A viable and sustainable European path into space – for cargo and astronauts

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A think tank of DLR space-research institutes and ArianeGroup reflected on the launcher perspectives, broadening the mission capabilities for European access to space including human mission towards LEO, supporting potential future commercial markets. The Ariane 6 launcher is to be evolved with perspective of enhanced versatility and reduced costs. Also, a new generation of modern, high-performance launchers is to be prepared that is capable of serving all major missions relevant to Europe.

The paper summarizes major results of preliminary technical design studies performed in Germany, both for a future Ariane 6 evolution including astronaut transport to and from LEO and for the next generation of partially reusable concepts. Return of the RLV is pondering options of powered descent and vertical landing and smart winged return technologies with horizontal landing close to the launch site. The main propulsion system is considering LOX-LH2 in different cycle architectures as well as LOX-LCH4 of the ongoing PROMETHEUS development program.

For the studied concepts, the overall shape and aerodynamic configuration, the propulsion system, the architecture and lay-out of the stages are described and different technical solutions of RLV concepts are compared.

Keywords: Ariane 6, Apollo capsule, RLV, LOX-LH2-propulsion, LOX-LCH4-propulsion, VTHL, VTVL, in-air-capturing

Nomenclature

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

Subscripts, Abbreviations

3STO	Three-Stage-To-Orbit
AEDB	Aerodynamic Database
ALM	Additive Layer Manufacturing
AOA	Angle of Attack
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
ISS	International Space Station
LAS	Launch Abort System
LCH4	Liquid Methane
LEO	Low Earth Orbit
LFBB	Liquid Fly-Back Booster
LH2	Liquid Hydrogen
LLPM	Lower Liquid Propulsion Module (of Ariane 6)
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
ULPM	Upper Liquid Propulsion Module (of Ariane 6)
VTHL	Vertical Take-off and Horizontal Landing
VTVL	Vertical Take-off and Vertical Landing
CoG	center of gravity
cop	center of pressure

1 INTRODUCTION

The recent rapid development of advanced technologies in the field of international space transportation systems requires an updated and refined European strategy based on two pillars, considering short-term as well as long-term aspects:

1. The reliable and powerful Ariane launcher system is to be systematically further improved with perspective of reducing costs while improving versatility. Within the next few years, an expansion of the mission spectrum including astronautic transport and the possibility of bundling missions should be planned.

2. A new generation of modern, high-performance launchers is to be prepared that is capable of serving all major missions relevant to Europe - in a highly cost-efficient, sustainable, flexible and performance-oriented manner.

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

It should be noted that the launcher concept options described here have not been part of ESA's recent study *New European Space Transportation Solutions* (NESTS) [2]. Nevertheless, it is not completely surprising that some similarities in the proposed concepts can be identified. The future space transportation options presented in this paper are similar as well as complementary to other European studies.

2 EVOLUTION OF THE ARIANE 6 LAUNCHER SYSTEM

Europe's Ariane 6 developments are progressing [1]. Both configurations under development, A62 with two solid strap-on boosters and A64 with four solid strap-on boosters, are scheduled to be fully operational in 2023.

Different improvement options of the Ariane 6 launcher are evaluated to enhance its mission versatility and reduction of specific transportation cost. Two of these potential future evolution options are described in this section.

2.1 Option of mission enhancement to astronautical transport

Astronautical space transportation is a prerequisite for the construction and operation of near-Earth space stations, and for the future commercialization of space travel. In order to assess the current technological expertise and missing elements in Europe, DLR and ArianeGroup performed the EURASTROS (European Astronautical Space Transportation) study [23]. The capsule, service module, launch abort system and upgrade of the ground service infrastructure have been identified as main development items for the astronautical space transportation missions.

Europe, in particular Germany, has already gained some practical experiences by means of its partnership in several astronautical space missions like the D1/D2 Spacelab with Space Shuttle or those to the space stations Mir and ISS. The development, design and manufacturing of the service module ESM for the Orion capsule of NASA is a further example of competence.

2.1.1 Mission assumptions and design requirements

The following mission and key design requirements have been defined for the European Astronautical Transportation System (EATS):

- Major prerequisite for all EURASTROS activities is aiming for low-cost development and missions by maximum re-use of developed hard- and software, minimum investment for ground- as well as operational infrastructure and for an optimized demonstration and qualification logic.
- A short development time is intended and the EATS shall have its first mission within 5 to 7 years after full development started.
- The EATS shall be able to take 2 (in option 3) astronauts to LEO and back to Earth.
- The EATS shall be based on the Ariane 6 launcher. It shall complement the existing A6 launch system, limiting its modifications as far as possible.
- The EATS shall provide continuous autonomous launch abort capability from lift-off through orbital insertion in the event of a loss of thrust or loss of attitude control.
- The EATS shall use a new deceleration/landing system, which allows keeping the maximum loads of the ground impact within human tolerable limits. The landing shall be performed on water.

2.1.2 Launcher architecture

The A64-version with four strap-on boosters has been selected as baseline for an astronaut carrier to provide sufficient performance margin for all intended missions (see following section 2.1.6).

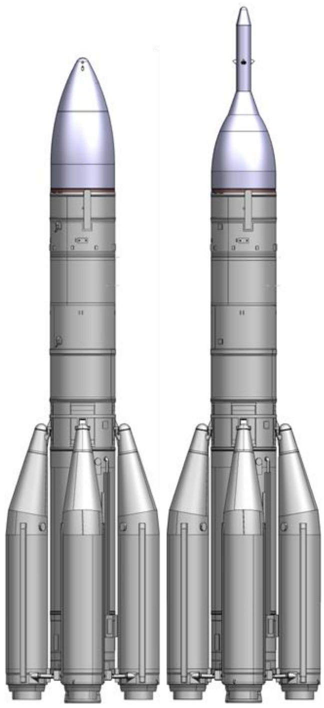

Figure 1: Sketch of A64 variants with forward sections adapted for human spaceflight

The architecture intends to keep all existing main stages, the P120C, the LLPM and the ULPM modules (see [1]) untouched as far as possible. The main difference is related to the forward payload section which is to be replaced by astronaut capsule, service module and emergency escape system protected by a new faring.

Two different design options for the integration of the launch abort system (LAS) are under assessment (see section 2.1.5). The concept with solid motors integrated beneath the fairing is shown in Figure 1 (left) and a classical escape tower in top position in Figure 1 (right).

The total length of both A64 variants suitable for manned spaceflight is not exceeding the length of A62 with short faring.

2.1.3 Capsule's shape, aerothermodynamics and structural lay-out

The Crew/Command module or capsule has to accommodate at least 2 astronauts with the option of a third one. A docking capability with ISS and a safe reentry in the nominal mission or after launch abort (controlled and uncontrolled/ballistic) is intended. The maximum duration in autonomous flight operation of the capsule should be 4 days.

The shape of the astronaut capsule is derived from Apollo for which already European experience exists with the ARD capsule reentry experiment launched on Ariane 503 in 1998. It has a spherical nose and a truncated conical back shell. To accommodate at least two astronauts, an outer diameter of 3.5 m is chosen which results in a volume of 11.6 m³ or about 72% of the Apollo capsule volume. A third astronaut can be accommodated as an option. The outer dimensions of the capsule are presented in Figure 2.

