Advanced TSTO launch vehicles

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Abstract

Different variants of a Two Stage to Orbit (TSTO) launch vehicle, based on technologies available in Europe, combining a solid rocket motor and a cryogenic upper stage propelled by the Vinci engine and aiming at exploiting synergies with other on-going European programs are presented in this paper. After a first sizing based on data from existing upper stages, preliminary structural analyses and tank designs have been performed, considering different first stages, such as P120, P150 and P160. A preliminary optimisation of the first stage thrust law and of the propellant mass of the upper stage has been done.

1. Introduction

Two new launchers, Soyuz and Vega, are scheduled to enter operation in the coming months at Kourou spaceport, increasing the range of missions able to be launched by Western Europe. It is however already foreseen that improvements of Vega will be required to at least partially fill the performance gap with the Soyuz launch vehicle and to allow, for instance, Galileo satellite replacement single launch missions, before the end of this decade. Indeed, the Soyuz launched from the French Guiana Space Centre, with its over 2700 kg payload capacity in GTO, is oversized for such a mission.

However in the longer term, the planned replacement of Soyuz by Rus-M in Russia may imply a halt of the production of Soyuz. In that case Western Europe might have to develop a new launch vehicle with a payload performance comparable to the one of Soyuz (i.e. 2700 kg to 3000 kg in GTO).

As a joint DLR-SART – EADS Astrium effort in the framework of a DLR space agency funded study called VENUS (Vega New Upper Stage), different evolution options of the Vega launch vehicle have been studied. This study considers in particular a 3-stage and a 4-stage configuration based on solid rocket motors for the lower stages and on different storable liquid propellant upper stages. Even if the performance estimations done during this study show a significant improvement compared to the Vega launch vehicle, these configurations are still far from being able to inject a Galileo replacement satellite into orbit.

Another promising concept is currently being investigated by DLR's group for Space Launcher Systems Analysis (SART), namely a two-stage to orbit launch vehicle (TSTO) based on technologies already existing in Europe and aiming at exploiting synergies with other European programs, in order to keep development and production costs low. All the configurations presented in this paper are based on a solid rocket motor for the first stage and a large cryogenic (LOX/LH2) upper stage propelled by the 180 kN Vinci expander cycle rocket engine with its deployable nozzle. The first stage makes use of the carbon-epoxy filament wound monolithic motor case technology developed in Europe for the Vega launch vehicle and is similar to the current Vega P80 FW first stage. The goal is to take advantage of the low structural index allowed by the employed technologies even for high combustion chamber pressure (up to 90 bars) and to be able to produce the motors with the existing facilities in French Guiana.

In 2007 SART has already considered, with the version "F" of the VENUS study, a cryogenic upper stage propelled by Vinci aiming at replacing the current Vega Z23 solid 2nd stage, Vega Z9A solid 3rd stage and the AVUM 4th stage [1]. It was calculated that a P80-H18 (18 tons of propellant in the second stage) configuration is able to put up to 2675 kg into the reference orbit of Vega launch vehicle (700 km, polar and circular). A P100 first stage and a cryogenic upper stage with the optimum propellant loading of 17 tons would have a payload performance over 3000 kg for the reference orbit of Vega, almost 1000 kg for a MTO (250 km x 23216 km, 56°) and slightly less than 950 kg for a GTO (250 km x 35943, 5.4°) [2]. One metric ton in MTO is, however, considered as insufficient to launch a Galileo replacement satellite equipped with the apogee engine necessary for the circularization manoeuvre. As a consequence bigger first stage solid rocket motors are considered. This paper presents the current state of the SART's investigation for such a TSTO launch vehicle.

2. Definition of the launch vehicle

Considering the technologies available in Europe a two stage to orbit launch vehicle appears to be the simplest and most cost effective option for a new launch vehicle. A solid rocket motor is considered for the first stage in order to take advantage of the experience gathered during the development of the Ariane 5 solid rocket booster (EAP) and of the lower stages of Vega. Another solution, like for example a cryogenic first stage, would require the development of a new engine in order to blast-off without strap on boosters. Therefore such a solution is not considered. For the upper stage, the high performances Vinci engine is chosen, as it is the most powerful upper stage engine available in Europe. An optimization of the thrust level was not done, even if it is expected that a thrust higher than 180 kN would be beneficial for the staging [3] and [4]. Indeed, it is known that an increase of the thrust level of the Vinci engine is not possible without a costly and thorough redesign of the engine; in addition the expander cycle implies also a limit in terms of maximum thrust. The characteristics of the Vinci engine considered in this study are summed up in Table 1. The choice of the diameter of the second stage is driven by the diameter of the first stage, itself limited by the existing production facilities. All the configurations presented in this paper have a diameter of 3 m, that is to say the same as for P80 FW. With such a diameter, classical tanks are perfectly adapted to the upper stage and both separate tanks and common bulkhead are considered.

Table 1: Characteristics of the Vinci engine

	Vinci
Vacuum specific impulse [s]	464
Mass flow rate [kg/s]	39.5
Mixture ratio [-]	5.8
Mass [kg]	546

In order to fill in the performance gap between a Galileo satellite replacement single launch mission and the injection into orbit of a 2700 kg to 3000 kg payload in a GTO, two strap-on boosters will be used.

The pre-sizing has been done with the help of DLR-SART software for the design of the feed system, for the estimation of the masses, for the determination of the aerodynamic characteristics and for the optimization of the trajectories. Only GTO (250 km x 35943, 5.4°) missions from Kourou in French Guiana have been considered up to now. The calculated performances into GTO are only slightly lower than those for a MTO mission (for example the injection of a Galileo satellite).

The staging and preliminary sizing of the second stage is done iteratively, the different step are presented in the following sections.

