Investigation of a European Reusable VTVL First Stage

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2 authors:

Jascha Wilken
German Aerospace Center (DLR)

Sven Stappert
Polaris Raumflugzeuge

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Jascha Wilken and Sven Stappert**
** German Aerospace Center (DLR), Institute of Space Systems, Robert-Hooke-Straße 7, 28359 Bremen, Germany*
jascha.wilken@dlr.de

Abstract

Based on a broad evaluation of possible Semi-RLV configurations with a vertical take-off, vertical landing (VTVL) first stage, a two staged concept with a methane fueled first stage and hydrogen fueled second stage was selected for further investigation. A number of parameter studies were undertaken to identify optimal design trends specific to VTVL-stages. These parametric studies and their results are the core of this paper. Based on the results a robust initial design was developed and will in further steps be investigated on a subsystem level in order to promote the understanding of reuse-related technologies within Europe.

Abbreviations

ASDS	Autonomous Spaceport Droneship
DRL	Downrange Landing
ELV	Expendable Launch Vehicle
FB	Flyback
GLOM	Gross Lift-Off Mass
IAC	In-Air Capturing
Isp	Specific Impulse
RLV	Reusable Launch Vehicle
RTLS	Return to Launch Site
VTHL	Vertical Takeoff, Horizontal Landing
VTVL	Vertical Takeoff, Vertical Landing

1 Introduction

Reusability of launch systems will strongly impact the launch service market if certain characteristics such as sufficient reliability and low refurbishment costs can be achieved. The German Aerospace Center (DLR) is performing a systematic investigation of return methods for a reusable first stage of a future European launch vehicle. The final goal is the determination of the impact of the different return methods on a technical, operational and economical level and the assessment of their relevance for a future European launch system. Within the first phase (called ENTRAIN 1 [1]) a wide variety of reusable first stages was investigated by the DLR. Fuel (LH2, LCH4, LC3H8, RP1), staging velocity, engine cycle (gas-generator and staged-combustion) and return modes (VTVL with downrange landing on a barge (DRL) or return to launch site (RTLS) and VTHL with In-Air Capturing (IAC) or flyback (FB)) were varied in order to identify the most promising combination for a future partially reusable European launch vehicle. Some of the launchers generated within ENTRAIN 1 are shown in Figure 1.

At the end of this first phase, promising combinations for both VTVL and VTHL configurations were selected. These are evaluated in more detail within the ENTRAIN 2 study in order to arrive at realistic designs for potential future reusable vehicles. The current results from this second phase will be the focus of this paper, specifically for the VTVL configuration, hereafter referred to as the ENTRAIN-VL. It is composed of a methane-fueled first stage and hydrogen fueled upper stage. The two main goals are, first, the identification of design trends for RLV type systems, since these can differ significantly from the known ELV trends, by varying a range of design parameters. Secondly, the generation of a robust initial reference design for the future analysis of various subsystems with sophisticated and computationally expensive methods. This planned future work is described in chapter 4.3.

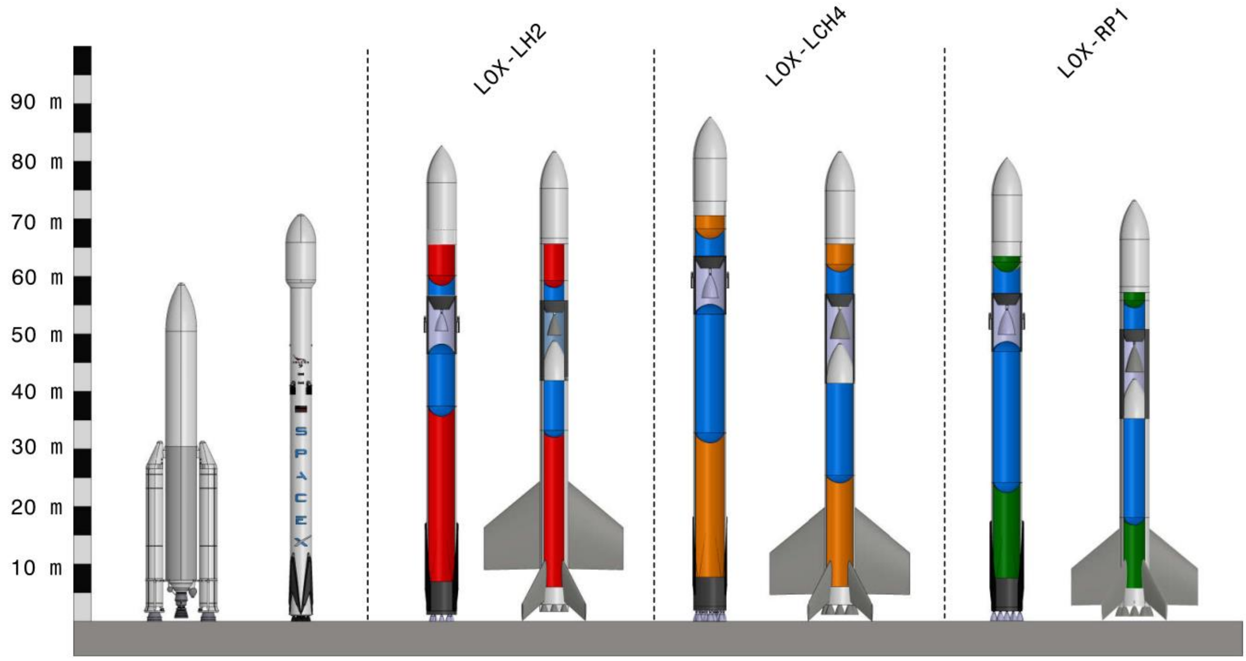

Figure 1: Selection of Semi-RLV configuration that were investigated in the ENTRAIN 1 study. Blue tanks contain LOX, red tanks LH2, orange tanks LCH4 and green colored tanks RP1

