SP2018_00478

Assessment of a European Reusable VTVL Booster Stage

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KEYWORDS: RLV, VTVL, Reusability, Ariane, DRL, RTLS, Gas Generator, Staged Combustion

ABSTRACT: Reusability of launch systems has the potential to strongly impact the launch service market if sufficient reliability and low refurbishment costs can be achieved. This study focuses on the vertical takeoff and vertical landing (VTVL) method as currently used by SpaceX. The goal is to determine the impact of this method on a technical, operational and economic level and to identify its potential for a future European reusable launch system with a VTVL booster stage. Therefore, different propellant combinations, stagings, engine cycles and landing methods (downrange landing vs. return to launch site) were considered for a launch system with a reusable VTVL booster stage and a payload capability of 7 tons to GTO (geostationary transfer orbit). The most promising concepts were subjected to a preliminary design loop at subsystem level and are presented in this paper.

ABBREVIATIONS

CNES	Centre national d'études spatiales
CSG	Centre Spatial Guyanais in Kourou
ELV	Expendable Launch Vehicle
LC3H8	Liquid Propane
LCH4	Liquid Methane
LH2	Liquid Hydrogen
LOX	Liquid Oxygen
RLV	Reusable Launch Vehicle
SART	Department of Space Launcher System Analysis
SCORE-D	Staged Combustion Rocket Engine Demonstrator

1 INTRODUCTION

Reusability in space transport can have a strong impact on launch costs and thus the launch market. However, the historic Space Shuttle system, consisting of two reusable solid booster stages and a fully reusable Orbiter with crew compartments, showed that reusability can also lead to increasing launch costs if refurbishment costs cannot be kept low.

Nonetheless, the recent successes of SpaceX (with Falcon 9 and Falcon Heavy) and Blue Origin (New Shephard) in landing, recovering and reusing their respective booster stages by means of retropropulsion have shown the possibility of developing, producing and operating reusable launchers at low launch service costs that are able to compete on the current launch market worldwide. Hence, the need in Europe to thoroughly investigate different methods of reusability to keep up with the evolving launch market has become of essential importance [1], [2].

Several studies on partly or fully reusable first stages using different return methods have been conducted in the past at DLR. The liquid fly-back booster concept was studied extensively during the early 2000s [3]. This concept features a winged first stage equipped with aerodynamic control surfaces to perform a lift-supported re-entry after MECO (Main Engine Cutoff), slowing down the booster from hypersonic to subsonic speeds. When reaching subsonic speed, the winged stage would either switch on several air-breathing turboengines to fly back to the launch site or would be captured by an airplane to be towed back to its landing site [4]. However, through the recent successes using the ballistic retropropulsion method the interest in studying and understanding this method has further increased. Thus, in the past year, an extensive study on both VTHL (vertical takeoff, horizontal landing) and VTVL stages has begun in the framework of the DLR XTRAS study [1]. This study focuses on identifying the impact of both methods on performance, mass, structure, recovery hardware, operations and finally launch costs. Main goal of this study is to compare both methods while emphasizing on optimizing the launch systems to a comparative level in order to avoid distortions by different optimization levels.

At the end of this comparative study, one promising return concept for each the VTVL and the VTHL method will be selected for a second step of the study with the aim to obtain realistic designs for an operational launch system and to identify required technology developments depending on the return mode.

This paper focuses on the VTVL launchers investigated in the framework of this study. To allow a broad picture of this method, different stagings, propellant combinations, engines, engine cycles and landing scenarios were considered. The launchers were designed using the following assumptions:

- 7000 kg + 500 kg margin payload to GTO of 250 km x 35786 km x 6° (standard Ariane 5 GTO)
- Launch from CSG, Kourou
- TSTO: Two Stage to Orbit
- Same propellant combination in both stages
- Same engines in both stages with exception of different nozzle expansion ratios (as for Ariane 4 1st stage and 2nd stage engine)
- Engine Cycles: Gas Generator (GG) and Staged Combustion (SC)
- VTVL with retropropulsion landing at either launch site (RTLS) or on downrange barge (DRL)
- 2nd stage Δv of 6.2 km/s, 6.6 km/s, 7.0 km/s, 7.6 km/s
- Propellant Combinations: LOX/LH2, LOX/LCH4, LOX/LC3H8

It is important to note that the argument of perigee of the reached GTO orbit was not set to 0° or 180°, as necessary for an actual GTO. This preliminary assumption still allows comparing the designed launchers considering delivered payload, but the payload delivered to a GTO with correct argument of perigee might differ. However, in the future course of the study the correct argument of perigee shall be taken into account.

The VTVL launchers were designed in a first design loop using preliminary assumptions. Therefore, structural index (SI) curves were derived from historical launchers to estimate the dry mass of both first and second stage. The thus calculated launchers are presented in detail in [1] and the results are briefly summarized in section 2.

Based on the results of this analysis with structural index calculated mass a more detailed launcher model with mass estimation from structural design tools was established and applied to selected launchers from the first design loop. This preliminary design model includes modeling of the propellant system, the engines and engine cycles, the tank, interstage and skirt structure and a refined mass estimation of several subsystems. The methods and assumptions used for these preliminary launchers are presented in section 3. The thus designed launchers are presented and discussed in section 4.

2 FIRST DESIGNS BASED ON STRUCTURAL INDEX ASSUMPTIONS

2.1 Assumptions

The general layout of all launchers follows the same principle; both stages use the same propellant combination and the same engines with slight modifications (vacuum adapted engine nozzle). This approach is considered cost-effective since only one type of engine has to be developed and produced. Furthermore, only the first stage is recovered by using retropropulsion of the own rocket engines. The further hardware necessary for recovery are grid fins for limited aerodynamic steering of the stage and deployable landing legs to allow for a soft landing. Fig. 1 shows the general layout of the designed launchers. As can be seen the launchers consist of (from bottom to top) a rear skirt containing the first stage engines, the two first stage tanks separated by a common bulkhead, an interstage accomodating the second stage engine's nozzle, the second stage tanks and a fairing.