3. Preliminary sizing of the first stage and of the boosters

3.1 Structure and thrust law of the first stage

For cost and simplicity reasons, the first stage of this advanced TSTO launch vehicle is an evolution of the P80 FW rocket motor used on Vega and takes advantage of the carbon-epoxy filament wound monolithic motor case technology. This allows using the existing production facilities with no or few modifications. The chosen diameter is for this reason set to 3 m. It is however now known whether some thrust laws may require a bigger diameter. This point will be considered in more detail in the follow-up of this study. According to data available on the first stage of Vega [5], the structural index of P80 FW, defined as the dry mass divided by the propellant mass, is about 8.4%. Due to the fact that only bigger motors are considered, namely between P120 and P160, a decrease of the structural index can be expected with the increase of the propellant mass. A structural index of 9% was however selected for each of the first stage motors to also include some margins. This value does not take into account the inter-stage, the separation system or the booster attachments which are considered separately. It was assumed that with this structural index, the casing will be able to withstand the same combustion chamber pressure as the one of P80 FW.

Consequently a maximum combustion chamber pressure of 90 bar was considered. The considered propellant is the same as for P80 FW, namely HTPB 1912 with a finocyl type grain shape.
For the different propellant loadings selected for the first stage motor: P120, P150 and P160, several thrust laws have been designed. A P100 variant which was first taken into account has been discarded early in the study due to excessively low performances. Each of the thrust histories has then been assessed with trajectory analyses; indeed for a given propellant mass, big variations in term of maximum acceleration, maximum dynamic pressure and payload performance can be observed. A trade-off has been done to avoid excessively high dynamic pressure which results from a high thrust level in the low layers of the atmosphere, but also to avoid too high gravitational losses resulting mainly from a low flight velocity during the firing of the first stage. The selected thrust laws for the different considered propellant loadings are plotted in Figure 1. The thrust law of P120 and P150 have been selected based on the results of the trajectory analyses done for un-boosted version. It was however found during the study that designing strap-on boosters capable of keeping the maximum dynamic pressure and the maximum acceleration at a reasonable level was difficult. For this reason the thrust law of the P160 solid rocket motor, which was not considered since the beginning of the study, has been selected based on the results of the trajectory of a boosted configuration. It can for example be noted that the first thrust peak of the P160 motor is just below 5000 kN contrary to the lighter P150 motor which reaches thrusts slightly higher than 5200 kN. As it will be shown in Section 6, this allows for a lower maximum dynamic pressure. The chosen nozzle expansion ratio is 16, that is to say the same as for P80 FW. The motor of the first stage of Vega has a vacuum specific impulse of 279.6 s, however P120, P150 and P160 were considered with a slightly lower value, namely 277 s, which gives an additional margin on the launcher performance. The specific impulse at sea-level is 242.1 s. The nozzle erosion was also taken into account and estimated based on data from the P80 FW motor.

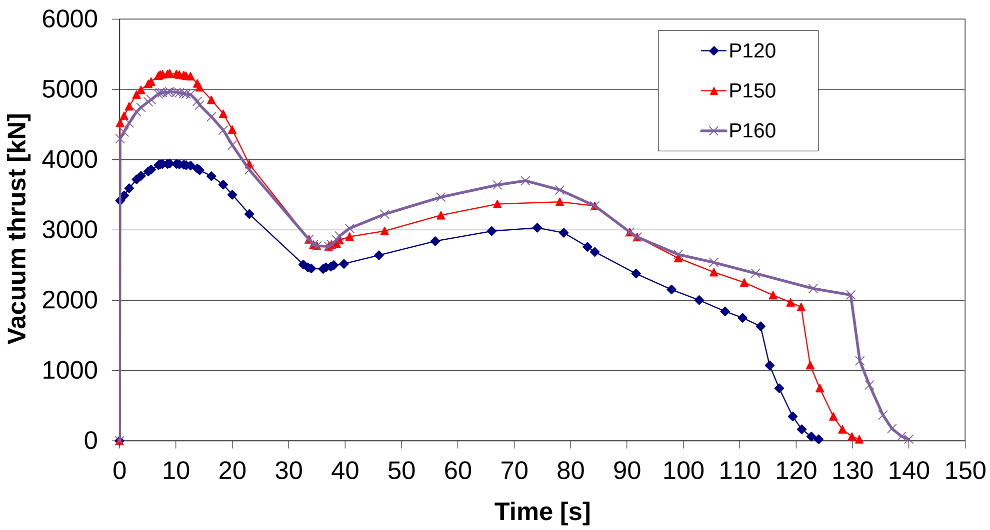

Figure 1: Assumed vacuum thrust law of P120, P150 and P160

3.2 Structure and thrust law of the booster

In order to increase the payload performance and allow the injection of 2700 kg to 3000 kg payloads into GTO, two strap-on boosters are used. No configuration using a larger number of smaller boosters has been considered. This would allow several performance steps which would however be quite small and whose need could be only barely justified. The first estimations show that about 30 tons of propellant would be required for each of the boosters. However they cannot be directly derived from an existing European solid rocket motor. Indeed the Z23 motor which is the closest European motor in terms of propellant mass does not have a thrust history adapted to a sea-level start. Only the carbon-epoxy filament wound monolithic motor case technology can be reused. The structural index of Z23 is lower than 8% [5] for a diameter of 2.2 meters and a maximum combustion chamber pressure of 95 bar, however this motor was not designed to be a booster. Notwithstanding, the GEM-60 booster produced by ATK to increase the payload performance of the Delta IV M+ launcher, which has a propellant mass of 29.7 tons, is characterised by a structural index of 13.2% [6] for a maximum combustion chamber pressure of about 56 bar. It is believed that using the technologies available in Europe the structural index could be reduced and a value of 11% including margins seems realistic. The thrust law of the Delta IV M+ booster is nevertheless not well adapted to the TSTO launch vehicles presented in this paper. A new thrust law allowing a reduction of the maximum dynamic pressure and of the maximum axial acceleration but with the same propellant mass as the GEM-60 has been designed and is plotted in Figure 2. Note that no detailed sizing of the booster has been done. For a matter of simplicity it was assumed that for the same propellant mass as the GEM-60, the same diameter and length could be used, that is to say a diameter of 1.52 m and a length of 13.16 m. The specific impulse of this P30 strap-on booster is 275 s in vacuum and 245 s at sea-level. In the follow-up of this study an optimisation of the booster will be done.

