, M., Bussler, L., Krause, S., Cain, S.: Bringing Highly Efficient RLV-Return Mode „In-Air-Capturing“ to Reality, HiSST 2018-1580867, 1st HiSST, Moscow 2018
6. Stappert, S.; Wilken, J.; Bussler, L; Sippel, M.: A Systematic Comparison of Reusable First Stage Return Options, 8th EUROPEAN CONFERENCE FOR AERONAUTICS AND SPACE SCIENCES (EUCASS), Madrid 2019
7. Sippel, M., Stappert, S., Bussler, L., Dumont, E.: Systematic Assessment of Reusable First-Stage Return Options, IAC-17-D2.4.4, 2017, https://elib.dlr.de/114960/1/IAC17-D2.4.4.pdf
8. Sippel, M.; Stappert, S.; Bussler, L; Krause, S.; Cain, S.; Espuch, J.; Buckingham, S.; Penev, V.: Highly Efficient RLV-Return Mode "In-Air-Capturing" Progressing by Preparation of Subscale Flight Tests, 8th EUROPEAN CONFERENCE FOR AERONAUTICS AND SPACE SCIENCES (EUCASS) , Madrid July 2019
9. Sippel, M., Wilken J.: Preliminary Component Definition of Reusable Staged-Combustion Rocket Engine, Space Propulsion 2018, Seville, May 2018
10. Haeseler, D.; Wigger, F.; Fortier, Th.; Humbert, E.; DeKorver, V.: Vinci® Upper Stage Engine Development, Test, Qualification, and Industrialisation Status for Ariane 6, IAC-18-C4.1.4, 69th International Astronautical Congress, Bremen 2018
11. Patentschrift (patent specification) DE 101 47 144 C1, Verfahren zum Bergen einer Stufe eines mehrstufigen Raumtransportsystems, released 2003
12. Sippel, M.; Klevanski, J.; Kauffmann, J.: Innovative Method for Return to the Launch Site of Reusable Winged Stages, IAF-01-V.3.08, 2001
13. Sippel, M., Klevanski, J.: Progresses in Simulating the Advanced In-Air-Capturing Method, 5th International Conference on Launcher Technology, Missions, Control and Avionics, S15.2, Madrid, November 2003
14. Sippel, M.; Stappert, S.; Wilken, J. ; Darkow, N.; Cain, S ; Krause, S.; Reimer, T.; Rauh, C. ; Stefaniak, D.; Beerhorst, M.; Thiele, T.; Briese, L.E.; Schnepper, K.; Riccius, J.: Focused research on RLV-technologies: the DLR project AKIRA, 8th EUROPEAN CONFERENCE FOR AERONAUTICS AND SPACE SCIENCES (EUCASS), Madrid July 2019
15. Sippel, M.; Manfletti, C.; Burkhardt, H.: Long-Term / Strategic Scenario for Reusable Booster Stages, Acta Astronautica 58 (2006) 209 – 221
16. Sippel, M.; Stappert, S.; Bussler, L.; Singh, S.; Krummen, S.: Ultra-Fast Passenger Transport Options Enabled by Reusable Launch Vehicles, 1st FAR conference, Monopoli, September 30th – October, 3rd 2019
17. Sippel, M.; Stappert, S.; Bussler, L.; Forbes-Spyratos, S.: Technical Progress of Multiple-Mission Reusable Launch Vehicle SpaceLiner, HiSST 2018-1580839, 1st HiSST: International Conference on High-Speed Vehicle Science Technology, Moscow, November 2018
18. Sippel, M.; Stappert, S.; Koch, A.: Assessment of multiple mission reusable launch vehicles, in Journal of Space Safety Engineering 6 (2019) 165–180, https://doi.org/10.1016/j.jsse.2019.09.001
19. Sippel, M., Trivailo, O., Bussler, L., Lipp, S., Kaltenhäuser, S.: Evolution of the SpaceLiner towards a Reusable TSTO-Launcher, IAC-16-D2.4.03, September 2016
20. Sippel, M.; Klevanski, J.; Atanassov, U.: Search for Technically Viable Options to Improve RLV by Variable Wings, IAC-04-V.8.07, 2004

Further updated information concerning the SART space transportation concepts is available at:
http://www.dlr.de/SART