2 Methods

In general, the following procedure was used to identify the optimal design parameters: Starting with the initial design, the chosen parameter was varied and the effects on the payload performance and system were evaluated. However, full design iterations would result in a large amount of effort that would severely limit the number of design parameters that could be investigated. Accordingly, secondary effects were usually neglected and only the portions of the design reevaluated that were directly affected by the change. Thus the values far from the reference design have to be used with care since the effect of these secondary effects will be larger for those points. For each parametric study these secondary effects are noted and discussed alongside the results within the following chapter. The software tools and modelling methods used for the various relevant subsystems are described in [1].

The following chapter contains the result for the studies of the following parameters:

- Thrust-to-weight ratio of both stages
- Upper stage propellant loading
- First stage dry mass
- Length-to-diameter ratio of entire launcher
- Maximum allowed heat flux during descent

2.1 Initial reference design

The reference design on which the following parametric studies are based is a derivation of the two staged launchers with a reusable VTVL first stage presented in [1], with two changes: The propellants are methane in the first and hydrogen in the second stage and the overall size was scaled down. It is assumed that payloads heavier than 5.5 t are launched in ELV Mode, thus the overall size could be reduced. The boundary conditions and methods used for the design of the stage can be found in [1]. The engines are based on the design described in [4] with one slight change: The expansion ratio of the first stage engines was increased from 20 to 22.
Since this reference design is an inherently outdated step in the design process the stage will not be discussed in detail here. Instead the final design including the insights from the following results will be shown and discussed in section 4.2.

Table 1: Main parameters of the initial reference design

GLOM (GTO-ASDS)	485 t
GLOM 1st stage	411 t
GLOM 2nd stage	68 t
SI 1st stage	8.7 %
SI 2nd stage	11.9 %
Stage diameter	5.0 m
Payload GTO - ASDS	6.3 t
Upper stage engine Isp, vac	444 s
Upper stage engine mass flow	142 kg/s
First stage engine Isp, ground and vac	288.0 s 322.5 s
First stage engine mass flow	9 * 265 kg/s

2.2 Metrics of comparison

The comparison of the results of the parametric studies are trivial if only the achievable payload mass is considered. However, the goal of the design optimization has to be the minimization of cost per kg of payload. The use of this metric is severely hampered by the fact that the cost estimation of space transportation systems is a notoriously inexact undertaking and prone to large errors. This is especially true for orbital RLV-type systems where no practical experience exists within Europe and only limited experience exists in the US. While it is assumed that companies such as SpaceX and Blue Origin have data and insight into the cost drivers for development, operation and refurbishment of VTVL reusable first stages, this data is not published.
In order to avoid the quagmire of cost estimation, alternative metrics of comparison are developed and while the details differ for each parametric study the basis is always the following assumption: For liquid fueled systems the dry mass is the driver for the development cost and also has a considerable impact on the production and refurbishment costs. This is also one of the basic principles of the TRANSCOST [2] cost models. Since the raw total dry mass of the systems fails to account for some factors (the price difference between engine mass and stage mass or the reuse of the first stage) the dry mass is adapted as necessary for each parametric study. The exact method this basic assumption is implemented will be explained for each parametric study to clarify the methodology used. While this method is a simplification and neglects many factors that influence the final cost of a system it does enable the comparison of the various results of each parametric study with regard to costs.

3 Parametric studies

The following sections each focus on one specific parametric variation. For each study first the underlying boundary conditions and simplifications will be described. Secondly, the raw results will be shown and finally the logic described in section 2.2 is used to gain further understanding of the results with regard to a cost-effective optimum.

3.1 Thrust-to-weight ratio of second stage

This parametric investigation focuses on the optimal thrust-to weight ratio (T/W) of the second stage. Since the two stages of the ENTRAIN-VL concepts use different propellants and engines, the T/W of the second stage can be chosen independently from the first stage. The mass flow of the upper stage engine was varied and the thrust and mass of the engine changed accordingly.
The structure mass of the upper stage was not reevaluated for this parametric study. While it is expected that the thrustframe will be impacted, for large parts of the remaining structure the loads are highest during the flight of the first stage. These are, in the reference case, the trajectory phases with the highest accelerations and aerodynamic forces. However, if the thrust is increased dramatically as a result of this study, this assumption should be reevaluated. The propellant needed for engine chilldown and ignition was scaled with the engine mass flow.

3.1.1 Results

The reference T/W was ~0.84. For this study the values were varied from 0.76 to 0.96. The ascent trajectory was reevaluated for each of those points. The effect on the payload performance is shown in Figure 2.