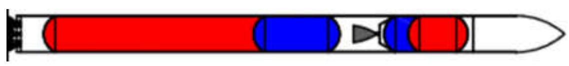


Figure 1: Generic layout of designed launchers

In the first design loop, the respective launchers were designed using structural indices derived from actual and historical launchers to estimate the dry mass. Since no actual launchers using methane and propane were built, the corresponding structural index formula was interpolated by using the bulk densities and known structural indices of the LOX/LH2 and LOX/RP-1 propellant combinations. The engine mass was calculated with the DLR-SART software *lrp* (liquid rocket propulsion). The mass of the grid fins and landings legs was estimated by scaling the respective masses of the Falcon 9 with the launchers' dry mass. A detailed explanation of the assumptions and methods used for this first design loop is given in [1].

Only the LOX/LH2 launchers were calculated using different engine cycles, namely staged combustion and gas generator (see section 3.1 for more detail), while all other launchers were designed using the gas generator cycle. The gas generator cycle is considered more cost-efficient since gas generator engines are less complex and thus development and production are not as expensive as for staged combustion engines. Furthermore, staged combustion engines are generally heavier than gas generator engines. However, the staged combustion cycle offers a higher specific impulse and thus payload mass.

In the first design loop two different landing options were considered; the RTLS method with landing at or close to the launch site and the DRL method with landing on a barge downrange. The advantage of the former is the fact that the rocket is brought back to the launch site, thus minimizing recovery effort and turnaround times. However, the RTLS method has the great expense of severely reducing the performance and the maximum payload mass, since more propellant is required for the return manoeuvers [2], [5]. In case of an RTLS landing three burns are performed: the boostback burn (setting the trajectory to the landing site), the re-entry burn (reducing aerothermal loads on the stage) and the final landing burn to slow the rocket down to safe landing velocity. In case of a downrange landing, only two burns are performed: the re-entry and the final landing burn.

2.2 Results

The results of the first design loop for launchers performing RTLS showed that these launchers reach sizes that are unfeasible regarding the delivered payload mass to GTO. The LOX/LCH4 launcher with an upper stage Δv of 7.6 km/s reached a GLOM (gross liftoff mass) of 3800 tons of which 500 tons were propellant needed for the return manoeuvers. The payload fraction of this launcher is less than 0.2%, thus rendering the RTLS performing launchers highly inefficient. Hence, it was decided that no further calculations shall be done for RTLS launchers within the second design loop, since RTLS is only economically feasible for LEO missions due to the lower Δv required.

Fig. 2 shows the GLOMs of all launchers calculated in the first design loop performing a downrange barge landing. Generally, all hydrogen launchers are characterized by the letter H, all methane launchers by the letter C (with TPC for triple point methane) and all propane launchers by the letter PR, followed by the amount of propellant in tons for the respective stage (two numbers are necessary for two stages). The upper stage Δv describes the Δv the second stage delivers after MECO excluding all losses. Hence, the higher the upper stage Δv, the lower the MECO and separation velocity of the first stage.

As expected, the LOX/LH2 launchers are the lightest by GLOM with a noticeable mass advantage of the staged combustion launchers due to their higher performance. The LOX/propane launchers are slightly lighter than the LOX/methane launchers. Generally, the launchers have a lower GLOM with decreasing upper stage Δv due to the fact that the upper stage's mass decreases. The first stage mass is driven by two counteracting effects; first, lighter upper stages lead to lighter lower stages, since the structure has to bear less loads. Second, the lower stage needs to deliver more Δv (hence needs more propellant) with decreasing upper stage Δv. However, the effect of decreasing second stage mass seems to have much more influence on the total GLOM in the range from 6.6 km/s to 7.6 km/s upper stage Δv.

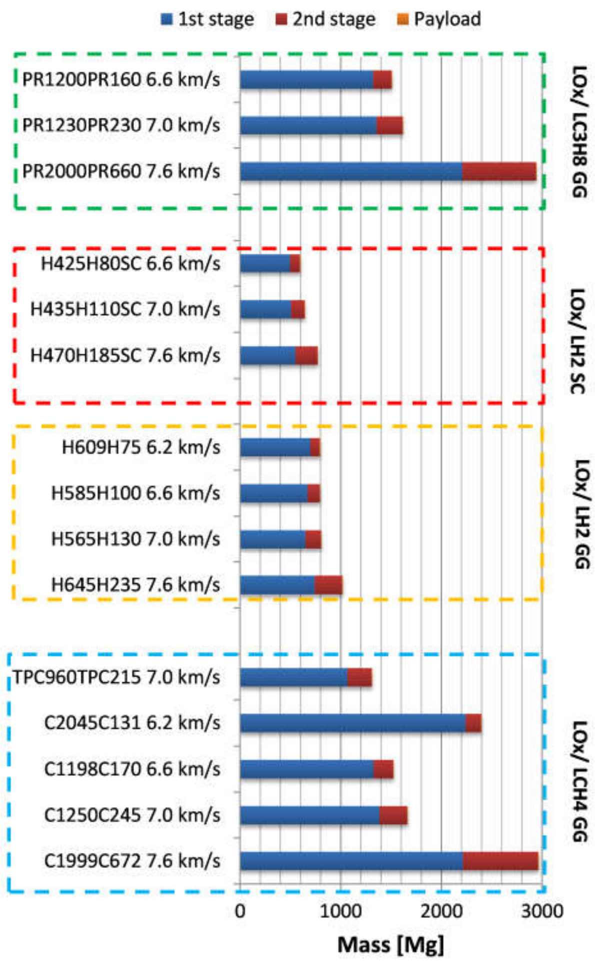


Figure 2: GLOM of the LOX/propane, LOX/LH2 with staged combustion, LOX/LH2 with gas generator and the LOX/methane launchers of the first design phase

Nevertheless, the LOX/methane launchers clearly show that an optimum is reached between 6.6 km/s and 7.0 km/s upper stage Δv based on the observation that the GLOM of the 6.2 km/s version is much higher than the GLOM of the 6.6 km/s variant. This effect was also observed for the LOX/LH2 launcher but is much less pronounced which can be explained by a different behavior of the underlying SI curve.

The results of the preliminary design loop led to the conclusion that an upper stage Δv between 6.6 km/s and 7.0 km/s is favorable for downrange barge landings of the suggested launcher configuration. Therefore, the second design phase with a more detailed model was applied to the LOX/LH2, LOX/LCH4 and LOX/LC3H8 launchers with an upper stage Δv of 6.6 km/s. For an upper stage Δv of 7.0 km/s the LOX/LH2 and LOX/LCH4 launchers were designed.