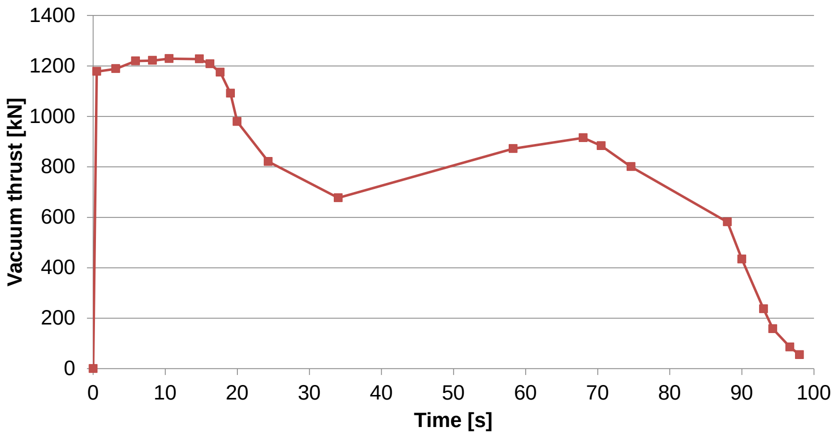

Figure 2: Assumed vacuum thrust law of a P30 strap-on booster

4. Preliminary results based on existing upper stage characteristics

Simultaneously to the definition and pre-sizing of the boosters and the first stages, a first estimation of the characteristics of the second stage has been done. In order to determine in which range the optimal propellant loading is situated a first analysis is done with the help of data available on existing cryogenic upper stages, independently of the tank type and the country where they were designed and built. Despite an important scattering, explained by the various stage architectures, it is possible to determine a trend of the dependence of the structural index on the ascent propellant mass for upper stages. The curve in Figure 3 has been determined by considering stages with an ascent propellant mass between 8 tons and 30 tons; this includes the H8 and the different variants of the H10 tank used on Ariane launcher family, but also among others the different variants of the Centaur upper stage, the different variants of the Delta upper stage, the second stage of the Japanese H-II launch vehicle or the ESC-A upper stage of Ariane 5. The structural index is defined here as the ratio of the dry mass to the ascent propellant mass, where the dry mass does not include the engine, the inter-stage, the fairing and the non-propulsive fluids but does take into account the avionic platform or the vehicle equipment bay.

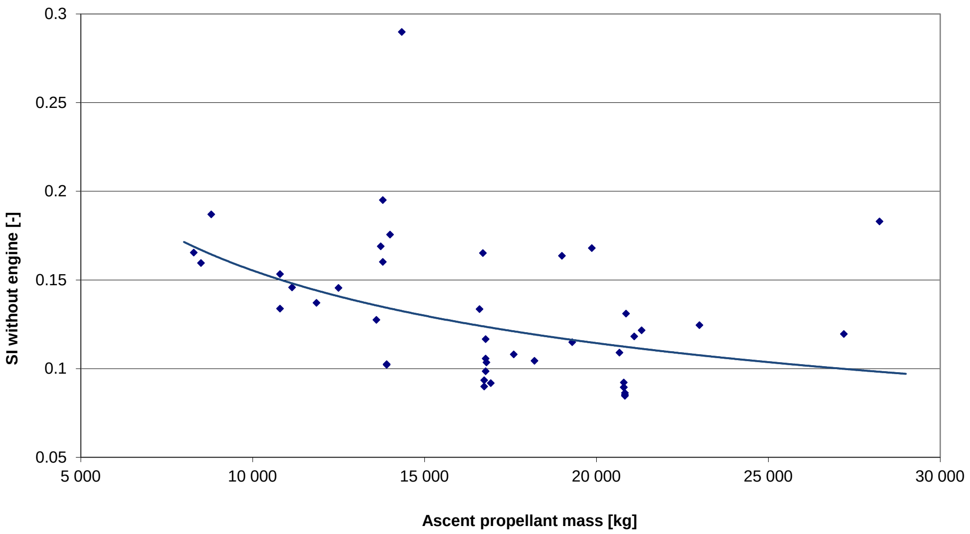

Figure 3: Estimated evolution of the structural index depending on the ascent propellant mass for cryogenic upper stages based on existing upper stages

Based on this ascent propellant dependant structural index and the results of trajectory analyses, it was found that the optimal propellant mass for a TSTO launch vehicle equipped with a P120 motor for the first stage was situated between 17000 kg and 23000 kg. Preliminary sizing of upper stages with 17 tons, 20 tons and 23 tons propellant have then been done for the common bulkhead and the separate tanks architecture with the method described in Section 5.