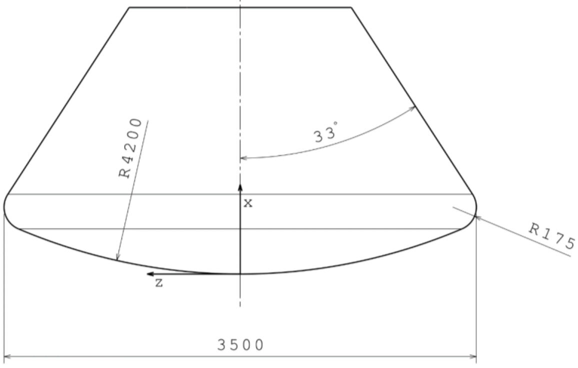

Figure 2: Preliminary shape of capsule with its major dimensions

The capsule's center of gravity is offset from the symmetry axis to obtain trimmed AoA of 25° in hypersonics which delivers an L/D ratio of approx. 0.35. The corresponding reentry trajectories of the capsule have been analyzed for nominal as well as for off-nominal atmospheric flight. Controlled and uncontrolled/ballistic trajectories are considered for return from ISS but also for cases of launch abort. The aerothermal and mechanical loads have been analyzed with approximative techniques and high-resolution aerothermal simulations. The points of maximum heat flux and maximum dynamic pressure were analyzed in detail to provide a basis for the TPS and structural design. An example of the pressure distribution at the point of maximum dynamic pressure for a case of ballistic reentry after launch abort at high altitude is shown in Figure 3.

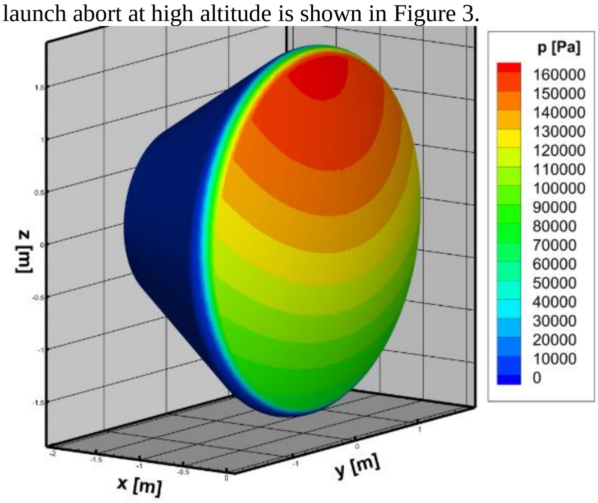

Figure 3: Calculated pressure distribution of capsule in case of ballistic reentry after launch abort at M= 8.6, 27.5 km, α= 25°

The pressurized compartment of the capsule is designed as a 2.6 m diameter cylinder-cone section with outer orthogrid stiffening. The pressure vessel is designed for an internal pressure of 1 bar. To protect the structure from the aerothermal loads, the front and back shells are equipped with classic ablator type TPS. The dry mass of the capsule is estimated slightly above 5600 kg incl. ISS docking adapter.

2.1.4 Service module (SM)

Following the standard design approach of capsule systems, a separate service module will be used for

- providing structural continuity between the launcher and the Crew Module;
- providing thrust after upper stage/launch vehicle separation for propulsive orbital transfer maneuvers;
- generating electrical power;
- regulating the heat for the life support and avionics equipment;
- storage of potable water, oxygen, and nitrogen and deliver them to the Crew Module

The space-storable propellant combination N_2O_4/MON-MMH is selected compatible with the in-space operational and the engine reignition requirements. A dedicated development will be necessary but can be based on the extensive heritage at ArianeGroup Bremen with Ariane 5 EPS-stage, the ATV and the ESM service module of Orion. The ASTRIS kick-stage for Ariane 6 currently under development should serve as the baseline being modified in propellant loading (1315 kg for Δv 400 m/s) and thrust, further adapted to increased redundancy and safety requirements of manned missions. Four new pressure-fed BERTA engines (see references 3 and 4 on already performed pre-development work) are selected as the baseline and should provide a total thrust of 16 kN.

2.1.5 Dimensioning of emergency escape system

In the EURASTROS study on A6-based human spaceflight, abort safety is understood as the full chain from detecting critically dangerous conditions to the escape maneuver execution. An end-to-end abort capability during the full launcher ascent mission is envisaged.

An essential safety element for evolving Ariane 6 to a launcher capable of bringing astronauts to orbit is the integration of a launch abort system (LAS). Several design options exist for such a system to be activated in case of extreme emergencies. The classical "Tower"-concept has been used with many launchers like Saturn V – Apollo or Soyuz and is again selected for the Orion. A possible adaptation of this type to Ariane 6 is shown in Figure 4 at left. An alternative design with the abort motors attached to the side of the capsule is realized with SpaceX Crew Dragon and Boeing CST-100. In Figure 4 at right a design is visible with circumferential arrangement of eight separation motors within the boundaries of the external fairing.

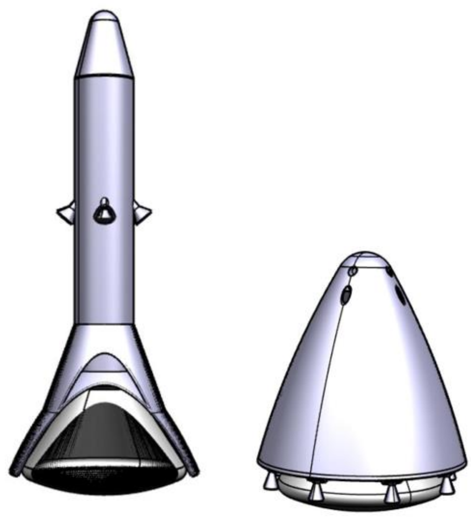

Figure 4: Potential capsule integration in launcher nose section with two options for LAS

Currently, only solid motors are considered for a potential future Ariane 6 LAS. Several aspects are preliminarily evaluated as the external aerodynamic shape of the launcher, integration, the overall mass impact as well as flight stability issues.

A pre-dimensioning of the escape system is performed using a simplified point-mass-model and aerodynamic forces calculated from the coefficients of dedicated computed AEDB. Simulations are run for both LAS-concepts of Figure 4 and for three critical emergency situations:

- **L**aunch **P**ad **E**scape (‘**LPE**’)
- maximum dynamic pressure (‘**maxQ’**)
- maximum launcher acceleration (‘**maxA’**)

In all these situations, the LAS is dimensioned against the requirements to establish within 3 seconds a safety distance of 200 m between the demising launcher and the capsule This distance is measured relative to the nominal LV trajectory in case of the ‘maxQ’ and ‘maxA’ deemed accurate enough for preliminary LAS-dimensioning. Requirements for the LPE-case are a minimum apogee height and a minimum downrange to reach a suitable off-shore capsule landing area. Additional influence factors by wind, or deployed parachute have not been considered in these early analyses.

For all three cases, AoA and pitching rates have been systematically varied while searching minimum thrust to meet the corresponding escape requirements. Results show that the LPE-scenario is a sizing case with regards to the minimum required LAS-thrust level, which was found to be 950 kN for both concepts. The other scenarios were less critical with regards to thrust, but identified as potentially sizing with regards to experienced accelerations, with increased lateral accelerations (n_Z) observed in ‘maxQ’ and the highest axial accelerations (n_X) observed in the ‘maxA’-cases. The derived minimum-thrust-escape-trajectories for the LPE-case are illustrated in Figure 5.