3 METHODS & ASSUMPTIONS FOR PRELIMINARY DESIGN PHASE

3.1 Main Propulsion Rocket-Engines

The systematic assessment of future RLV-stages and technical options, as intended in the XTRAS-study, requires the definition of generic engines with similar baseline assumptions in order to reach maximum comparability. Despite the engines being generic, their selected technical characteristics for simulation are strongly oriented towards data of existing types or previous or ongoing development projects, whenever possible.

The two rocket engine cycles most commonly used in first or booster stages are included in the study:

- Gas-Generator-cycle
- Staged-Combustion cycle.

The main combustion chamber (MCC) pressure is commonly set to 12 MPa for the gas-generator type. This pressure is not far from the useful upper limit of this cycle but is assumed necessary to achieve sufficient performance for the RLV stages. Europe has considerable experience in this range with Vulcain 2 operating at 11.7 MPa. In case of the staged-combustion engines, the main combustion chamber pressure is commonly fixed at 16 MPa. This moderate value in Russian or US perspective has been chosen considering the limited European experience in closed cycle high-pressure engines. Nozzle expansion ratios are selected according to optimum performance but also requirements of safe throttled operations when landing VTVL-stages. For the first stage engines data are calculated for expansion ratios of 20 and gas generator types and 23 for the staged-combustion variants.

The upper stage engines are derived from the first stage engines with the only difference being the expansion ratio. This value was set to 120 based on the results from the first structural index design cycle showing that the designed upper stage engine nozzles with an expansion ratio of 180 exceeded the interstage dimensions. Furthermore, all other engine parameters are equal to those of the first stage (mass flow, chamber pressure, engine cycle…).

The two engine cycle options have been applied to the three different propellant combinations mentioned in section 1. Furthermore, engines using the propellant combination LOX/RP-1 were calculated but are not described in this paper since no launchers using that combination were designed yet.

All preliminary engine definitions have been performed by simulation of steady-state operation at 100% nominal thrust level using the DLR-tools *lrp* and *ncc* (nozzle contour calculation program) as well as the commercially available tool *RPA* (rocket propulsion analysis). Any potential requirements specific to transient operations or deep-throttling are not considered in this early design study. Common baseline assumption of all engines is a vacuum thrust in the 2200 kN-class. Although all engine massflows are scaled to the required thrust level of the individual launcher configuration, the underlying assumptions on component efficiencies (e.g. turbo-pumps) are most likely no longer valid for smaller engines below 1000 kN. Turbine entry temperature (TET) is set around 750 K and kept in all cases below 800 K to be compatible with the increased lifetime requirement of reusable rocket engines.

Further, for all engines in this study regeneratively cooled combustion chambers are assumed with regenerative or dump-cooling of the down-stream nozzle extensions.

3.1.1 LOX-LH2 engines

The combination of liquid oxygen with liquid hydrogen delivers the highest practically achievable mass specific performance. Water as the reaction product is also the most environmentally compatible exhaust. The low bulk density due to the low density of hydrogen and its very low boiling temperature are the key challenges.

Europe has gained significant experience with these propellants in more than 50 years and has flown several hundred engines up to date (HM7 since 1979 and Vulcain since 1996).

The engine mixture ratio of all types has been set a priori at 6.0 which is a good compromise between performance, acceptable propellant bulk density of the stage, and technical feasibility of the combustion process. This choice of 6.0 is supported by the design of all existing LOX-LH2 main stages.

As clearly visible in Fig. 3, the main combustion chamber operating points are off the optimum performance, shifted to the right towards increased density. In case of the gas generator cycle this shift is more pronounced into less favorable regions (indicated by red arrow) because the turbines are driven by strongly hydrogen-rich hot gas. The closed cycle's MCC MR (mixture ratio) is exactly at the engine MR of 6.0.

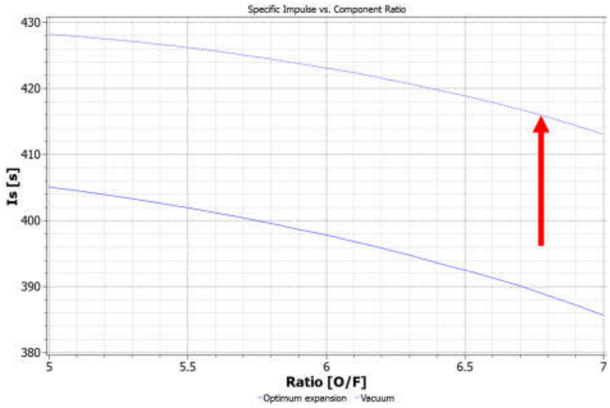
The architecture of the open cycle is following the typical approach with single gas generator and two separate turbopumps run in parallel. This is similar to the Vulcain engine. A comparison of calculated performance (see Table 1) with real operating engines is most suitable for the RS-68 with a similar launcher application and relatively low expansion ratio. The calculated engine performance is found slightly below those published for the RS-68. The very large size of the American engine (2950 kN) with related potential efficiency gains should be taken into account in the comparison as well as the reusability requirement for the VTVL engines. Therefore, estimated data are probably realistic.

The staged combustion cycle is derived of the SLME (SpaceLiner Main Engine) under investigation for several years at DLR [6]. A Full-Flow Staged Combustion Cycle (FFSC) with a fuel-rich preburner gas turbine driving the LH2-pump and an oxidizer-rich preburner gas turbine driving the LOX-pump is a preferred design solution for the SLME. In an FFSC, two preburners whose mixture ratios are strongly different from each other generate turbine gas for the two turbo pumps. All of the fuel and oxidizer massflows are routed through the preburners, thus subsequently powering the turbines before all are injected in hot gaseous condition into the main combustion chamber. The FFSC design is relatively complex but offers some operational and safety benefits [6]. Calculated performance of all closed cycles is independent of the internal architecture. Thus, a fuel-rich preburner variant like used in the SSME (Space Shuttle Main Engine) or proposed for the European SCORE-D demonstrator would achieve the same Isp as listed in Table 1. The difference is in engine complexity and resulting mass.