5. Preliminary sizing of the second stage

Due to the very different architectures of the existing upper stages considered in Section 4 to determine a structural index dependency, the optimum propellant loadings which were determined are only a first approximation of the real optimal. In order to assess more accurately the optimum propellant mass of the upper stage, upon trajectory analyses, an ascent propellant mass dependant structural index, corresponding to the architecture chosen for the advanced TSTO launch vehicle was determined with the help of preliminary stage sizing.
The preliminary analysis is carried out including, among others: the stage's structure, the propellant tanks and pressurization system and the de-orbit system.
The structural analysis is performed using an analytical tool capable of investigating stringer/frame stiffened shells or monocoque structures. Stringer and frame spacing as well as stiffener and skin dimensions are pre-optimized to find the minimum structural mass solution. Proper materials are selected.
As the TSTO launch vehicle cryogenic upper stage is based on the Vinci liquid rocket engine, elements of the Ariane 5 ESC-B upper stage are re-used as much as possible, in order to avoid excessive development costs.
Due to the specific design and required diameter for the TSTO upper stage, the propellant tanks cannot be adopted from the Ariane 5 ESC-B upper stage. Therefore, two different tank geometries are selected to be assessed: on the one hand a common bulkhead tank geometry and on the other hand two separate tanks. For both architectures, the tanks as well as the pressurization system are sized, based on the required propellant mass and the requirements of the Vinci engine. Afterwards, the most promising design, from a performance point of view, is selected.

5.1 Tanks, Feed Lines and Pressurization System

At first, the propellant tanks with its' feed and pressurization systems are investigated. Therefore the SART in-house tool PMP (Propellant Management Program) is applied. PMP is a system analysis tool which is used for preliminary design studies of launchers. PMP is able to calculate the propellant tank dimensions, the relating feed-lines and the pressurization system for both, separate tanks and common bulkhead tanks with the following input parameters: the specifications of the propellant and pressure gases, the propellant mass for the boost phase(s) including the propellant residuals and reserves, the propellant tank pressures, the pre-set tank radius and the duration of propelled phase(s). Additionally, the engine specifications (mixture ratio, propellant flow rate, required chill-down mass, start-up propellant) as well as the maximum axial accelerations have to be defined. Based on these inputs, PMP calculates the following parameters: the dimensions of the propellant tanks, the feed lines and the pressurization system and the total required propellant mass. Additionally, the masses of the feed and pressurization lines are computed and the mass of the pressure gas vessel(s) is scaled from the existing ESC-A vessel.

For this study, upper stages with five different propellant masses are calculated with PMP. For three of them, a separate tanks version and a common bulkhead version are compiled. But based on the analyses for this study, all separate tank geometries were rejected in an early stage, due to the fact that the upper stage becomes much heavier compared to the common bulkhead configuration which results in a decreased payload mass. This is due to the fact that an additional bulkhead and an inter-tank structure between both tanks are needed.
The disadvantage of the selected common bulkhead tank geometry is the insulation of the common bulkhead. The Vinci engine is fed with LOX (liquid oxygen) and LH2 (liquid hydrogen). These are cryogenic propellants which have to be stored at very low temperatures to keep them liquid but generate a high specific impulse. The nominal temperature for LOX is 90K and for LH2 20K. This results in a heat flow through the common bulkhead due to the temperature difference. This heat flow results in a temperature rise of the LH2 and a sub-cooling of the LOX. Both effects lead to the fact that the affected amount of propellant cannot be used to feed the engine. Consequently, an extremely effective insulation is required for the common bulkhead.

Nevertheless, the common bulkhead architecture is seen as the most promising for the upper stage of the TSTO launch vehicles; it is evaluated more precisely in the following. As the most meaningful stage configurations, the so called H24 and H26 upper stages are regarded since they promise a high payload mass. The H24 stage has 24000 kg of propellant for the boost phase and the H26 stage 2 tons more. In the following, the results calculated by PMP for these two stages are presented.

The PMP calculation of the H24 stage required the following inputs: as maximum axial acceleration during launch 5.2 g is assumed according to trajectory results and during upper stage operation, an axial acceleration profile is applied. The pre-set tank radius is 1.5 m with the nominally used propulsive propellant mass of 24000 kg. The used propellants are LOX and LH2 with the pressure gases GHe (gaseous helium) for the LOX tank and GH2 (gaseous hydrogen) for the LH2 tank. The implemented engine specifications are as follows: Vinci engine with a mixture ratio of 5.8, a propellant flow rate of 39.5 kg/s and a total chill-down mass of 144 kg prior to each start. The upper stage is ignited at 149 s after start and burns until 756 s, no second boost is regarded. As geometrical and thermal residuals for both tanks, 142 kg propellant is assumed. This value is based on experience of other studies. For propellant reserve, 413 kg LOX and LH2 are presumed. For the transient phase consumption of the Vinci engine, which represents the start and shut down phase, a total propellant mass of 80 kg is used.

As results of the PMP calculation for the H24 TSTO upper stage, the following can be summarized: the total length of the common bulkhead propellant tanks is 12 m; the total mass of propellant, required for the GTO mission is estimated to 3820 kg of LH2 and 21000 kg of LOX. The applied tank pressures are 2.8 bar for the LH2 tank and 3.33 bar for the LOX tank.

Based on the geometrical dimensions of the feed lines, their masses as well as the propellant masses for the lines priming, which represent the initial filling of the feed lines with propellant, are estimated. For the pressurization, GHe and GH2 are applied; the masses of the corresponding pressure lines and of the two He vessels are also calculated.

For the TSTO upper stage H26 with the common bulkhead configuration, the same inputs are used for the PMP calculation as for the H24 stage with the following differences: the nominally used propulsive propellant mass is 26000 kg. During the launch an axial acceleration of 5 g is assumed according to the flown trajectory; during the upper stage operation, an axial acceleration profile is applied. The upper stage is ignited at 156 s after start and burns until 814 s, no second boost is regarded. As geometrical and thermal residuals for both tanks, 149 kg propellant is assumed. The propellant reserve mass is estimated be 448 kg for LOX and LH2 together. For the transient phase consumption of the Vinci engine, which represents the start and shut down phase, a total propellant mass of 80 kg is used.