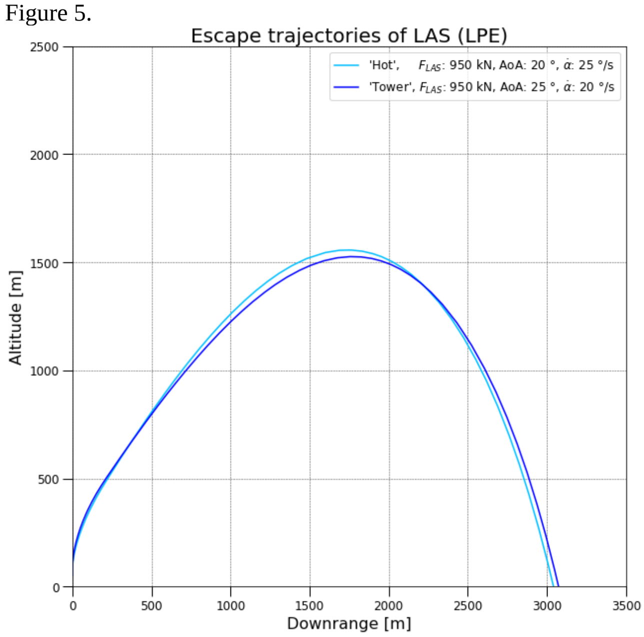

Figure 5: LPE trajectories for both LAS concepts

2.1.6 Launcher performance and feasibility

Independent of actual future missions, the transportation of crew to an ISS-transfer orbit is investigated as benchmark for an astronautical mission.

Different ascent options for both A62 and A64 to circular orbits of various altitudes have been preliminarily evaluated considering already the adapted fairing and emergency escape system. It is quickly revealed that the performance for both versions would benefit from a significant reduction of the ULPM nominal fuel loading. But despite an optimized loading of the upper stage, the performance of the A62 would still be insufficient for a viable and robust astronaut transport system.

However, the more powerful version Ariane 64 has sufficient payload capabilities, even without the need to operate the upper stage with reduced fuel loading. Though, optimizing fuel loading would further yield performance margins of around 2 t if found necessary. Figure 6 shows the calculated ascent profiles of A64 for the ISS-mission and different target orbit altitudes. The relatively low thrust of the ULPM’s Vinci engine (see Table 4 on page 8) requires the upper stage to initially climb to altitudes above the target orbit altitude. Calculated payload performance of all investigated cases is well above 18 t demonstrating the principal feasibility of using the A64 version for such a new mission type.

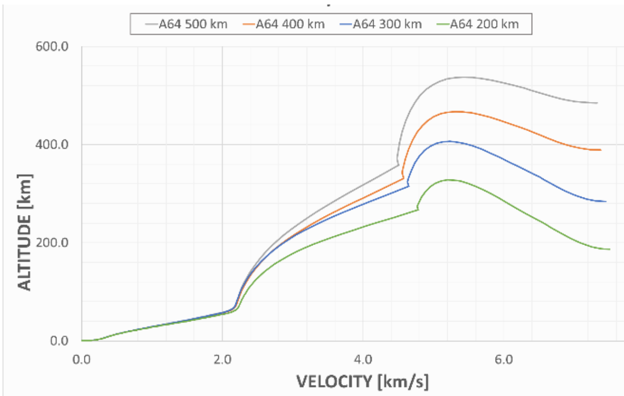

Figure 6: Altitude-velocity profiles of A64 ascent for different orbit altitudes in astronaut transport mission

Ground safety requirements of the expended P120C and LLPM are found not critical for any of the investigated trajectory options. A first simplified assessment of the LLPM ballistic reentry, disregarding analyses of stage break-up or tumbling, estimates the ground impact point more than 500 km south of the Azores and more than 700 km west of the Canary Islands. This ground safety margins should be sufficient for this early analysis and is to be confirmed by more detailed future studies. The upper stage is assumed to be actively deorbited after completing its primary mission.

2.1.7 Preliminary analyses of exo-atmospheric abort modes

At larger altitudes, i.e., above approx. 90 km the LAS is jettisoned and the service module supports the abort operation. In case of serious malfunction, the main engine on either LLPM or ULPM (depending on mission phase) is cut off and the SM separates from the launcher and ignites its engines to carry the capsule to a safe distance. After safe reentry conditions are reached, the command and service module separate and the capsule reenters the atmosphere to safely bring the astronauts back to the ground.

However, several requirements are to be fulfilled to guarantee a successful rescue along the entire launcher ascent. At any time, an abort should be made possible without exceeding human tolerable g loads. The launch profile of the A64 requires acceleration maneuvers with the SM engines at some sections along the launch trajectory to meet this criterion. Otherwise, without such maneuvers, launch abort from relatively high altitudes could lead to a steep reentry, and hence, excessively high g-loads, in case of ballistic descent. Hence, sufficient thrust of the SM is essential to allow decreasing the reentry angle.

An abort-to-orbit is the most promising option in the late phase of ULPM propulsion. Then the service module has the capability of raising perigee to a low emergency orbit, allowing for circling the Earth and preparing for more favorable controlled reentry conditions.

In earlier flight phases, in addition, the capsule must avoid emergency descent over densely populated areas like central Europe (e.g. in case of ISS-mission) and should splash down on water for all abort scenarios. Due to the thrust limitations of the Vinci engine, the ULPM acceleration is relatively low and the launcher covers a long distance before orbit insertion. Then, however, after turning the stack, the SM engines can be used for a braking maneuver to ensure a splashdown of the capsule in the Atlantic.

Any exact definition of the exo-atmospheric abort modes requires careful considerations of the launch trajectory options, the SM thrust level and propellant loading, and the reentry performance of the capsule. With the right balance, a successful abort is possible from launch pad to orbit, ensuring a safe transportation of astronauts to space.

2.2 Option of A6 evolution to liquid strap-on boosters and partial reusability

Another option of future Ariane 6 evolvement could be the replacement of the P120C SRM with potential new liquid booster "strap-ons" using LOX/LCH4 or LNG with PROMETHEUS engine. Such a launcher concept might operate the side boosters in expendable mode for high-performance missions or in RLV modes DRL or RTLS in medium / low performance missions.

2.2.1 Booster propulsion system

PROMETHEUS is the precursor of a new European large-scale (100-tons class) liquid rocket engine designed for low-cost, flexibility and reusability [5]. 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 [5]. Maximum capabilities of existing equipment for ALM have been used for the production of large casings, pump impellers and turbines [5].

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. Calculated data for realistic performance of a full-scale engine have been used (compare [5, 8, 11] for similar gas-generator type but slightly different assumptions on nozzle expansion).

2.2.2 Launcher architecture

The architecture intends to keep the existing liquid propulsion main stages LLPM and the ULPM including the interstage structure untouched as far as possible. The P120 C side booster of the Ariane 6, however, are replaced by new liquid boosters. This requires the same axial position for the booster attachment points at the H150's intertank and aft-skirt structure as for the Ariane 6 (see Figure 7 at right). The feasibility assessment presented here targets the maximum LOX/LCH4 propellant loading under tight geometry constraints. The liquid booster diameter is slightly increased from 3.4 m of P120C to 3.6 m instead. Both propellants are stored aft of the forward attachment ring in an integral tank with common bulkhead. Three PROMETHEUS engines are placed in linear instead of triangle arrangement in the booster's base. This design enables to put one of the engines in a center position simplifying the vertical landing with single operational engine in case of RLV-mode. The approximate nozzle exit diameter of 1.1 m is still compatible with this placement and the booster diameter.