Note the high thrust to weight ratio (T/W) estimated by DLR's *lrp* program for the gas generator cycle engines. These data are used for the launcher sizing procedures with system margins added on top. Nevertheless, it should be kept in mind that the RS-68's T/W is not more than 46.

Table 1: Key performance data of LOX-LH2 engines

Parameter	*1st stage*		*2nd stage*	
	GG **ε= 20**	**SC** **ε= 23**	**GG** **ε= 120**	**SC** **ε= 120**
Engine MR [-]	6	6	6	6
Sea level Isp [s]	366	394	-	-
Vacuum Isp [s]	405.5	428	440.4	458.6
Engine T/W [-]	98.1	73.5	82.4	70.2



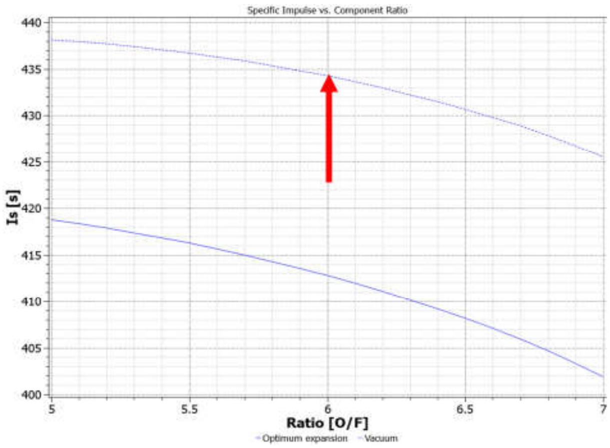


Figure 3: Influence of mixture ratio in main combustion chamber on performance (Gas generator top, staged combustion at bottom)

3.1.2 LOX-LCH4 engines

Several initiatives are currently working on engines with the propellant combination LOX-Methane. Although proposed several times in the past, this "softly cryogenic" blend has never yet been realized in an operational launcher stage.

The main combustion chamber MRs of this combination have been selected close to their optimum Isp, however, slightly shifted towards increased MR to reach increased bulk density. This approach is different to the LOX-LH2 engines and results in slight differences in MCC-MR and significant differences in the engine MR. The method is used in a similar way also for the other hydrocarbons and is justified by their increased propellant density but considerably lower mass specific impulse compared to LOX-LH2.

The gas generator operates methane-rich and its hot gas powers the single shaft turbine. Major characteristics are derived of the PROMETHEUS-Demonstrator [7] but the baseline assumptions remain similar to all other engines of the system study. Obtained data (Table 2) are not far off the expected PROMETHEUS-engine.

The staged combustion type is based on a fuel rich preburner design with a single-shaft turbopump. It's worth noting that both simulation tools *lrp* and RPA converged only for relatively high preburner pressures resulting in lower T/W than other engines. A direct comparison with another engine is not possible because the staged combustion methane engines under development in the US, Raptor and BE-4, intend to operate in FFSC and in LOX-rich-mode and at significantly different chamber pressures [6]. The LOX-Methane engines deliver the highest performance of all hydrocarbon types, yet roughly 80 s to 90 s below the LOX-LH2 engines.

Table 2: Key performance data of LOX-LCH4 engines

Parameter	*1st stage*		*2nd stage*	
	GG ε= 20	**SC ε= 23**	**GG ε= 120**	**SC ε= 120**
Engine MR [-]	2.5	3.25	2.5	3.25
Sea level Isp [s]	289	310.5	-	-
Vacuum Isp [s]	320	339	348	365
Engine T/W [-]	97.5	66	82.7	57.5

3.1.3 LOX-LC3H8 engines

The propellant combination of oxygen with propane has not yet been applied in the launcher sector. Characteristic specific impulse data is close to kerosene and methane, almost in the middle between both. Propane is also a "soft" cryogenic fuel, being in gaseous state under ambient conditions. Propane offers a higher bulk density compared to methane and its potential for densification was identified to be higher than that of methane [8].

The process for the selection of operational MR is the same as for the other hydrocarbon engines with MCC-MRs chosen close to their optimum Isp. However, cycle simulations of the propane engines are running into convergence problems with both tools *lrp* and RPA. The exact cause of these problems is not fully understood but is attributed to insufficient fluid property data for this propellant combination.

Table 3: Key performance data of LOX-LC3H8 engines

Parameter	*1st stage*		*2nd stage*	
	GG ε= 20	**SC ε=23**	**GG ε= 120**	**SC ε=120**
Engine MR [-]	2.45	2.8	2.45	2.8
Sea level Isp [s]	284	300	-	-
Vacuum Isp [s]	315	339	344	359
Engine T/W [-]	101.9	-	86.4	-

Obtained simulation data as listed in Table 3 are plausible, positioning the LOX-LC3H8 engines exactly in the middle between methane and kerosene. The cycle architectures are similar to the kerosene engines. A mass estimation attempt for the staged-combustion cycle was not successful because of the convergence problems. Any comparison to other engines is not possible due to the non-existence of this propellant type in spaceflight

3.2 Structure and Propellant System

The structural masses of first and second stage propellant tanks, the interstage, the second stage's front skirt and the first stage's rear skirt were calculated using the DLR SART tool *lsap* (launcher structural analysis program). Several load cases from the GTO trajectory were defined and imposed on the structure (including margins for dynamic loads) to determine the structural layout necessary to withstand possible failure modes. The safety factor was chosen to be 1.25, a standard value for unmanned launchers.

The tanks and skirts were designed using a conventional stringer/frame approach with "Z" – formed stringer and frames. The number of stringers and frames is subject to an optimization process within the program to determine the lightest configuration possible. The tanks are separated by a common bulkhead and made of the aluminum alloy AA2219. All tanks are pressurized with 3 bar which was chosen based on previous experience with launcher design and shall be subject to optimization in the future.

The interstage and fairing were modelled as an aluminum honeycomb structure with carbon fiber outer layers. Whereas the interstage mass is calculated with the tool *lsap*, the fairing mass was calculated by scaling the mass of the Ariane 5 fairing with the respective surface area.

Fig. 4 shows the structural model of the preliminary hydrogen launcher. The green parts are the front and rear skirt, the parts colored in red highlight the composite structures (interstage and fairing), the blue parts represent the tanks and the black lines display the outline of the tank domes.