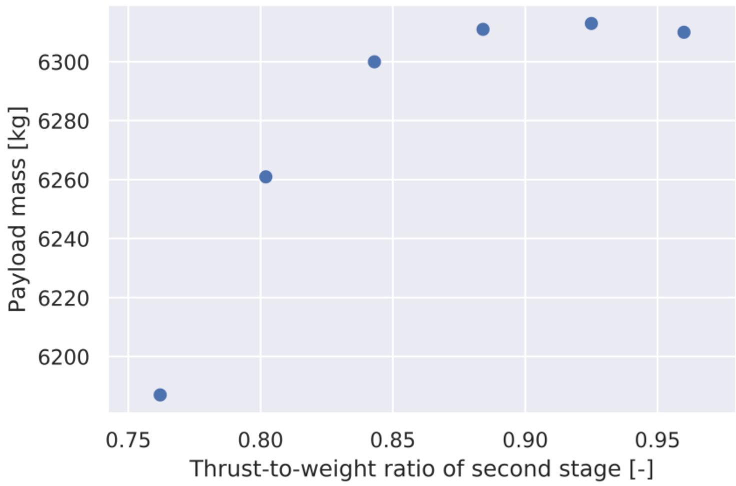

Figure 2: Payload mass over thrust-to-weight ratio of second stage

The trend of results is plausible: Up to a certain point an engine size increase leads to higher payloads but at some point the higher engine mass negates any further improvement and will even cause a payload reduction.
As can be seen in Figure 2 the payload varies only slightly between the shown T/W values. The difference between the highest and lowest payload is only ~2%. However, the negative slope on the left side is quite steep; reducing the T/W even further will be associated with increasingly larger performance losses.

3.1.2 Analysis

The core problem with the raw results shown above is that while it can be shown that in certain cases a larger upper stage engine can result in a higher payload performance; it does not consider the additional cost of the heavier engine. As mentioned in section 2.2 the dry mass is chosen as a replacement value for the cost. In order to account for the fact that the engine mass is usually more expensive than the comparatively simple stage mass a factor f_e is multiplied with the engine mass and the total sum, named *adapted dry mass* $m_{dry,ad}$ hereafter, is used for comparison. It is calculated as follows:

$$m_{dry,ad} = f_e m_{engine,1st\ stage} + m_{other\ dry\ mass,1st\ stage} + f_e m_{engine,2nd\ stage} + m_{other\ dry\ mass,2nd\ stage}$$

For this approach to be meaningful, the engine cost factor f_e has to be known. The TRANSCOST [2] model implicitly contains a factor of 3-3.5 when comparing the mass specific development costs of rocket engines to the mass specific development cost of ELV stages. It should be noted that this range can only be seen as a rough indicator and thus it was varied parametrically in order to assess its impact. For RLV the value of f_e is generally lower since the TRANSCOST model, understandably, estimates the RLV stages to be more expensive to develop and produce than ELV with the same dry mass. Depending on the type of RLV stage the f_e factor can even drop below 1.

As can be seen in Figure 3, a clearer optimum appears when employing the post processing steps described above. The figure shows the results for engine factors from 1 to 4 and in all cases a thrust-to-weight ratio of 0.85 to 8.75 appears close to optimal. Just as with the original raw results the overall magnitude of the impact stays small with a much steeper left than right slope.

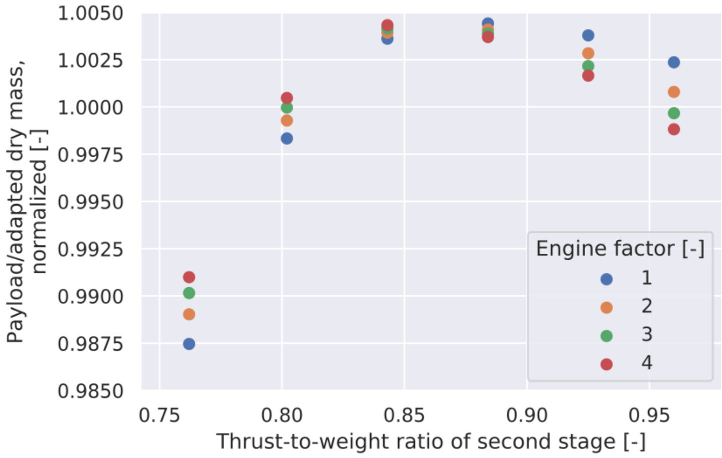

Figure 3: Normalized payload to adapted dry mass ratio over thrust-to-weight ratio of the second stage

3.2 Thrust-to-weight ratio of first stage

This parametric investigation focuses on the optimal thrust-to weight ratio of the first stage. For this purpose the mass flow of the first stage engines was increased along with their mass and thrust. Similar to the same parametric variation for the second stage, the structural design was not reevaluated. Since the maximum dynamic pressure experienced during the flight does increase with the thrust-to-weight ratio of the first stage, this simplification is expected to have a larger impact in this case than for the second stage. The impact of this is discussed in section 3.2.2. Only the ascent was evaluated for every variation, the descent propellant budget was kept constant.

3.2.1 Results

The results from this parametric variation are shown in Figure 4 below. The payload is given for each investigated T/W value of the first stage. The value for the reference configuration was 1.42. In total values from 1.35 to 1.5 were evaluated.

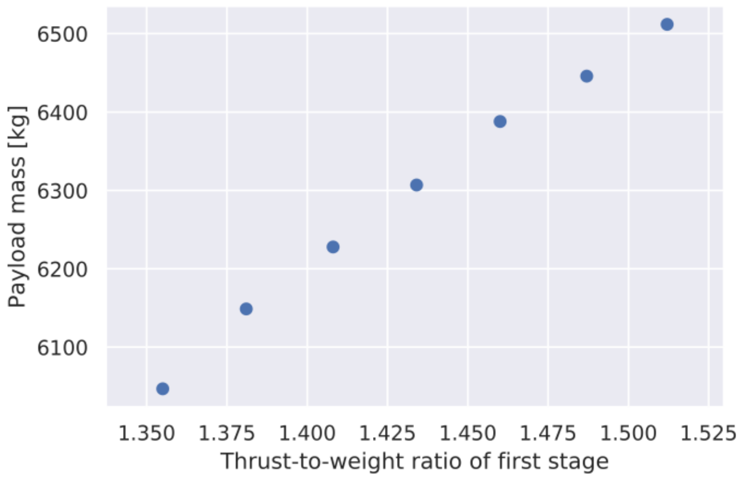

Figure 4: Payload over thrust-to-weight ratio of first stage

The impact of the T/W of the first stage is much larger than of the same parameter for the second stage which is feasible since the majority of the Δv losses occur during the flight of the first stage, especially the gravitational and aerodynamic losses. Since the T/W has a large impact on the magnitude of these losses this leads to a larger impact on the performance. The general trend of the results, that a higher thrust will increase the payload mass, is unsurprising, especially since the impact on the structural weight was not considered.