The results of PMP for the H26 stage are the following: the total length of the common bulkhead propellant tanks is 12.8 m. As total propellant for the GTO mission, 4130 kg of LH2 and 22730 kg of LOX are needed. The used tank pressures are 2.8 bar for the LH2 tank and 3.27 bar for the LOX tank. The propellant masses for the lines priming with propellant, the mass of the feed and pressure line as well as of the helium tanks are also calculated.

Figure 4 and Figure 5 depict the graphical outputs of PMP. The propellant tanks, the feed lines (red and yellow) as well as the pressurization lines (magenta and green) the pressure gas vessel (small green tank) can be seen.

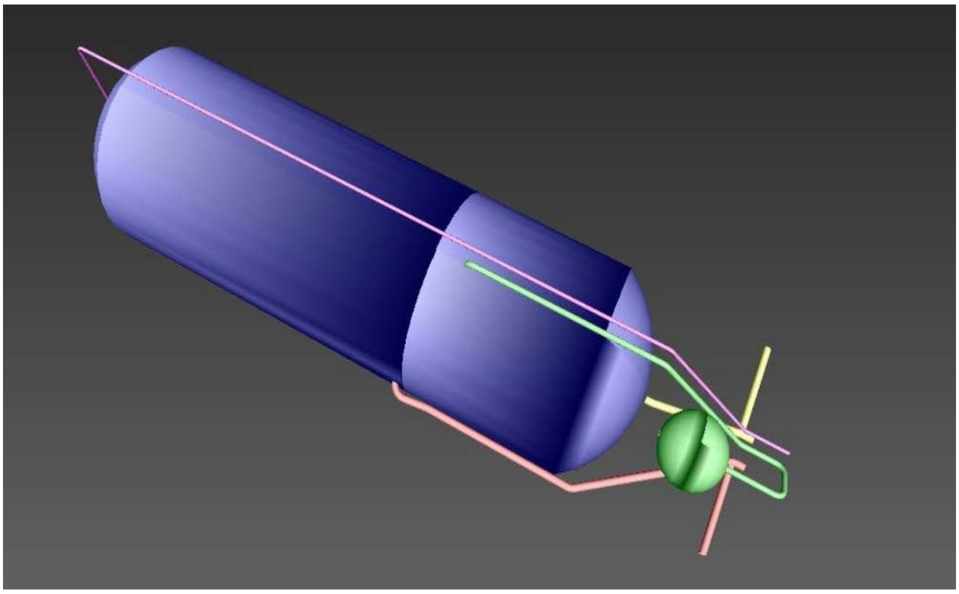

Figure 4: Homologous model of the tanks with feed and pressurization lines of the H26 upper stage

Based on these results, the structural analysis is performed in order to investigate the masses of the upper stage under the occurring loads. It is interesting to notice that a decrease of the maximum axial acceleration from 5 g to 4.5 g during the boost of the first stage has a negligible influence on the results previously listed.

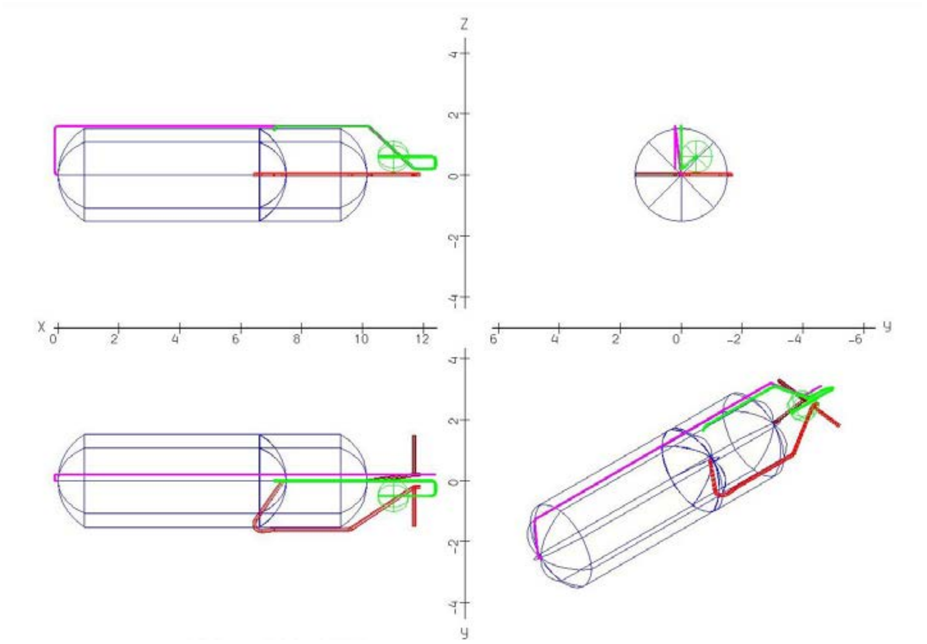

Figure 5: Homologous model of the tanks with feed and pressurization lines of the H26 upper stage: 2-D view

5.2 Structure

For the upper stage and the inter-stage structure a structural analysis has been performed using the newly developed SART tool LSAP (Launcher Structural Analysis Program). No detailed structural analysis has been done for the solid propellant first stage and the boosters. An early version of LSAP has been used in the frame of this project. The program will be further improved in the future and additional capabilities will be implemented.

LSAP treats a launcher as a 1-dimensional bending beam with rotational symmetry and uses analytical procedures for structural mass computation. Stringer/frame stiffened skins as well as monocoque structures with cylindrical or conical shapes can be modelled. For propellant tanks or pressure vessels, tank domes are also considered. Stiffener and skin dimensions are adapted iteratively for each structural segment in order to find the solution minimizing the mass. Several failure modes are investigated including global and panel instability failure, stiffener crippling and yield/ultimate stress failure. The masses of connection elements are considered via empirical formulations. Trajectories, aerodynamic coefficients, tank pressure and propellant masses as well as subsystem masses are provided by other SART-tools and are used as input data for LSAP. A pre-processor called LSAPPREP analyses and prepares the provided data and generates the input files for LSAP. The axial accelerations are provided by the trajectory analysis tool TOSCA. The lateral accelerations however will be computed by LSAP via lateral force and moment equilibrium.