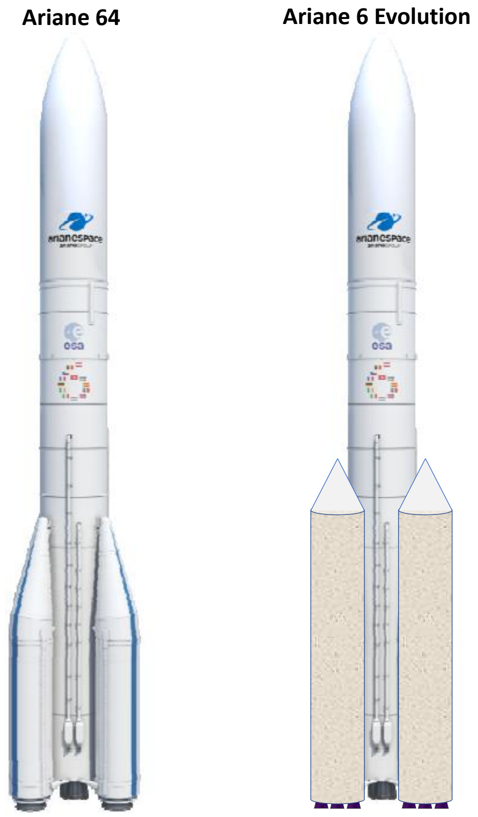

Figure 7: Sketch of Ariane 6 Evolution with four liquid strap-on boosters compared to A64 (left)

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

2.2.3 Performance in expendable operation mode

In expendable mode all four C-130 side boosters use their full amount of fuel up to MECO, except for residuals. The central LLPM and ULPM deliver in succession the payload into the target 250 km x 35786 km GTO. The separated payload mass is calculated to reach 13 tons. This is a significant performance improvement of roughly 2.5 tons compared to a standard A64 with double launch explained by the increased total impulse of the liquid boosters with roughly 50 s better Isp-performance compared to solids.

2.2.4 Performance in reusable boosters' operation mode

The payload performance in case of DRL with four booster landing on a barge in the Atlantic Ocean is 10.2 tons (-23%) and the respective payload for RTLS is 6.1 t (-53%) (compare Table 1) This decline in payload performance is remarkably lower than with the SpaceX Falcon 9 [6]. The reason is simply the different architecture of the Ariane 6, being a 2.5 stage system where the boosters burn simultaneously with the first stage and separate earlier than the first stage of the Falcon 9 TSTO. Thus, performance degrading of a potential Ariane 6 Evolution with RLV-boosters is lower because it would be a somehow "less reusable" system than Falcon 9.

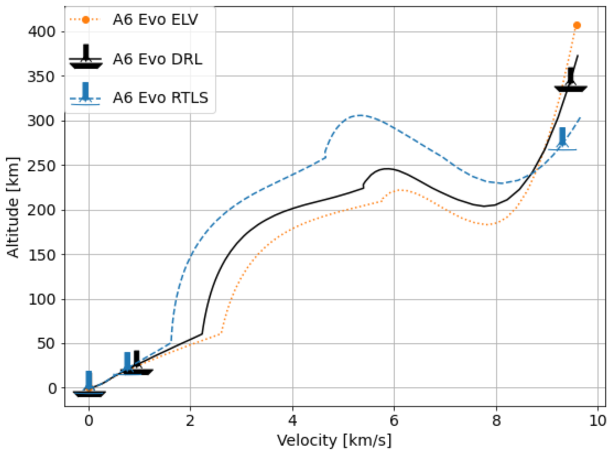

Figure 8: Ascent trajectories GTO mission for Ariane 6 Evolution with C-130 strap-on boosters as ELV and as RLV in DRL or RTLS-mode

Table 1: Payload performance for different potential variants of the Ariane 6 Evolution with C-130 booster

Operational Mode	Payload (GTO) [Mg]	Descent Propellant of single booster [Mg]
A6 Evolution ELV	**13**	-
A6 Evolution C-130 RLV-mode **DRL**	**10.2**	6
A6 Evolution C-130 RLV-mode **RTLS**	**6.1**	21

2.2.5 Reusable booster reentry and return

The A6 Evolution in RLV-mode, has to land up to four side boosters at the designated landing site. Regarding the simulated trajectory of only one exemplary booster would be sufficient in this early analysis since all are following very similar trajectories from performance perspective. The intended capability to reignite the PROMETHEUS engine in flight allows for vertical landing of the reusable C-130 either downrange on a landing barge (comparable to SpaceX' droneships) or back on the launch site (RTLS).

In case of downrange landings, two post-MECO maneuvers are necessary: a re-entry burn which lowers the re-entry loads to acceptable limits, and the landing burn which slows the stage down to landing velocity (see following section 3.4.2). RTLS requires an additional burn shortly after MECO which reverses the direction the booster is flying into. Thus, this burn is commonly labelled "boostback burn". All propulsive return maneuvers of the C-130 are flown with only one engine operative.

Comparing the ascent and re-entry trajectories of the ELV and RLV version, some differences have to be highlighted. For the ascent part of the mission, the RLV versions are flying much steeper, especially in the case of RTLS missions (compare Figure 8). This is favorable in terms of propellant use for re-entry maneuvering but increases the gravitational losses during ascent which also contributes slightly to the payload performance loss related to ELV.

As expected, the DRL version booster is much faster at MECO than the respective RTLS booster (2250 m/s vs. 1600 m/s). Hence, re-entry loads, such as heat flux and dynamic pressure, are higher for the DRL booster (compare Figure 9). Nevertheless, those loads are well within the loads that a Falcon 9 downrange landing stage experiences according to DLR simulations [6].

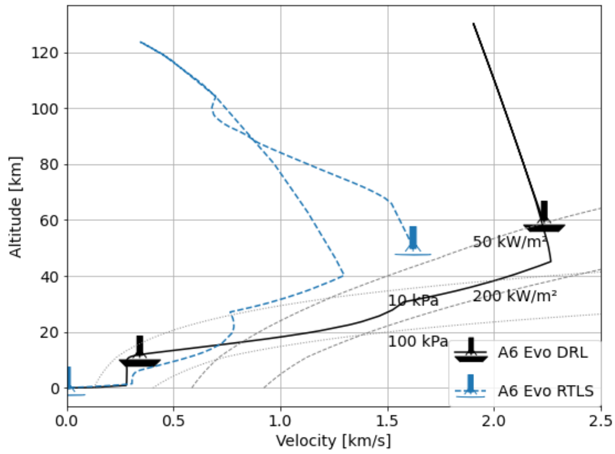

Figure 9: Re-entry trajectories of reusable C-130 strap-on boosters for DRL and RTLS (GTO mission)

One critical point linked to an Ariane 6 Evolution is the fact that for the partially reusable launches it is necessary to land up to 4 boosters almost simultaneously. This means that 4 barges or 4 landing sites have to be operated, including the infrastructure to handle 4 returning stages. This increased demand of infrastructure compared with single returning stages is to be considered when evaluating the cost improvement capability of Ariane 6 evolution options.