3.2.2 Analysis

Again, the raw results shown above can't account for the impact of the higher cost of larger and heavier engines. In order to evaluate this aspect, as done for the second stage in section 3.1.2, an adapted dry mass is created with an engine factor to account for the assumed higher complexity and cost of the engine mass in comparison to the comparatively simple stage mass. As above the adapted dry mass is calculated with

$$m_{dry,ad} = f_e m_{engine,1st\ stage} + m_{other\ dry\ mass,1st\ stage} + f_e m_{engine,2nd\ stage} + m_{other\ dry\ mass,2nd\ stage}$$

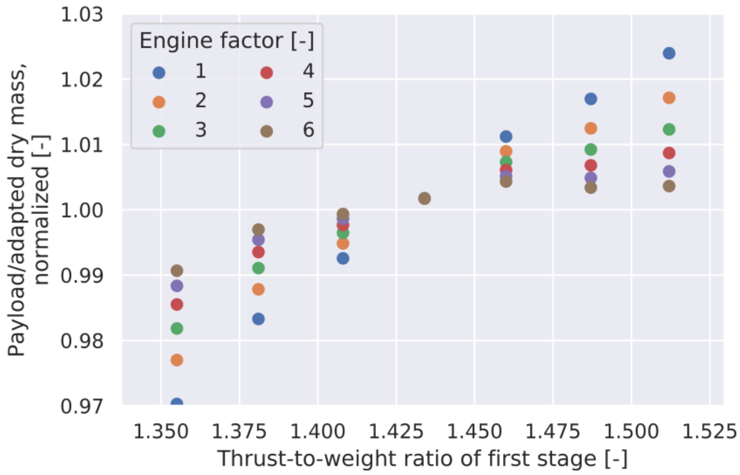

Figure 5: Normalized payload to adapted dry mass ratio over thrust-to-weight ratio of the first stage

This result of this post processing is remarkable because, contrary to the results for the second stage, it indicates that the trend of higher thrust-to-weight ratios being better is only reversed for extremely high values of f_e. Considering that the engine factor f_e can be lower than 1 for RLV stages, this indicates that the thrust-to-lift off ratio should be as high as feasible for VTVL-RLV stages. Bearing in mind that VTVL RLV stages have a much higher total Δv requirement than equivalent ELV stages this seems plausible: The reduction of gravitational losses during ascent yields additional benefits for the higher Δv RLV-missions.

Some caveats have to be considered for the result shown above: Since the structural mass is not recalculated at every step, the higher dynamic pressure that results from higher T/W at launch is not considered. This does not necessarily mean a higher structural mass; it could also be curtailed by throttling the engine prior to the maximum dynamic pressure. Since the first stage is designed to perform vertical landings, the engines already inherently are required to be throttleable. Another constraint that was not considered is the T/W at landing, which should at minimum throttling be close or even below to unity to allow hovering at landing. However, in the final design this can be handled by varying the number of engines in the first stage while keeping the thrust constant.

Considering the large impact of this parameter, future investigation is needed into boundaries that dictate the current T/W levels. These range from the aforementioned structural consideration, the feasibility of engines of a certain size and even fairly mundane aspects such as the available space in the vehicle aft bay.

3.3 Variation of upper stage propellant loading

Another key parametric variation was the variation of the upper stage propellant loading from the nominal value of 59 t. This essentially results in the variation of the staging of the launcher. While this was investigated within ENTRAIN 1, the large number of configurations under investigation made it infeasible to investigate the staging in small step sizes. Within this study ascent and decent trajectories were reiterated and the first stage propellant budget for the reentry maneuver adapted according to the new staging velocities. The structural index and the T/W of the upper stage were kept constant. The thrust of the first stage was also kept constant. This leads to a small shift in T/W at Lift-off, however since the hydrogen upper stage is much lighter than the methane first stage the impact is small.

3.3.1 Results

The payloads that resulted from this study are shown in Figure 6:

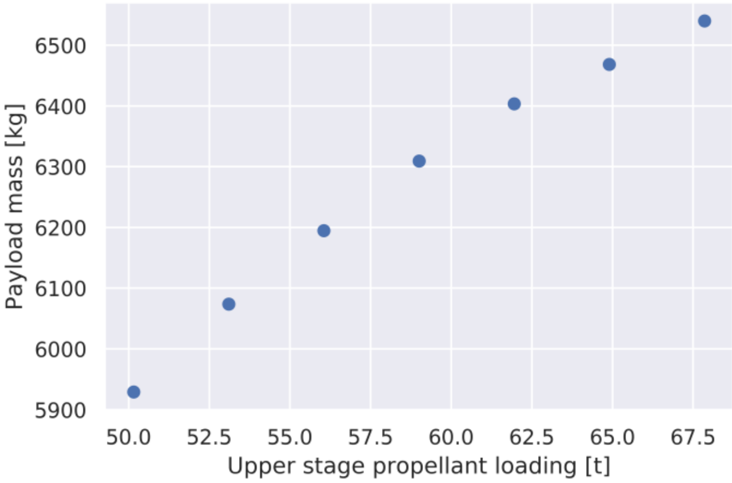

Figure 6: Payload mass over upper stage propellant loading

The achievable payload increases markedly with the increasing upper stage size. In theory there is a point in upper stage size where the trend would reverse and the payload would decrease with additional upper stage propellant, because any additional propellant would lead to a suboptimal staging thus negating any benefits.