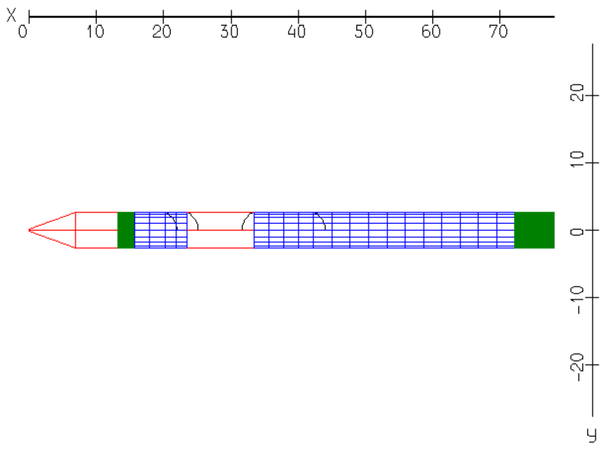


Figure 4: Structure Model of the preliminary launchers

The propellant supply system including feedlines, fill/drainlines and the pressurization system was modelled using the SART tool *pmp*. This program is able to calculate the respective masses for these systems by calculating the propellant and pressurizing gas flow throughout the whole mission and thus sizing the required hardware.

The pmp tool also calculates the mass of the cryogenic insulation of the tanks. It is important to note that insulation was only considered a necessity in the case of LOX/LH2 launchers due to the low temperature of LH2. In the case of LOX/LCH4 and LOX/LC3H8 launchers no insulation is used, since it adds mass and it is technically feasible to fly cryogenic propellants without insulation (e.g. Falcon 9).

3.3 Mass Model

The GLOMs of the preliminary launchers were calculated using the SART tool *stsm* (Space Transport System Mass Estimation). Fairing, tank, interstage, skirt and propellant system masses were calculated using the model described in section 3.2 and engine masses were calculated using the data assumed in section 3.1 for each propellant combination and required thrust. The grid fins and landing legs were linearly scaled with those of the Falcon 9 with respect to the stage's dry mass. This is a conservative approach since a preliminary analysis of the landing legs' structure and mass showed that the mass is overestimated with this approach. For future studies, a more detailed mass estimation model for grid fins and landing legs shall be applied.

The remaining subsystems were calculated using either empirical formula included in *stsm* or were scaled with values of the Ariane 6. It is important to note that these values are based on estimations and preliminary calculations done at DLR-SART. Subsystem masses calculated with empirical formulas include the engines' thrustframe, engine equipment including thrust vector control and engine controllers, electrics and finally harness.

Figure 5: Model of the propellant management system for a generic LOX/LH2 launcher

The model of the propellant system is shown in Fig. 5. The respective propellant system is that of the hydrogen launcher presented in detail in section 4. The LOX tanks are colored in blue, the LH2 tanks in red. The first stage main propellant feedline is branched into several smaller feedlines to individually supply the engines at the rear end of the launcher. The pressurization lines run along the backside of the tanks and supply the LOX tank with gaseous oxygen and the LH2 tank with gaseous hydrogen.

Subsystems scaled with Ariane 6 subsystem masses include the power system and batteries, the pyro-stage and fairing separation system, chill-down, start-up and RCS propellant, the avionics and RCS system and the payload adapter.

Since the mass estimation especially for RLVs is subject to uncertainties margins were included in the mass model. The margins for all first stage structure, subsystems and thermal protection components was set to 14%, the margins for propulsion components was set to 12%. The second stage margin was set to 10% for all components. The margin for the first stage was increased in order to reflect the larger uncertainties involved in sizing this comparatively new stage type.

3.4 Ascent and Descent Trajectory Optimization

The maximum payload of the designed launchers is calculated by a combined optimization of both ascent and descent trajectory of the first stage. The optimal ascent trajectory for maximum payload might not necessarily be the optimum for minimum descent propellant required for the return of the first stage. Hence, there exists an optimum combination of ascent and descent trajectory that leads to maximum payload. This combination is iteratively calculated by parametric variation of both trajectories control parameters which include the pitch manoeuver during ascent, the AoA (Angle of Attack)-profile during ascent and descent and the timing and thrust profile of the burns during descent.

Certain assumptions with regard to the descent trajectories were established to ensure that the calculated launchers perform feasible return trajectories not exceeding load factor and dynamic pressure limits. Generally, higher loads are experienced during re-entry and return of the first stage compared to the ascent, but since the stage is almost empty and is not carrying the upper stage atop, these loads are not dimensioning for the first stage's structure. This has been successfully proven in [9]. However, a maximum lateral load factor n_z of 3 g was set such as well as a maximum dynamic pressure of 200 kPa. First calculations showed that these seemingly high values can be beared by the first stage structure [9], [10]. The maximum allowable heat flux was set to the maximum value experienced during the Falcon 9 SES 10 mission re-entry [5]. This approach represents a "realistic engineering approach", but actual data on how SpaceX manages the loads during re-entry is publically not available. Hence, setting the boundary value for heat flux to a value similar to a Falcon 9 re-entry might be a practicable way but should be counterchecked with in-house calculations. First CFD calculations with the retropropulsion method were already conducted at DLR but the assumptions have to be refined to consider the heat intake into the engine nozzles during re-entry [10].

4 RESULTS OF PRELIMINARY DESIGN PHASE

In this section the results of the XTRAS launcher study with preliminary design assumptions are presented. It is important to note that all launchers calculated so far are using gas generator engines. Furthermore, the results in this paper are focused on the propellant combinations at standard temperatures, details considering subcooled propellants are presented in [8]. All data regarding Falcon 9 was gathered using models and tools developed at SART, that were validated with actual missions and trajectories [2],[5]. The mass and trajectory data of the Falcon 9 were taken from the SES 10 mission recalculated with DLR in-house tools. The upper stage Δv of the Falcon 9 was around 7.5 km/s for this mission [5].