Another point worth mentioning is the relatively high thrust-to-weight ratio of the proposed configuration at landing. For the C-130 booster, this is found not lower than 1.8 with the PROMETHEUS engine throttled down to its targeted 30%-minimum thrust level. Though, any such T/W-value in vertical landing requires very accurate engine TVC and GNC performance or might be mitigated by an increased landing weight of the booster. The latter, however, is compromising payload performance.

3 OPTIONS FOR NEXT-GENRERATION RLV-CONFIGURATIONS

Potential opportunities for the technical and mission evolution of Ariane 6 after its current development is completed have been described in the previous section. A somehow more radical 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* [10]. The CNES proposal 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 [10]. 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 [7, 8, 9, 11].

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 [7, 8].

Approaching or even exceeding the payload performance expected for Ariane 6 in GTO or Lunar exploration missions would require extremely tall launcher configurations in case of tandem-staged TSTO with reusable first stage. Therefore, for this class of RLV a parallel stage arrangement is preferable: a winged stage is connected to an expendable upper segment with potentially various internal architectures. References 9 and 11 have demonstrated that a payload range between 12 to 15 tons GTO-class with multiple payload capability can be achieved by a 3-stage architecture while still remaining at relatively compact size. Less demanding missions to different LEO can be served as TSTO. Further refinements of such powerful launcher options with new characteristic data are in focus of this section. Beyond the already investigated winged VTHL-concepts [11] improved now, VTVL options of similar size are studied as a potential alternative.

3.1 Assumptions and Design Requirements

The launcher is to be designed for the most suitable combination of high commonality in major components and providing good mission flexibility. The payload range should be in the 14 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 studies of CNES [10] and DLR [9] 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.

3.2 Mission assumptions

All presented RLV-configurations in this section 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]. All the recent designs presented here have been sized for approximately 14000 kg payload.

3.3 Propulsion systems

Staged combustion cycle rocket engines with a moderate 16 MPa chamber pressure are first choice 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) [18]. 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 2 gives an overview about major SLME engine operation data as obtained by cycle analyses [18] 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 [18].

Table 2: SpaceLiner Main Engine (SLME) technical data [18] 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 [18].

The PROMETHEUS as LOX-LCH4 gas generator type engine (see [5] and previous section 2.2.1) is another suitable propulsion option. Calculated performance data as used for the launcher pre-design described in this section are listed in reference 11.

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 type PROMETHEUS 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 [19]. 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 sections 3.6 and 3.7.

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

3.4 RLV recovery methods considered

Two recovery and return strategies offer attractive conditions for high performance missions. Both are related to a Down-Range "Landing" (DRL).

3.4.1 VTHL 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" [20] 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 [21]. 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 capturing aircraft of adequate size for the to be towed RLV. Used, refurbished and modified airliners should be sufficient for the task.

A schematic of the reusable stage's full operational circle when implementing IAC is shown in Figure 10. 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.

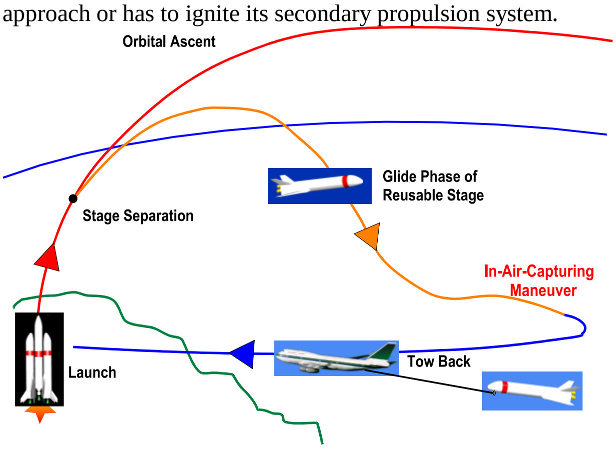

Figure 10: Schematic of the proposed in-air-capturing

Differently, within the in-air-capturing method, the reusable stage is awaited by an adequately equipped large capturing aircraft (most likely fully automatic and probably unmanned), offering sufficient thrust capability to tow a winged launcher stage with restrained lift to drag ratio. Both vehicles have the same heading still on different flight levels. The reusable un-powered stage is approaching the airliner from above with a higher initial velocity and a steeper flight path, actively controlled by aerodynamic braking. The time window to successfully perform the capturing process is dependent on the performed flight strategy of both vehicles, but can be extended for up to more than one minute. The entire maneuver is fully subsonic (around 160 m/s) in an altitude range from around 8000 m to 4000 m [16]. In order to keep the two large vehicles always in a safe distance to each other, the actual contact and towing rope connection is established by a small agile vehicle [17]. After successfully connecting both vehicles, the winged reusable stage is towed by the large carrier aircraft back to the launch site. Close to the airfield, the stage is released, and autonomously glides like a sailplane to Earth.

From a performance perspective, the IAC mode is highly attractive. In a systematic comparison of different RLV-stage return modii [8, 15] 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 [15]. In combination with the performance advantage, the "in-air-capturing"-method based on current analyses seems to be an attractive technology for future RLV.

DLR together with European partners is currently preparing for flight testing the "in-air-capturing"-method on a laboratory scale by using two autonomous test vehicles. The EC-funded project FALCon should bring the TRL of the advanced IAC-recovery method beyond 4 in 2022. The project does not only address the lab-scale but also more sophisticated and refined simulations of a full-scale launcher-capturing. An existing airliner (study reference Airbus A340-600) with typical constraints is regarded for towing and the effects of its generated turbulent wake field are taken into account [16]. Preliminary results from the project are published in [12, 15, 16, 17].

3.4.2 VTVL with down-range sea-landing

Vertical Landing downrange is another viable option for future RLV proposals which has also been considered for the Ariane 6 Evolution option with liquid boosters described in the previous section 2.2. Currently, SpaceX is using this method to land Falcon 9 and Falcon Heavy booster stages on the so-called autonomous spaceport droneships (ASDS) which are positioned downrange of the launch site either in the Atlantic Ocean (Cape Canaveral launches) or in the Pacific Ocean (Vandenberg launches).

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

Compared to VTHL, vertical landing stages are not equipped with conventional wings or rudders and flaps. Instead, landing legs are required and some kind of aerodynamic controls, like grid fins for the Falcon 9, which usually are adding less dry mass as the VTHL recovery hardware. However, VTVL instead require a certain amount of propellant to be kept for the return maneuvers, thus adding to the inert mass of the launcher acceleration mission and hence reducing payload performance [8, 15].

In another EC-funded project RETALT coordinated by DLR the aerodynamic performance of such returning stages is investigated in windtunnel experiments and numerical simulations. An example of such a first-stage configuration's model is shown in Figure 11.

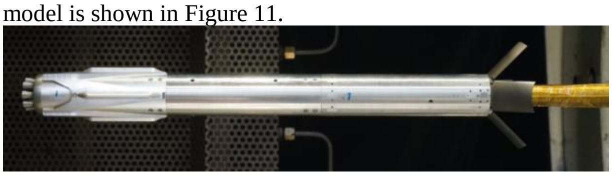

Figure 11: RETALT1 configuration in DLR windtunnel TMK (https://www.retalt.eu/2021/05/03/aerodynamics-wind-tunnel-tests/)

3.5 Orbit injection strategies GTO and LEO

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

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

In case of LEO-missions the launcher can best be operated as TSTO. The orbital injection conditions of the expendable 2nd stage will require an active deorbiting of the H150. In case of an ISS-resupply mission the stage's splashdown is foreseen in the Pacific Ocean in the vast remote areas east of New Zealand [11].