14 different launcher configurations have been investigated with the help of LSAP, most of them are equipped with common bulkhead tanks for the upper stage. The modelling of the structure and the results of the calculations will be discussed in the following sub-sections.

5.2.1 Modelling and Assumptions

The structural elements of the upper stage investigated by LSAP are:

- Fairing
- Vehicle equipment bay (VEB)
- LH2 tank
- Inter-tank structure (for configurations with separate tanks)
- LOX tank
- Inter-stage structure (ISS)

All segments are modelled as stiffened shells with longitudinal stringers and circumferential frames. Tank domes are not stiffened. The fairing geometry is modelled by assembling a cylindrical and a conical segment. The material for

all structural segments including stiffeners is aluminium 2219T87. A safety factor of 1.25 has been applied since only unmanned payloads are planned to be launched.
The solid propellant first stage is not analysed by LSAP (see Section 3.1). However, its geometry and mass properties are modelled for LSAP to receive the correct launcher aerodynamics and to generate the force and moment equilibrium. The geometry model of the first stage consists of a large solid propellant vessel and a smaller aft skirt. The boosters are not investigated by LSAP. Their thrust surplus, drag and lift forces are introduced as external loads. The mounting position of the boosters is assumed to be at the aft skirt. Configurations with boosters use a fairing with a length of 9.0 m, while launchers without booster have an 8.2 m fairing. The lengths used for the fairing are first estimations which will need to be updated in a next step, based on the achieved payload performances. The VEB is assumed to have a length of 0.5 m. The inter stage structure has a length of 4.76 m. Inter tank structures, if existing, have a length of 2.24 m.

Four standard load cases are considered for configurations without boosters and five for variants with boosters:

- Launch pad, fully fuelled and pressurized, wind/gust loads
- Maximum dynamic pressure (max q)
- Maximum product of dynamic pressure and angle of attack (max (qα))
- Maximum axial acceleration prior to booster separation (max n_x)
- Maximum axial acceleration after booster separation (max n_{x2})

To account for dynamic loads the axial accelerations have been increased by 1.0 g for the max q and max (qα) cases, and by 1.25 g for the maximum acceleration cases. No particular load case has been generated for the upper stage alone since the accelerations are lower than for the first stage powered ascent, while the thrust frame of the second stage is not part of structural analysis. Load case data is automatically generated by LSAPPREP from the trajectory, aerodynamic and propellant tank characteristics.

5.2.2 Results

For the three configurations (P120 + H17, P120 + H20, P120 + H23) a comparison of common bulkhead and separate tank architectures has been performed. For the variants with common bulkhead the LOX tank is heavier than those of the variants with separate tanks. This is the result of the large volume required by the dome of the LH2 tank, which extends into the LOX tank. Consecutively, the cylindrical surface of the LOX tank becomes larger. This mass penalty for the LOX tanks could not be outbalanced by the advantage of utilising only one dome. The LH2 tanks on the other hand become heavier for the separate tanks configurations, since their lower dome requires a higher wall thickness than the common bulkhead (which is counted to the LH2 tank mass in the mass balance).

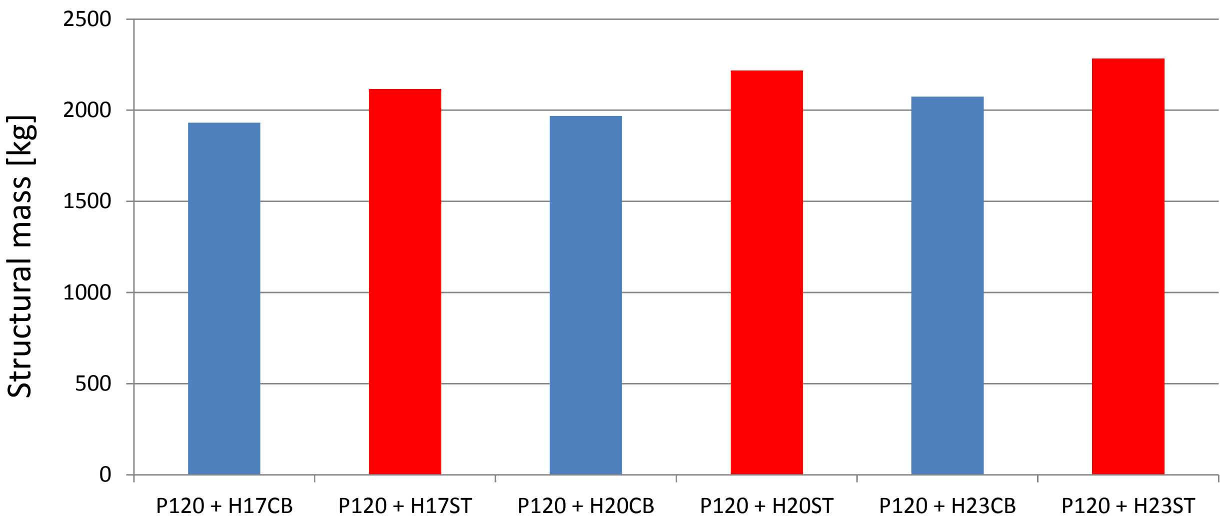

Figure 6: Estimated structural mass of upper stages with common bulkhead (CB, in blue) and separate tanks (ST, in red) architecture for the TSTO launch vehicle

Figure 6 presents the resulting structural masses. The masses given include the fairing, VEB, LH2 and LOX tanks, inter tank structure, thrust-frame and payload adapter. However, the structural masses of the thrust-frame and payload adapter have not been calculated by LSAP, instead they have been generated empirically.