3.3.2 Analysis

As with the previous parametric investigations, simply looking at the payload performance of the variations is not sufficient since it neglects the additional cost of a larger upper stage with a larger dry mass. For an RLV another, even more critical, aspect emerges: The second stage is only used once and by increasing its size the reusable portion of the vehicle is decreased. To what degree a higher payload can be worth an overall larger dry mass and smaller reusable fraction depends on the number of reuses and the costs associated with reuse. Again, because of the problems discussed in section 2.2 an alternative metric is used instead of direct cost estimations. In this case the payload is seen relative to the total expended dry mass for each mission:

$$m_{dry,exp} = \frac{m_{dry,1st\ stage}}{n} + m_{dry,2nd\ stage}$$

In the equation above n signifies the effective number of uses of the first stages. Thus for an ELV n will always be unity since the entire dry mass is expended. By variation of this parameter the impact of the reuse costs can be represented in a simplified manner. This is done by decreasing the effective number of reuses. For example, if a stage is used six times but the cost of recovering and refurbishment is one fifth of the cost of producing a new stage, this would result in five effective uses since one “use” is consumed for the cost of refurbishment and recovery. Since these costs are unknown, the number of effective uses is varied broadly. The results gained this way have to be seen qualitatively and are used here mainly to gain an understanding of the robustness of the staging selected.

The results based on the deliberations above are shown in Figure 7. As a maximum number of uses six was selected. This results from the assumptions that high payload GTO missions are serviced in ELV mode. Thus, based on the mission scenario used as the baseline, every sixth launch will be in ELV mode and thus the stages can be reused five times on average. As mentioned above, the recovery and refurbishment cost can be accounted for by reducing the effective number of uses.
The general direction of the results is trivial: The more reuses are assumed, the more beneficial a smaller upper stage becomes. The difference in propellant loading between the two optima for the extreme cases is quite large. Since the costs for recovery and refurbishment cannot easily be estimated with any accuracy it seems prudent to take a staging that works reasonably well for the entire range of three to five reuses. Thus the upper stage loading will be slightly reduced, compared to the reference design loading of 59 t.
It is also noteworthy, that the impact of the staging on the adapted payload ratio decreases with more reuses: The impact of staging is largest for the ELV case and is smallest for the largest number of reuses.

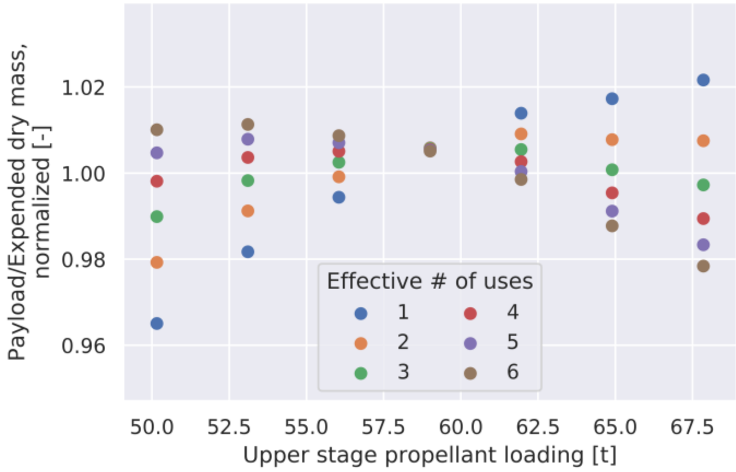

Figure 7: Normalized ratio of payload to expended dry masses over upper stage propellant loading and for different numbers of uses

3.4 Impact of first stage dry mass on payload mass

For an upper stage the impact of additional dry mass on the payload mass is trivial: Every kg of second stage dry mass costs one kg of payload. For any other type of stage the impact depends on the number of stages, the staging of the entire system itself as well as a number as other factors such as the propellant combinations et cetera. For the first stage of the ENTRAIN-VL configuration this value had to be determined through variation of the first stage dry mass. For the sake of brevity the results are not shown herein, it will merely be noted that an additional ~7.5 kg of additional first stage dry mass results in 1 kg payload performance loss. This value is used for the interpretation of the following parametric study.

3.5 Length-to-diameter ratio

The L/D of the launcher is an important design aspect that has an impact on structural design and the aerodynamics of the stage. In reality this parameter is often decided by external factors and not chosen for optimal performance. Such external factors can be already existing manufacturing equipment, limitations imposed by the chosen logistical infrastructure or limitations of available testing equipment. Within this study all such boundary conditions were ignored and only the impact on the structure considered. While the diameter obviously also has an influence on the aerodynamic forces, the overall impact of these on the performance of launch vehicles is small (~6% of the loss during ascent of the initial reference configuration), so that this was considered a secondary effect and not evaluated for each case.