3.6 VTHL: 3STO-Configurations to GTO

Fulfilling the payload requirement of approximately 14 t in GTO but maintaining the architecture of three hydrogen stages ("HHH") has been iteratively investigated in previous papers [9, 11]. The RLV's propellant loading in this architecture is limited despite the powerful performance which makes a relatively compact lay-out possible. The 2nd expendable stage is defined as an H150 in case of hydrogen and becomes even more compact than the core stage of the classical Ariane 5G. An important design constraint is the requirement of using similar engines in the reusable stage and the expendable second stage, however, with adapted nozzles. This engine similarity allows for reduced development costs and might permit the reusable engine to be expended after certain number of missions on the RLV. The upper stage for high performance missions, mainly GTO-injection, is selected as H14 for all concepts and is placed under the large fairing. An external tank diameter of 5.4 m is no longer suitable for that loading if the stage's dry mass should be attractive. Vinci is the sole engine choice in the 3rd stage.

The architecture definition of the 3STO launcher in its latest design iteration is visible in Figure 12. Note the expendable stage arrangement with the H150 forward skirt or 2-3-interstage adjacent to the RLV intertank ring. This design adjustment compared to [11] improves load path, ascent controllability and 2nd stage engine environment. The stages' placements are similar for all RLVC4-III-concepts discussed in this section. However, the actual size and dimensions as well as the wing concepts significantly differ as presented in the subsequent paragraphs.

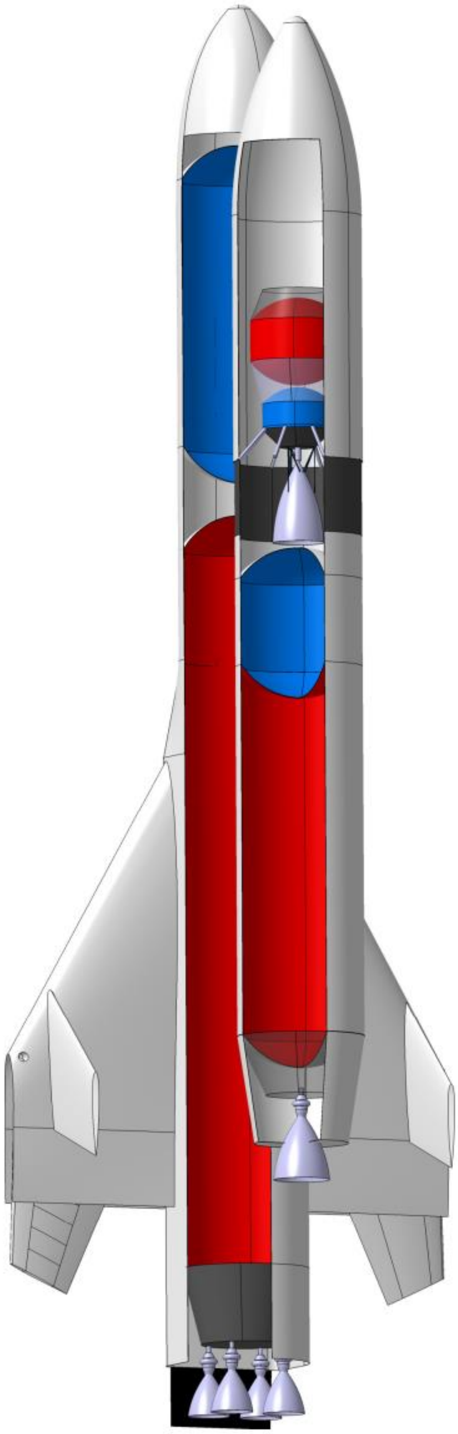

Figure 12: Launcher architecture sketch of 3STO RLVC4-III-B configuration

3.6.1 Type RLVC4-III-B

The Staged Combustion version of the RLV concept features the LOX/LH2 propelled SLME engines in the first and second stage: the 1st stage is equipped with 4 SLMEs and the 2nd stage is propelled by one single SLME.

This version features foldable wings as shown retracted to the back in Figure 13 for ascent and hypersonic reentry. The advantage of such a design is related to reduced flow perturbances during ascent and avoiding interaction between the nose shock and the wing's leading-edge shock, otherwise resulting in potentially critical heat flux values. These in turn demand for a reinforced TPS at the affected wing parts. This phenomena was discussed in several DLR studies and was identified as being more critical, the higher the re-entry velocity [8, 24, 25]. 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 better than with a fixed-wing configuration.

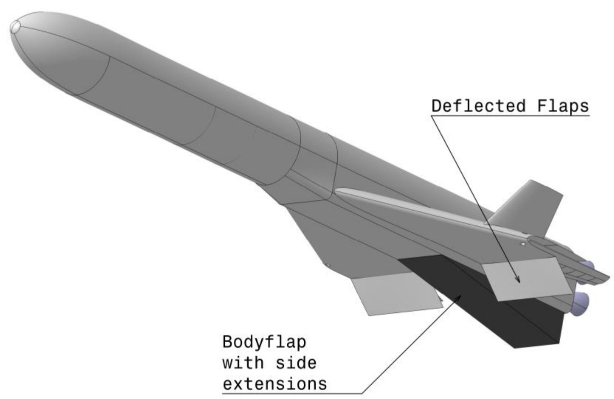

Figure 13: View on lower side of RLVC4-III-B configuration in hypersonic reentry mode

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 [27]. Recently, a new investigation on potential updates of the SpaceLiner Booster has been furthermore looking into swept-wing design options [24].

The wing geometry parameters and the wing position with respect to the fuselage are offering several degrees of freedom to the design. A favorable swept-wing configuration was found by comparing a vast range of different possible wing configurations in a partially automatic variation of parameters [11].

Furthermore, this RLV version features a rather large bodyflap extending over some part of the lower fuselage in order to improve pitch trim characteristics. The bodyflap (highlighted in dark in Figure 13) will be extended only during hypersonic reentry and high AoA. Adding sidewalls to the flap could help improving the vehicle's yaw stability when the vertical stabilizers on top of the wing show limited efficiency.

The RLVC4-III-B overall dimensions are listed in Table 5. The additional 35 t of propellants on the RLV compared to the previous design [11] because of increased dry weight add to the fuselage length.

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

H370 RLV stage	
total length (incl. bodyflap)	60.4 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

Any large movable structure will add to the complexity and to the weight of the RLV-stage. Structural pre-sizing activities have been initiated. Reconfiguration of the wing is planned in low dynamic pressure conditions to save on the actuator mass. Future analyses should assess current weight estimates.

The B – version with SC engines with their high performance is the lightest of all RLVC4-III-variants. The GLOM of roughly 667 t and the payload mass of almost 14 t leads to a comparably high payload ratio of almost 2.1%, the highest among all herein presented versions.