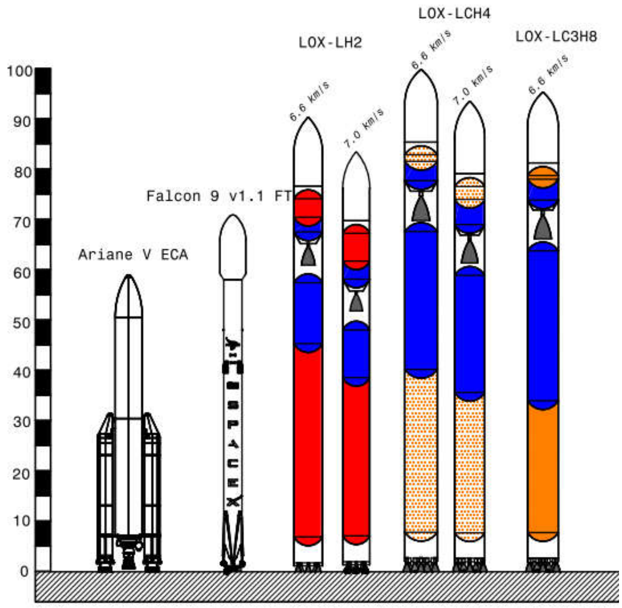


Figure 6: Ariane 5, Falcon 9 v1.1 FT, DLR LOX-LH2, LOX-methane and LOX-propane launchers

The DLR LOX/LH2, LOX/methane and LOX/propane launchers with an upper stage Δv of 6.6 km/s and 7.0 km/s compared to the Ariane 5 and the Falcon 9 v1.1 FT are shown in Fig. 6. The launchers with an upper stage Δv of 7.0 km/s are about 20% lighter than the respective 6.6 km/s launchers (see Table 4**Fehler! Verweisquelle konnte nicht gefunden werden.**). The hydrogen launcher with second stage Δv = 7.0 km/s is even lighter than the Falcon 9 (550 tons GLOM), although it is bigger due to the much lower bulk density of hydrogen compared to RP-1. This also explains the similar size of the methane and propane launchers compared to hydrogen. However, the hydrogen launcher only needs 9 smaller engines, whereas the hydrocarbon launchers need 15 to 17 heavier engines that only fit onto the rear skirt of the rocket when allowing slight overlapping of the outer engines over the perimeter of the aft skirt.

Table 4: Launcher Data of DLR LOX/LH2, LOX/LCH4 and LOX/LC3H8 launchers

Propellant combination		**LOX/LH2**		**LOX/LCH4**		**LOX/ LC3H8**
2nd stage Δv	**[km/s]**	**6.6**	**7.0**	**6.6**	**7.0**	**6.6**
1st stage						
No. of engines	[-]	9	9	17	15	17
Single Engine Mass	[kg]	1055	906	1605	1443	1460
Single Engine Thrust (sea level)	[kN]	906	773	1419	1278	1370
2nd stage						
Engine Mass	[kg]	1345	1150	2110	1898	1954
Engine Thrust (vacuum)	[kN]	1088	929	1711	1540	1653
Total Launcher						
Height	[m]	89.6	82.2	99.2	92.8	94.2
Diameter	[m]	5.7	5.4	6.45	6	6.1
GLOM	**[t]**	**602**	**479**	**1761**	**1384**	**1705**

The second stage Δv of 6.6 km/s also seems to be unfavorable considering the structural design and the engine mass to dry mass ratio of the second stage. The mass of the second stage's engine of the LOX/LH2 and LOX/LCH4 launchers with an upper stage Δv of 6.6 km/s is higher than the engine mass of the respective stages with 7.0 km/s Δv, even though the propellant loading is less for the 6.6 km/s Δv version (see Table 4). This can be explained by the fact that the heavier first stages need powerful engines to achieve sufficient thrust for lift-off. Since the second stage engine is an adapted first stage engine, the Δv = 6.6 km/s upper stage engines are overpowered and overweight. Thus, the engine thrust and engine mass to dry mass ratio is superior for the Δv = 7.0 km/s upper stages.

The GLOM breakdown of the calculated launchers is shown in Fig. 7. The Falcon 9 was added to provide comparison with an operational launcher. However, the Falcon 9 is able to deliver around 5.5 tons to a GTO orbit in RLV configuration while the launchers designed within this study can carry around 7.5 tons to GTO. In general, the hydrocarbon propelled launchers weigh around three times more than the respective hydrogen launchers due to the lower performance of this propellant combination. Furthermore, the LOX/propane combination has a slight advantage over the LOX/methane combination. The comparison of the GLOMs shows the advantage of the 7.0 km/s staging, leading to a 20% lower GLOM of the respective launchers. In the further course of the XTRAS study, launchers using LOX/RP-1 will be designed to allow for better comparison with the already calculated launchers.

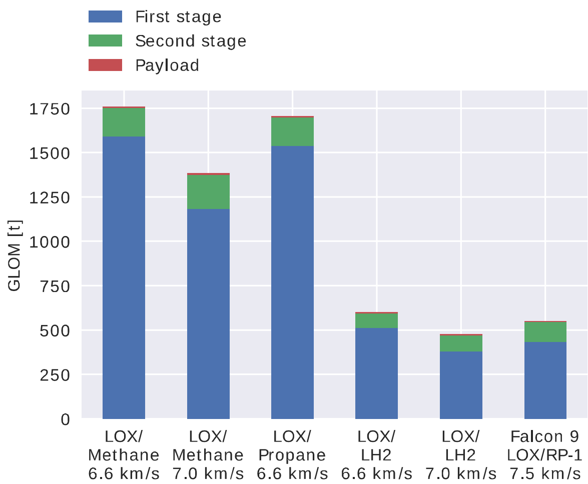


Figure 7: GLOM of the LOX/LH2, LOX/LCH4, LOX/LC3H8 DLR launchers and Falcon 9 (SES 10 mission)

The structural index (SI) shown in Fig. 8 is defined as shown in Eq. *1*. It is important to note that the structural index of the first stage includes the interstage mass and the SI of the second stage includes the fairing mass.

$$SI = \frac{m_{dry}}{m_{prop}} \quad \text{Eq. 1}$$

As expected, the structural indices of the LOX/hydrogen stages are higher compared to the hydrocarbon stages. The Falcon 9 bears the lowest structural index which can be explained by the fact that the stage is built out of light Al-Li alloys and uses subcooled propellants and light engines. Generally, the second stages were expected to have a higher SI since they carry less propellant than the first stage. However, expect for the Falcon 9 and the LOX/LH2 6.6 km/s launcher, the second stage SI is lower than the first stage SI. This effect can mainly be explained by the fact that the comparably heavy recovery hardware and interstage are added to the first stages, thus increasing the stage SI. The similar stage SIs of the LOX/LH2 6.6 km/s launcher can be explained by the fact that the engine- to dry mass ratio is not optimal for the second stage, since the stage is equipped with a too powerful and heavy engine and the structural layout is disadvantageous.