3.5.1 Results

For different length-to-diameter ratios the length of all tanks and structural segments was recalculated and the structural design including the number of stringers and frames optimized. The resulting mass of the structural elements of both stages is shown in the figure below.
From the raw results shown in Figure 8, it seems beneficial to increase the stage diameter further. However, the mass trend is different for upper and lower stage: While the lower stage profits from the diameter increase it leads to a structural mass increase for the upper stage. The structure mass decrease of the lower stage is caused by two factors: First the overall bending moments are reduced since the stage length is decreased. Additionally the larger diameter leads to increased area moments of inertia of the overall structure. While the same things also benefit the second stage, the high diameter leads to small cylindrical portions and thus to reduced structural efficiency for the propellant tanks.

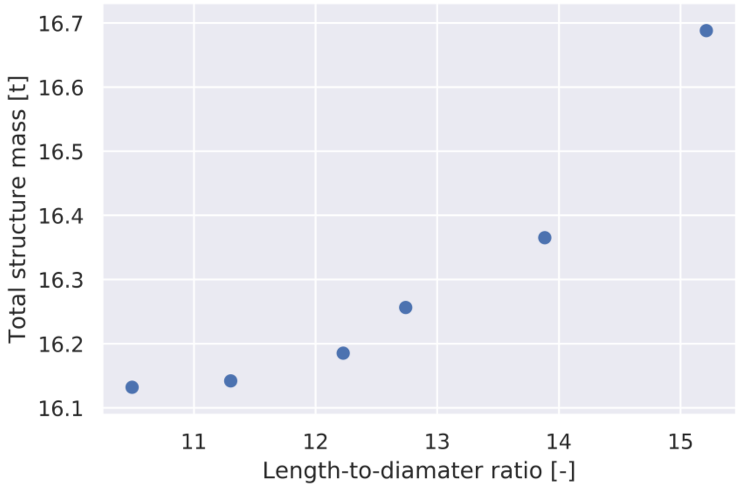

Figure 8: Structure mass of both stages over length-to-diameter ratio

3.5.2 Analysis

Using the results from section 3.4 the values shown above are used to calculate an adapted structural mass where the upper stage structural mass is weighted according to its impact on the payload performance:

$$m_{str,ad} = m_{str,1st\ stage} + f_s m_{str,2nd\ stage}$$

This adapted structural mass allows the selection of a performance optimal length-to-diameter ratio.

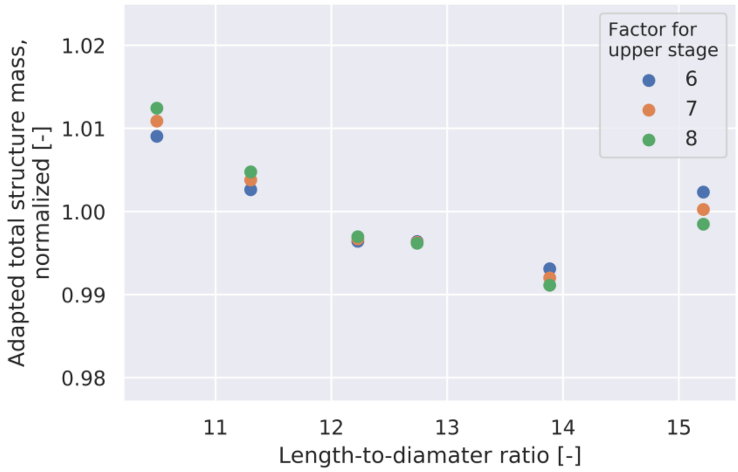

Figure 9: Adapted total structure mass over length-to-diameter ratio

The resulting values for the adapted structural mass are shown in Figure 9. As indicated above, the dry mass of the upper stage has a 7-8 times increased impact on the payload, when compared to the dry mass of the first stage. Thus the results indicate that a length to diameter ratio of slightly less than 14 is the best from a structural perspective.

3.6 Maximum heat flux during descent

For this parametric variation the maximum allowed convective heat flux $\dot{q}_{max}$ during descent was varied in order to assess its impact on the payload performance. The estimation of this value is done via a modified Chapman equation, described in [3], with respect to a nose radius of 0.5m. For each new value of $\dot{q}_{max}$ the descent propellant budget was reiterated until a new converged solution was found for a new payload. For all of the cases shown below the heat flux remained the main constraint for the descent, none were constrained by the dynamic pressure.

3.6.1 Results

As discussed in [3] the reference value for $\dot{q}_{max}$ was 200 kW/m². Within the parametric study values between 140 kW/m² and 220 kW/m² were evaluated. The results are shown in Figure 10:

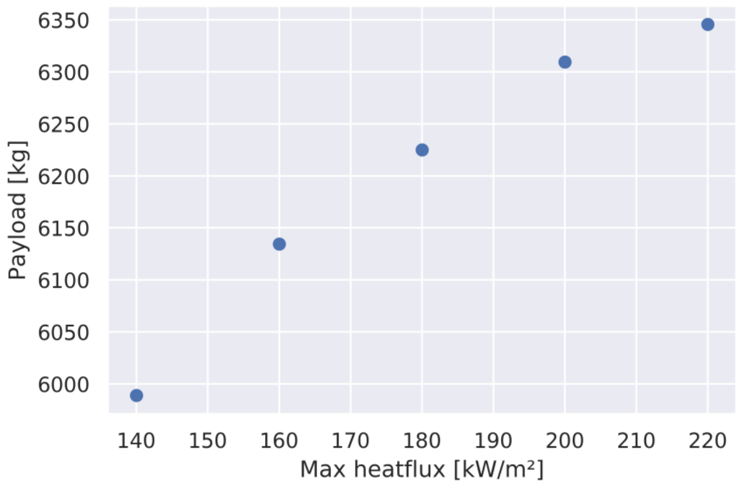

Figure 10: Payload over maximum allowed heatflux during descent

As expected a decrease in $\dot{q}_{max}$ causes an increase in the descent propellant budget, which in turn decreases the ascent propellant budget and thus the possible payload performance. The slope of the data points grows flatter the higher the values for $\dot{q}_{max}$ are. The curve should, in theory, be approaching a maximum where no reentry burn is necessary. Before that happens, the maximum allowable dynamic pressure would turn into the constraining factor and negate any possible performance increase through the increase of $\dot{q}_{max}$.