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

1st stage	**H 370**
Ascent Propellant	370.0 t
Dry Mass	80.0 t
GLOW	458.8 t
Structural Index incl. Engines	21.6 %
2nd stage	**H150 ELV**
Ascent Propellant	150.0 t
Deorbit Propellant	-
Dry Mass (w/o fairing)	17.3 t
GLOW (incl. fairing)	174.2 t
Structural Index incl. engine and 3rd stage adapter w/o fairing	11.2 %
3rd stage	**H14 ELV**
Ascent Propellant	14.3 t
Deorbit Propellant	0.5 t
Dry Mass	4.1 t
GLOW (incl. P/L)	33.7 t
Structural Index incl. engine and SYLDA	26.7 %
Separated Payload GTO	**13913 kg**
Total GLOM	**666.8 t**

3.6.2 Type RLVC4-III-G

The previously described B -version requires 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 450 tons, which is an addition of

80 tons compared to the III-B version. 10 PROMETHEUS-H engines are used for the ascent until stage separation resulting in a thrust-to-weight ratio of 1.56 at lift-off. The 2nd stage is marginally larger than -B with a propellant mass of 152 tons and keeps one engine. Insertion into GTO requires the same small third stage with slightly more than 14 tons of propellant. See an overall mass-breakdown of the vehicle in Table 7. The launcher GLOW increases by almost 100 t (+ 14%).

Table 7: Launcher masses by stage 3STO RLVC4-III-G, GTO mission

1st stage	**H 450**
Ascent Propellant	450.0 t
Dry Mass	95.0 t
GLOW	551.8 t
Structural Index incl. Engines	21.1 %
2nd stage	**H152 ELV**
Ascent Propellant	152.0 t
Deorbit Propellant	-
Dry Mass (w/o fairing)	15.8 t
GLOW (incl. fairing)	174.7 t
Structural Index incl. engine and 3rd stage adapter w/o fairing	10.4 %
3rd stage	**H14 ELV**
Ascent Propellant	14.3 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	**14141 kg**
Total GLOM	**760.0 t**

In order to limit the length of the RLV-stage and increase the lift-capabilities during re-entry, the fuselage diameter is set to 6 m. The primary lift is generated by a fixed double-delta wing with a total span of 35.5 m resulting in a trimmed lift-to-drag ratio of approximately six during subsonic in-air-capturing conditions (compare [11]). The trailing edge of the wing is positioned approximately 3.4 m before the end of the fuselage to ensure pitch stability in the hypersonic range during which the RLV-stage flies with high angles of attack.

The wingtips feature large-size vertical fins which are implemented for directional stability. After preliminary computations, these are necessary in combination with a reaction control system in the nose (similar to the Space Shuttle) to provide stabilization during the supersonic re-entry. However, given the complex aerodynamic behavior of such an RLV-stage, the research will be extended on this topic in the future. The overall dimensions of the launcher are listed in Table 8.

The ascent profile is similar compared to the -B-variant, with a few key distinctions [11]. 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).

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

H450 RLV stage	
total length (incl. bodyflap)	64.5 m
fuselage diameter	6.0 m
total span	35.5 m
H152 ELV	
total length (incl. fairing)	50.8 m
fuselage diameter	5.4 m
H14 ELV (under fairing)	
total length w/o Vinci	6.5 m
fuselage diameter	4.0 m

3.6.3 Type RLVC4-III-M

The (M)ethane proposal for the RLVC4-III consists of two different versions: The first ("MH") uses LOX/LCH4 in the RLV-stage and LOX/LH2 in the second stage while the second ("MM") utilizes LOX/LCH4 in both the RLV and second stage. The propulsion is based on the PROMETHEUS engine currently in development as well as the drafted PROMETHEUS-H variant. Also in case of the primarily methane-based concepts, the third stage uses LOX/LH2 with the Vinci engine.

Given the lower specific impulse of the LOX/LCH4 propellant combination, these versions require significantly more propellant than their hydrogen counterparts. Because of the LOX/LH2 second stage, the MH version only requires 620 tons while the MM RLV-stage propellant mass is to be increased to 800 tons. Consequently, 11 and 15 PROMETHEUS engines are used on the RLV-stages respectively.

While the second stage in the MH-version is identical to the G-version, the MM variant requires a complete redesign. With 230 tons of propellant in the second stage, it is significantly heavier. Therefore, this stage requires two PROMETHEUS engines to inject the third stage and payload into the transfer orbit. This design change decreases the compatibility with the current Ariane 6 LLPM significantly and is therefore not considered favorably.

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

1st stage	**C620**	**C800**
Ascent Propellant	620.0 t	800.0 t
Dry Mass	84.6 t	105.5 t
GLOW	712.7 t	915.7
Structural Index incl. Engines	13.6 %	13.0 %
2nd stage	**H152 ELV**	**C230 ELV**
Ascent Propellant	152.0 t	230.0 t
Deorbit Propellant	-	-
Dry Mass (w/o fairing)	15.9 t	16.0 t
GLOW (incl. fairing)	174.8 t	254.1 t
Structural Index incl. engine and 3rd stage adapter w/o fairing	10.5 %	8.2 %
3rd stage	**H14 ELV**	**H14 ELV**
Ascent Propellant	14.3 t	14.4 t
Deorbit Propellant	0.5 t	0.5 t
Dry Mass	4.0 t	4.0 t
GLOW (incl. P/L)	33.4 t	33.5 t

Structural Index incl. engine and SYLDA	25.9 %	25.7 %
Separated Payload GTO	**14011 kg**	**14047 kg**
Total GLOM	**920.9 t**	**1203.3 t**

Because of the higher density of liquid methane, the size of the vehicles is smaller even though they have significantly higher lift-off masses. Both versions are roughly 55 m long, the only difference being the diameter with 5.4 m (MH) and 6.0 m (MM). While the wing is similar in size with the G-variant, its layout and positioning is different because of disparate mass distributions with more engines at the back of the vehicle. For the MH-version, the trailing edge is placed in line with the end of the fuselage and the wing has no sweepback. In contrast, the mass of the 15 engines in the MM-version requires a trailing edge sweepback of 1.5 meters in the inner part of the double-delta wing in order to guarantee pitch stability in the hypersonic range. The wingtip fins are similar in size to the G-version and the vehicles also feature an RCS placed in the nose. The dimensions of the stages can be found in Table 10. The current geometries of the methane-boosters fulfill the aerodynamic and accommodation requirements. Nevertheless, further geometry iteration and improvement of these RLV-stages is likely in the future.

Table 10: Major stage dimensions 3STO RLVC4-III-M

	C620 MH RLV stage	**C800 MM RLV stage**
total length (incl. bodyflap)	54.5 m	56.25 m
fuselage diameter	5.4 m	6.0 m
total span	34.9 m	35.5 m
	H152 ELV	**C230 ELV**
total length (incl. fairing)	50.8 m	41.5 m
fuselage diameter	5.4 m	5.4 m
	H14 ELV	**H14 ELV**
total length w/o Vinci	6.5 m	6.5 m
fuselage diameter	4.0 m	4.0 m

3.6.4 Comparison of RLVC4-types

Most obvious when comparing the mass data in Figure 14 are the large differences between the RLVC4 versions analyzed within the scope of this study.