It appears that the separate tank configurations are significantly heavier for all three propellant masses. Due to this drawback for the global performance, it was decided not to consider separate tank configurations in the next steps of the present study.
The structural analyses presented here provide a first estimation of the structural masses of the upper stage. During the analyses it has been found that for these stages, high dynamic pressures and high angles of attack at low altitudes are more critical than high axial accelerations. A reduction of structural mass may be achievable by keeping axial acceleration comparatively low until the dense atmosphere layers have been left. Higher accelerations and angles of attack should be flown only at high altitudes. Of course, this has to be balanced against other requirements.
The structural masses computed by LSAP have been used for a second optimisation of the ascent propellant mass of the second stage.

5.3 De-orbit strategy

Most launch vehicles in use nowadays finish their mission with the injection of their payload on the targeted orbit, they are then deactivated and continue on the same orbit during years, decades or even longer, increasing the risks of collision with other objects and of multiplication of space debris. To mitigate the influence of launch vehicles upper stages on the increase of space debris, a strategy has to be considered from the very beginning of the design. Indeed this requirement can have a strong impact on the payload performance. For a GTO mission, different solutions can be considered. The simplest one would be to reduce the perigee of the transfer orbit after the payload separation in order to allow a fast re-entry of the upper stage. The second solution would be to inject the upper stage in a graveyard orbit after the payload separation. Due to the fact, that many objects are already in low Earth orbits, an important increase of the perigee would be needed and would have a high impact on the payload performances. In case of the injection of the payload directly in geostationary orbit, the simplest solution would be to inject the stage after payload separation in an orbit slightly higher than the geostationary orbit. However only injections in GTO have been considered in this study until now. Thanks to lower ΔV requirement (about 23.3 m/s) the reduction of the perigee altitude from 250 km to 30 km has been selected.
Different options have been considered to provide the boost needed for the de-orbit manoeuvre. The first solution consists in reigniting the Vinci engine, to avoid additional systems and take advantage of the high specific impulse of this engine. The second solution is based on cold gas thrusters making use of the vaporised hydrogen in the LH2 tank. This kind of thruster is also considered for the reaction and attitude control system as for the ESC-A upper stage of Ariane 5. The third option is based on a solid rocket motor. Among all these solutions, the one based on the Vinci engine has the highest specific and consequently the lowest propellant mass requirement for the de-orbit boost, however prior to a restart, some propellant is needed for the chill-down of the engine. In addition after the ignition a transient phase consumes additional propellant. All in all more than 225 kg of propellant are needed before the steady state is reached. Consequently it is very likely that a sufficient impulse will be provided before reaching the steady state. The average specific impulse and required propellant mass is however strongly dependant on the temperature and pressure of the propellant at the end of the ballistic phase predating the de-orbit boost and the evolution of this characteristics during the boost itself. The same problem is encountered for the cold gas thrusters option. Based on the performances of the SCAR thrusters used on Ariane 5, a specific impulse situated between 120 s and 160 s is assumed. Finally the solid rocket motor option was assessed with a specific impulse of 270 s and a casing plus fixation mass representing one third of the needed propellant mass.

Table 2: Mass of the de-orbit system for the P160-H26 configuration, including propellant and structure

	Vinci	**Cold gas thruster** (Isp = 120 s)	**Cold gas thruster** (Isp = 160 s)	**Solid motor**
De-orbit system mass [kg]	225[a]	93	73	46

[a]Until the end of the transient phase

Table 2 sums up the mass of the de-orbit system for the different options for a P160-H26 launch vehicle. It appears that the lightest solution with about 46 kg would be to use a solid rocket motor. The cold gas thrusters option, even in the best case with a specific impulse of 160 s, would have a mass over 70 kg. Concerning the Vinci engine based option, the exact propellant mass is difficult to determine, as it does not depend only on the engine but also on the tanks and its insulation. Even if a mass below 225 kg might be possible, it is not estimated that this solution could reach the level of the solid rocket motor; consequently solid rocket motors have been considered in the frame of this study for the de-orbit manoeuvre.

5.4 Subsystems

The masses of the other sub-systems have been estimated based on heritage. For example the avionics is considered to be similar to the one of Vega, however an additional mass was considered for component duplication. The mass of the pressure control assembly, the actuators and the reaction and control system propellant mass are based on data available on Ariane 5 future upper stage ESC-B. Other masses like the mass of electric lines have been determined with the help of the DLR program STSM for mass estimation. A margin of 10% has been considered for all these subsystems.

5.5 Selected configurations

The results of the preliminary sizing of upper stages for a P120 + HYY configuration, which have also been extrapolated for propellant masses between 15 and 28 tons, are plotted in Figure 7. It can be seen that the structural index derived from existing stages (curve marked with triangles and already plotted in Figure 3) is close to the one determined for the common bulkhead architecture in the range 17 tons to 23 tons with an absolute difference lower than 0.5%. In the same range the structural index for the separate tanks architecture is higher by 1% to 1.5%. This result, which was expected, strongly disadvantages the separate tanks configurations in terms of performance and this drawback cannot be compensated by the good thermal insulation between the two tanks in the case of a GTO mission for which only one boost is foreseen. For example for such a mission, the difference in terms of payload performance for a P120-H21 is over 220 kg. Additional analyses will be done to assess the effects of propellant evaporation for missions including several boosts like missions including a circularisation and to check if the common bulkhead architecture can be at least as good as the separate tanks configuration with an appropriate but reasonable inter-tank insulation.