3.6.2 Analysis

Contrary to the previous subsection no additional models were used to assess the results of this parametric study. In theory the heat loads for every case can be used to calculate the mass of the necessary thermal protection systems and the new mass could be factored into the system mass and an optimal solution could be found. However, the design of the thermal protection systems for this type of systems is not trivial and necessitates computationally expensive, high resolution CFD calculations of the base plate area just to arrive at the local heat flux. The impact on the nozzles and their thermal protection needs also have to be considered. The improvement of the understanding of the design drivers of such subsystems is one of the core goals of the ENTRAIN 2 study, which will allow a retrospective reassessment of these results.

While a change in $\dot{q}_{max}$also results in a change of the dynamic pressure experienced by the stage, the impact on the structural design is assumed to be small. Analysis done on the structural design of the ENTRAIN-VL has shown that while small scale reinforcements at critical points will be necessary, the large scale loads during descent are smaller than during the main ascent load cases. For a similar VTVL configuration this is shown in [5]

For now the reference $\dot{q}_{max}$was reduced to 140 kW/m² in order to lower the requirements for the thermal protection system and increase the robustness of the design. The payload impact of this is deemed acceptable. For an actually operational vehicle the experience gained with each flight would allow for an extension of the envelope to its limits, as SpaceX has done with the various design iterations of its Falcon 9 first stage.

4 Conclusion

4.1 Scope of results

The nature of the parametric investigation shown above means that while a large number of variations can be investigated, not all degrees of freedom are investigated simultaneously and thus the final design found can still not be considered truly optimal. It is important to note that for the ENTRAIN 2 study a robust initial design is of far greater importance than an optimal one. Especially considering the uncertainties attached to estimating masses for subsystems or system that have never been developed or built in Europe. Nevertheless, the trends do allow general observations on possible design guidelines for RLV and the differences to ELV concepts.

4.2 Final design

Based on the results from the parametric studies shown above the following reference configuration was generated using the same methods described in [1]. The geometry and size of the resulting launcher is shown in Figure 11 and Figure 12:

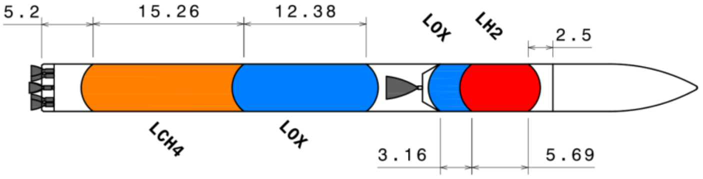

Figure 11: Sketch of ENTRAIN-VL layout

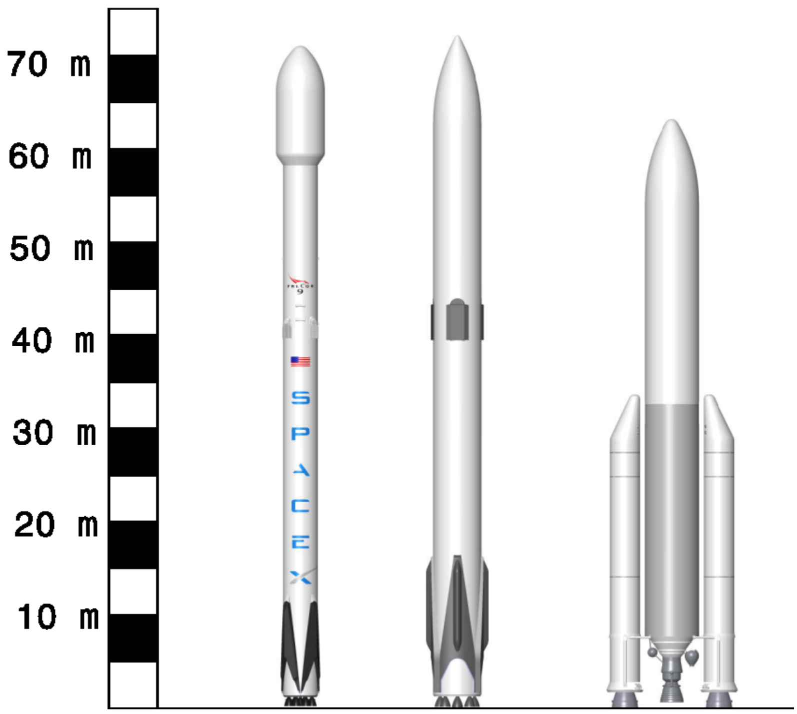

Figure 12: Size comparison of the ENTRAIN-VL to currently operational launchers

The ascent and descent trajectories of the ENTRAIN -VL for various mission types can be seen in Figure 13 and Figure 14. The performance into SSO and LEO orbits, as shown in Table 2, is beyond any currently planned missions and would enable the system to fly multiple payloads into those orbits. For the SSO missions it can be seen that the upper stage is brought directly into the desired circular orbit, this necessitates a comparatively steep ascent trajectory which is beneficial in the SSO-RTLS case since less horizontal velocity has to be neutralized.