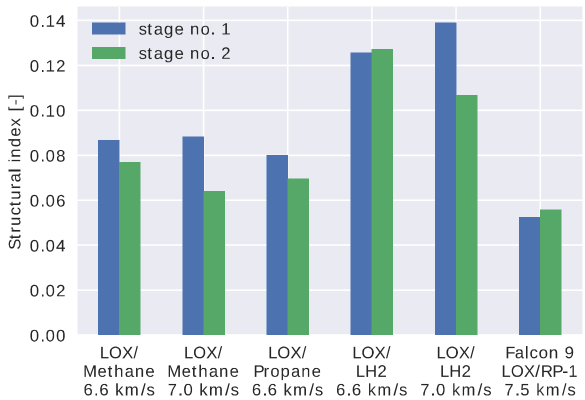


Figure 8: Structural index of first and second stage of the LOX/LH2, LOX/LCH4, LOX/LC3H8 DLR launchers and Falcon 9 (SES 10 mission)

The hydrogen launchers' first stages both have a SI in the range of 12.5% to 14%, the second stage's SI range from 10.5% to slightly above 12%. As a comparison, the second stage SI of the Saturn V was about 10% with around 450 tons of propellant. The respective third stage SI was in the order of 16% with a propellant loading of 106 tons. Hence, the Saturn V third stage has a similar propellant loading than the two second stages of the DLR hydrogen launchers. Nevertheless, the DLR hydrogen launchers only carry a payload of up to 7.5 tons while the Saturn V was designed to carry the up to 50 tons of Apollo spacecraft plus lunar lander, thus posing more loads on the structure. This shows that the design methods used within the study do lead to an optimistic stage mass estimation.

Fig. 9 shows a mass breakdown of the first stages. The largest portion of the stage mass is composed of ascent propellant. The descent propellant necessary to keep the stage within the designated limits while landing it safely on the barge is between 6.5% and 10% (LOX/LH2) and 7% to 9% (LOX/hydrocarbons) of the stage GLOM, with less propellant share for an upper stage Δv of 7.0 km/s.

It is important to note that a lower second stage Δv leads to a higher amount of ascent propellant in the first stage and thus a higher stage mass. Combined with the higher velocity at MECO, the first stage also has to carry more propellant for the return manoeuvers. Hence, the first stages whose upper stages deliver 6.6 km/s of Δv are substantially heavier and further away from the optimal point compared to the launchers with 7.0 km/s upper stage Δv.

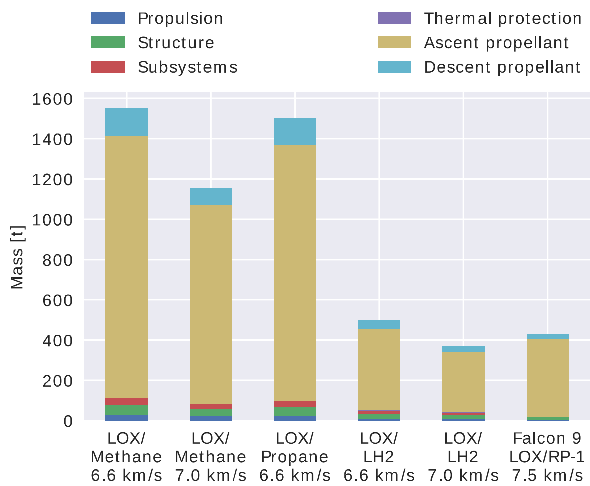


Figure 9: First Stage Mass breakdown of LOX/LH2, LOX/LCH4, LOX/LC3H8 DLR launchers and Falcon 9 (SES 10 mission)

The launchers' first stage dry mass breakdown is presented in Fig. 10. Even though the Falcon 9 has a higher propellant mass than the hydrogen launcher with an upper stage Δv of 7.0 km/s, the total dry mass is lower. As explained previously, this was achieved by using subcooled propellants and lightweight materials such as advanced manufacturing procedures. The greatest share of dry mass is posed by structure, followed by propulsion and the remaining subsystems. Remarkably, the recovery hardware (grid fins & landing legs) mass reaches up to 15 tons for the LOX/methane launcher, thus accounting for about 12.5% of the total dry mass. As explained previously, a more detailed structural model of the landing legs shall be established in the further course of the study.

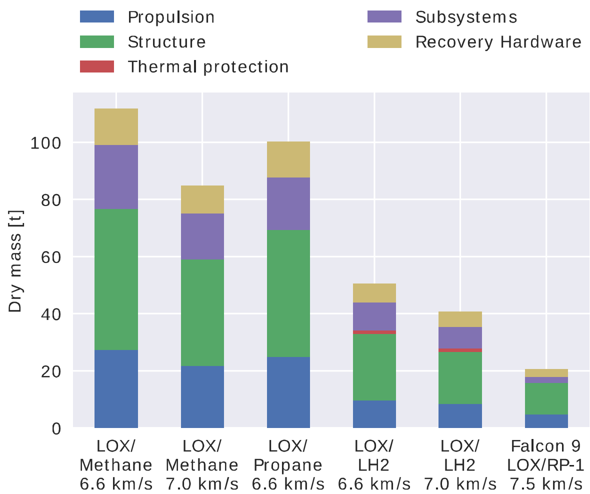


Figure 10: First stage dry mass of LOX/LH2, LOX/LCH4, LOX/LC3H8 DLR launchers and Falcon 9 (SES 10 mission)

Fig. 11 shows the respective dry mass breakup of the second stages. The second stages with 6.6 km/s and 7.0 km/s upper stage Δv are almost equal in dry mass, although the Δv = 7.0 km/s stages carry more propellant. This can be explained by the disadvantageous layout of the Δv = 6.6 km/s second stage with short tanks and comparable big domes and the heavy, overpowered engine being derived from the powerful first stage engines. These reasons combined with the fact that the first stages carrying the 6.6 km/s Δv upper stages are 20% heavier than the first stages with upper stage Δv = 7.0 km/s leads to the high GLOMs of those respective launchers.