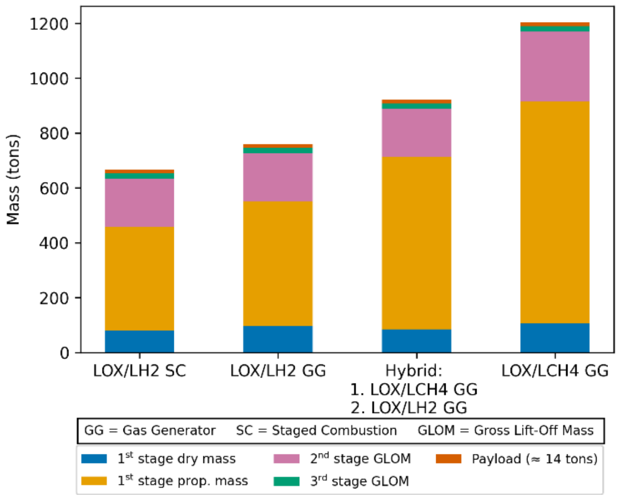

Figure 14: Mass comparison of RLVC4-types

All types dimensioned for a similar payload mass of 14000 kg to GTO, the gross lift-off mass almost doubles from LOX/LH2 with staged combustion (B) to LOX/LCH4 with a gas generator cycle (MM) in RLV and 2nd stage. This increase is mainly due to propellant mass of the RLV-stage but has also an impact on its dry and engine mass.

Increased bulk density of the propellant combination LOX-/LCH4 compared to LOX/LH2 helps in limiting the growth of the RLV-stages' size and dry weight. Nevertheless, the estimated mass of the C800 RLVC4-III-M is more than 30% above the -B variant's dry weight with staged combustion engines. Note also the increased complexity when moving from 4 main engines on the RLV-booster stage to 15. Both factors will have a negative impact on production and refurbishment costs.

3.7 VTVL: 3STO-Configurations to GTO

The VTVL concept presented here is derived of the RLVC4-III-B winged concept from section 3.6.1. The first stage is stripped of all hardware specific for VTHLs, namely wings, fins and flaps, rudders and landing gear. Instead, landing legs and grid fins are added which are scaled by dry mass with the respective masses of the Falcon 9 grid fins and landing legs.

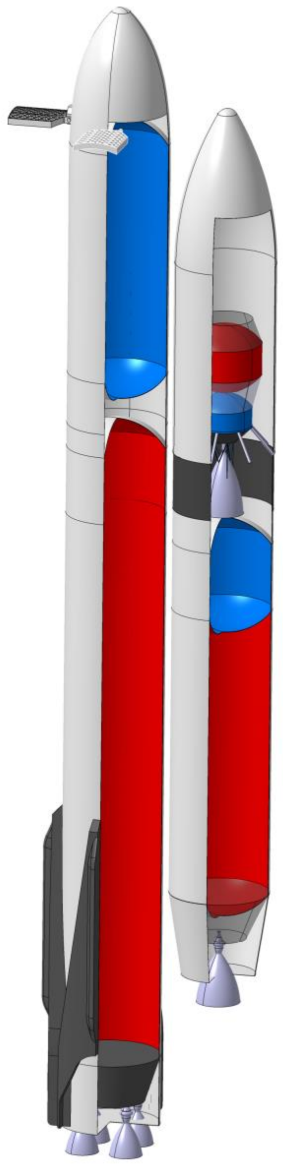

Figure 15: 3STO Configuration with VTVL-type RLV and extended grid fins

The ascent profile stays the same as for the VTHL concepts, so three stages are delivering the payload into its final GTO. A certain amount of the propellant in the 1st stage tanks is kept for performing the two re-entry maneuvers, the re-entry-burn and the landing burn. The whole launcher is sized to deliver the same 14 tons of payload into GTO, as the VTHLs. The system is iterated for target payload mass and minimal landing propellant mass.

Table 11 shows the mass breakdown of the VTVL version. The main difference to the VTHL system is the increase in propellant mass. The dry mass is less compared to the VTHL system due to the fewer hardware that is needed while propellant loading has to increase by 10% and hence also length of the RLV-stage by approximately 3 m. In total, the GLOM of the VTVL system is at 676 t including the 14 t payload, leading to a total difference of only 10 t more for the VTVL system. Although, dry mass is lower than for C4-IIIB, the net mass fraction at MECO is about 10% worse. This result agrees with previous systematic research [8, 15].

Table 11: Launcher masses by stage 3STO VTVL, GTO mission

1st stage	**H 407**
Ascent Propellant	380.0 t
Descent Propellant	27.3 t
Dry Mass	64.6 t
GLOW	478.8 t
Structural Index incl. Engines	15.6 %
2nd stage	**H150 ELV**
Ascent Propellant	150.0 t
Deorbit Propellant	-
Dry Mass (w/o fairing)	17.3 t
GLOW (incl. fairing)	174.2 t
Structural Index incl. engine and 3rd stage adapter w/o fairing	11.2 %
3rd stage	**H14 ELV**
Ascent Propellant	14.2 t
Deorbit Propellant	0.5 t
Dry Mass	4.2 t
GLOW (incl. P/L)	33.5 t
Structural Index incl. engine and SYLDA	26.9 %
Separated Payload GTO	**13933 kg**
Total GLOM	**686.5 t**

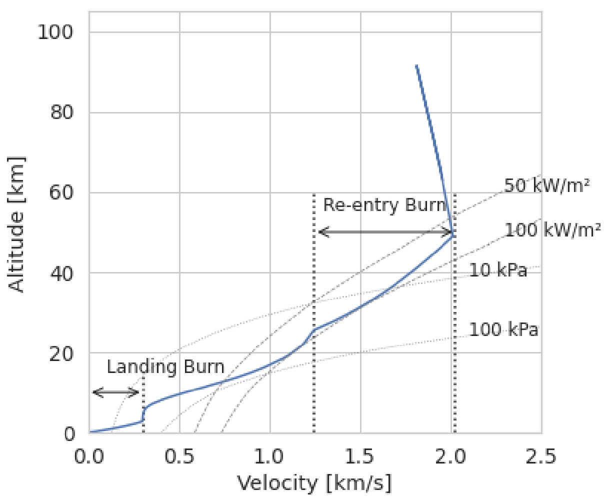

Figure 16: VTVL simulated atmospheric reentry for down-range landing

4 CONCLUSION

Different near future evolution options of Ariane 6 are investigated in the first part of the paper.
The EURASTROS study explored the feasibility of adapting A64 to astronautical missions into LEO following a low-cost approach. A conceptual definition of all major components has been performed and human safety aspects of the system in flight abort has been assessed. Study results confirm the general feasibility of the concept, capable of transporting up to 3 astronauts from Kourou to space.

An alternative evolution of Ariane 6 could be the replacement of the solid P120C boosters by liquid side-mounted stages with the propellant combination LOX/LCH4. Using such new boosters in expendable mode would raise A6 payload performance up to 20%. Switching to reusability, the achievable payload mass will be reduced. However, reusable side boosters seem to offer the possibility to introduce reusability to the Ariane launcher family without a severe impact on the payload capability when compared to the baseline Ariane 64. Missions could be flown either as ELV or RLV with DRL or RTLS, depending on the requested payload to target orbit.

A more radical approach of new European launcher design still based on Ariane heritage could be introduced in the mid-2030s. Different partially reusable launcher concepts have been investigated in 3-stage to orbit (3STO)-configurations for heavy-lift GTO-missions. All concepts are of asymmetric architecture with a winged RLV booster in parallel arrangement to expendable stages. Baseline for winged 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, however, the impact of choosing Methane in the lower stages is also investigated. Lift-off mass would almost double for the same payload performance when using LOX/LCH4 and gas-generator cycle.

The investigation on high-performance 3STO-configurations for Europe has been applied for the first time to reusable stages with vertical landing on a droneship in the Atlantic. While the dry-mass of the RLV would be reduced, the stage’s length and lift-off weight will increase compared to the corresponding winged concept.

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