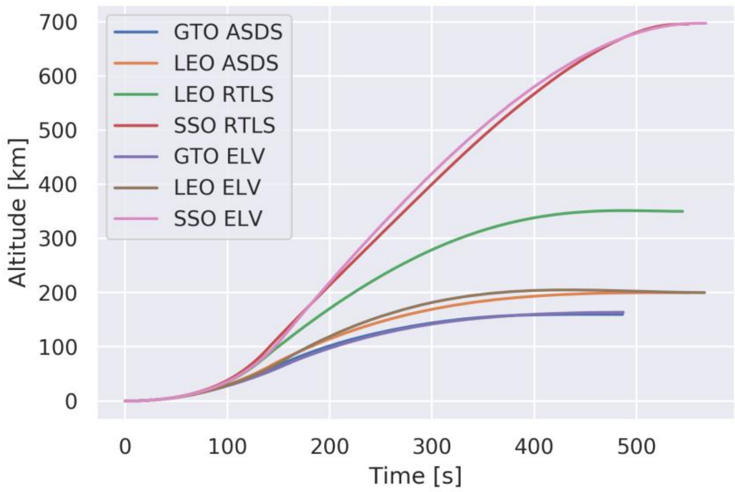

Figure 13: Ascent trajectories of the ENTRAIN-VL configuration for various target orbits

For the descent trajectories shown in Figure 14 the maximum allowed $\dot{q}_{max}$ was set at 140 kW/m². For the LEO-RTLS missions, the initial conditions are so benign, that this boundary is never actually reached. For the RTLS configurations it can be seen that the initial boost-back burn accounts for most of the propulsive velocity change.

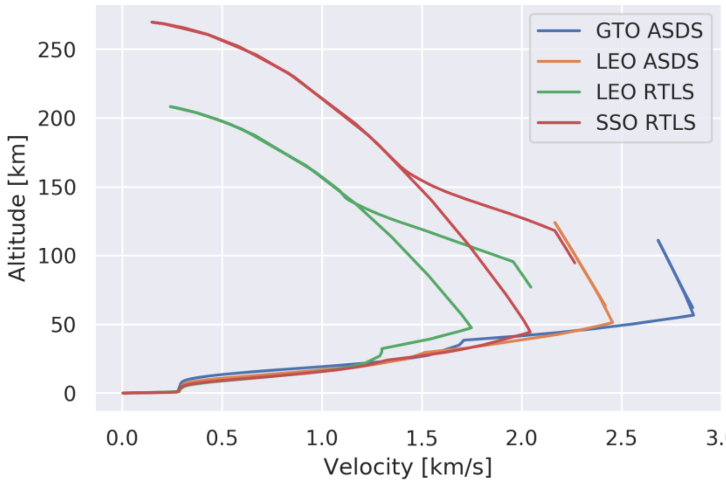

Figure 14: Descent trajectories of the ENTRAIN-VL configuration for various mission types

Table 2 contains the main design parameters of the current ENTRAIN-VL configuration including the changes based on the results of the abovementioned parametric studies. Compared to the initial reference design shown in section 2.1, the total mass of the launcher and the stages only changed minimally, however the payload actually decreased slightly. Considering that one goal of the parametric investigations was the optimization of the design, this is counterintuitive. This is caused by two decisions that were made in order to increase the robustness of the design: Firstly, as described in section 3.6 the $\dot{q}_{max}$ during descent was reduced and thus additional fuel is necessary for the propulsive deceleration. Secondly, within the structural models, additional masses were considered for the connection elements of the individual segments. While these won't be needed if certain technologies such as stir friction welding are used, they were included since the details of the structural design have not yet been frozen. These two factors negatively impacted the payload performance of the launcher so that even with the improvements based on the parametric studies, the performance decreased. However, without these improvements the launcher size would have to be increased to achieve the desired payload.

As can be seen from the performance data shown in Table 2 the payload performance is roughly comparable to the Falcon 9, with the Falcon 9 being slightly heavier in total but also slightly more performant. At first glance this is puzzling since the Falcon 9 uses a propellant combination with a lower specific impulse in both stages which should impact the performance heavily. However, this can be explained by the impressively low structural indices both Falcon 9 stages exhibit.

Table 2: Main parameters of the ENTRAIN-VL configuration

GLOM (GTO-ASDS)	485 t	
GLOM 1st stage	411 t	
GLOM 2nd stage	67 t	
SI 1st stage	9.0 %	
SI 2nd stage	12.3 %	
Stage diameter	4.8 m	
Payload GTO - ELV	8 t	
Payload GTO - ASDS	5.5 t	
Payload LEO - ELV	>20 t	
Payload LEO - RTLS	>10 t	
Payload SSO - ELV	10.5 t	
Payload SSO - RTLS	5 t	
Upper stage engine Isp, vac	444 s	
Upper stage engine mass flow	146 kg/s	
First stage engine Isp, ground and vac	288.0 s	322.5 s
First stage engine mass flow	9 * 265 kg/s	

4.3 Future Work

The ENTRAIN-VL and the, not discussed herein, ENTRAIN-HL configurations are now the basis for detailed investigations by the specialized DLR institutes. This includes but is not limited to the generation of aerodynamic and aerothermodynamic databases, detailed structural designs, analysis of controllability, 6-DOF trajectory simulations as well as additional investigation of recovery hardware such as thermal protection systems, landing legs or aerodynamic control surfaces. This work is currently ongoing and will be published in due time.

5 References

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