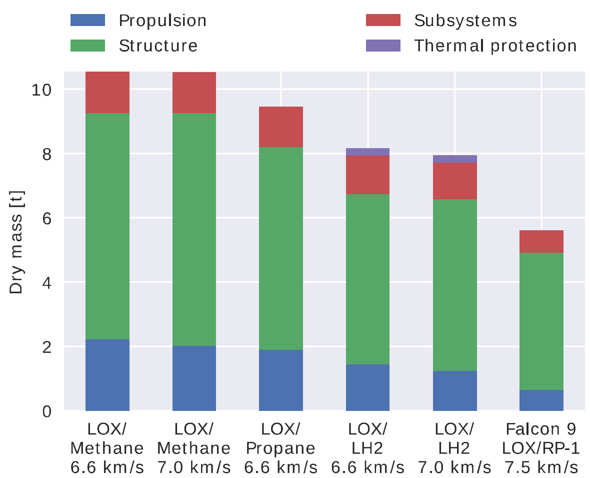


Figure 11: Second Stage dry mass of LOX/LH2, LOX/LCH4, LOX/LC3H8 DLR launchers and Falcon 9 (SES 10 mission)

The descent trajectories of the first stages show a good compliance and consistency with each other (see Fig. 12). However, the hydrocarbon first stages generally have a slightly lower re-entry velocity than the respective LOX/LH2 stage with the same second stage Δv. This can be explained by the difference in ballistic coefficient which is lower for the hydrogen launchers. Hence, the LOX/LH2 launchers are able to reduce more Δv by aerodynamic deceleration. The re-entry burn of the hydrocarbon launchers (including Falcon 9) start in an altitude of around 50 km while it is favorable to start the hydrogen launchers' burn in an altitude of up to 80 km. This can partly be explained by the difference in ballistic coefficient and partly by the different trajectory profile in general (see Fig. 12).

It is important to note that in the first design loop with dry masses based on SI assumptions, the optimum upper stage Δv was observed to be tending towards 6.6 km/s. Nevertheless, with a change of the boundary conditions during descent and the refinement of the structural and mass estimation model, an upper stage velocity of 7.0 km/s appears to be more of advantage now.

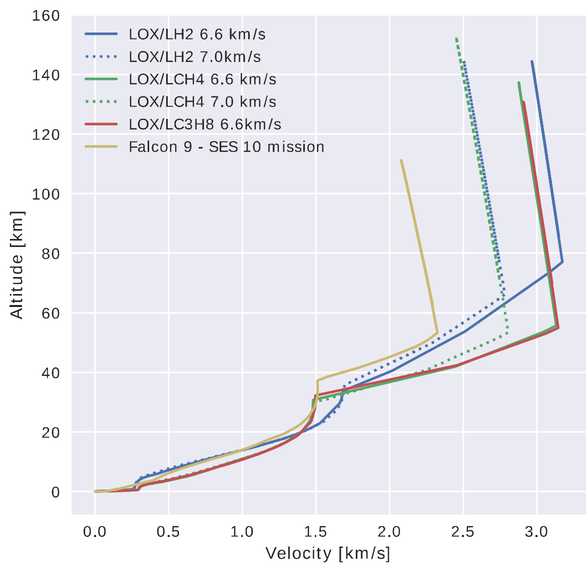


Figure 12: Re-entry trajectories of DLR LOX/LH2, LOX/LCH4 and LOX/LC3H8 DLR launchers and Falcon 9 (SES 10 mission)

5 CONCLUSION & OUTLOOK

The first design loop showed that RTLS landings for GTO missions are of low economic interest and lead to very high launcher masses. However, considering downrange landings on a barge, the launchers have reasonable sizes and masses. The comparison of different propellants and engine cycles showed that LOX/LH2, LOX/LCH4 and LOX/LC3H8 launchers with reusable VTVL booster stages and gas generator engines have the potential to be viable options for future European reusable launchers. As expected, using LOX/LH2 results in the lightest launchers followed by LOX/LC3H8 and LOX/LCH4 with around three times higher GLOMs compared to LOX/LH2. While the methane and propane launchers remain close in size, the advantage of propane actually increased when compared to the first iteration of launchers. Furthermore, propane offers more potential for densification compared to methane [8].

A low second stage Δv, and thus a higher velocity at MECO, leads to a higher share of the first stage's mass with respect to the launcher's GLOM. Hence, the proportion of the launcher being recovered and reused is greater. Nevertheless, a higher velocity at MECO also increases the demand of descent propellant, since more velocity has to be reduced by retropropulsion to keep the aerothermal loads within the designated limits. The results show that the launchers with an upper stage Δv of 7.0 km/s are around 20% lighter than the 6.6 km/s launchers and thus more efficient. Hence the optimal second stage Δv for GTO missions appears to be in the order of 7.0 km/s considering launcher design, masses, re-entry loads and probably costs.

In the future course of the study, more launchers will be subjected to the preliminary design loop including launchers using LOX/RP-1 as propellants and launchers using the staged combustion cycle engines shown in section 3.1. Furthermore, further detail will be put into the structural modelling and mass estimation of recovery hardware components (landing legs and grid fins). Also, the direct flight into a GTO with a correct argument of perigee with 0° or 180° shall be considered.

Considering the re-entry loads, a detailed analysis of those loads and their impact on the structural design, the design of the baseplate and the engines during the engine-first approach is necessary. Preliminary CFD calculations were conducted at DLR to determine the heat intake on the baseplate and the tanks during retropropulsion but these calculations have to be refined to also consider the heat intake into the engine nozzles and the re-entry loads during the aerodynamic phases [10]. Nevertheless, without actually flying the respective hardware, those re-entry loads are difficult to assess and are thus subject to ongoing reevaluation during the study. Hence, it is considered a necessity to build and fly RLV demonstrators to gather a better understanding of both VTHL and VTVL technologies. Currently, two projects are running at DLR, with CALLISTO (Cooperative Action Leading to Launcher Innovation in Stage Tossback Operations) [11] in cooperation with CNES representing the VTVL method and ReFEx (Reusability Flight Experiment) representing the VTHL method [12].

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