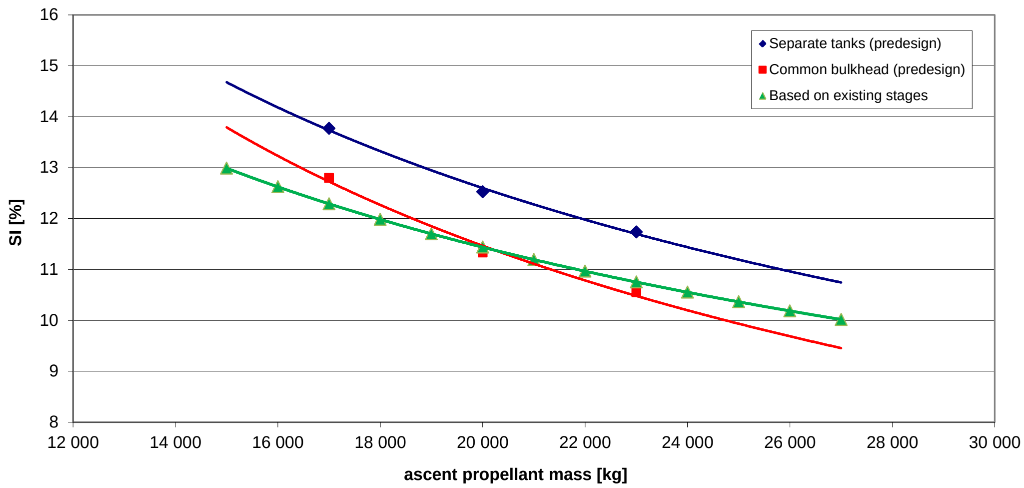

Figure 7: Estimated evolution of the structural index for cryogenic upper stages depending on the ascent propellant mass

With the help of the ascent propellant based structural index dependency determined for stages with common bulkhead architecture, the optimal propellant loading has been determined for the six following configurations all using common bulkhead tanks:

- P120 + HYY
- P120 + HYY + 2 P30
- P150 + HYY
- P150 + HYY + 2 P30
- P160 + HYY
- P160 + HYY + 2 P30

Normal and boosted versions based on the same first stage do not have the same optimal upper stage propellant mass. Indeed for versions with boosters, this optimal mass is about 3 to 4 tons higher than for the launch vehicle without boosters, see Figure 8. To avoid the design of two different upper stages for a given first stage, a trade-off is done and an intermediate value is selected, the goal is that the version with boosters and the one without use the same structure and components. Only the fairing differs from one version to the other. For the TSTO launch vehicle based on P120, a propellant mass of 24 tons has been selected for the upper stage. For the launchers based on P150 and P160, the upper stage has been designed for 26 tons of propellant. A 3D view of the different configurations can be seen in Figure 9. The lengths of the different configurations are also given and are situated between 36 and 41.5 m. A potential problem for the controllability of the launch vehicle has not been investigated yet. However the increase of the launch vehicle diameter, required by the thrust law of the first stage, which was not taken into account until now, might reduce control challenges. This issue is to be addressed for future iterations.

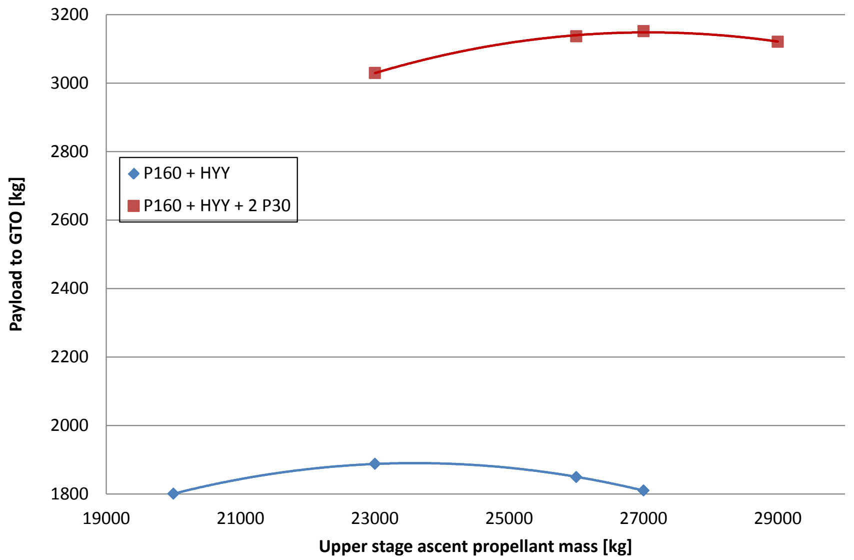

Figure 8: Evolution of the payload performance to GTO as a function of the upper stage ascent propellant mass for configurations based on P160

For each launcher the boosted configuration is the one which has the highest structural requirements, consequently all the components of the rocket have been sized to withstand the loads on the corresponding flown trajectory. The corresponding structural indexes are summed up in Table 3. Compared to what was previously expected from the ascent propellant based structural index dependency, these structural indexes are higher. This can however be explained by higher loads and an improvement in the LSAP program, calculating the structural masses, which took place during the study. Thanks to a thrust law optimized for the boosted version which is the most demanding in terms of structural loads, it is possible to adapt on the top of the P160 booster a lighter upper stage than with the P150 first stage, even if both have the same ascent propellant mass and the same tank pressure. Indeed, due to the thrust law of P150 the maximum axial accelerations, the maximum dynamic pressure and the maximum qα are higher for the variant based on P150 than for the one using P160 as a first stage.

Table 3: Structural index of the upper stage of the selected configurations

	P120 + H24 (+2 P30)	**P150 + H26 (+2 P30)**	**P160 + H26 (+ 2P30)**
Structural index [%]	11.6	11.2	11.0

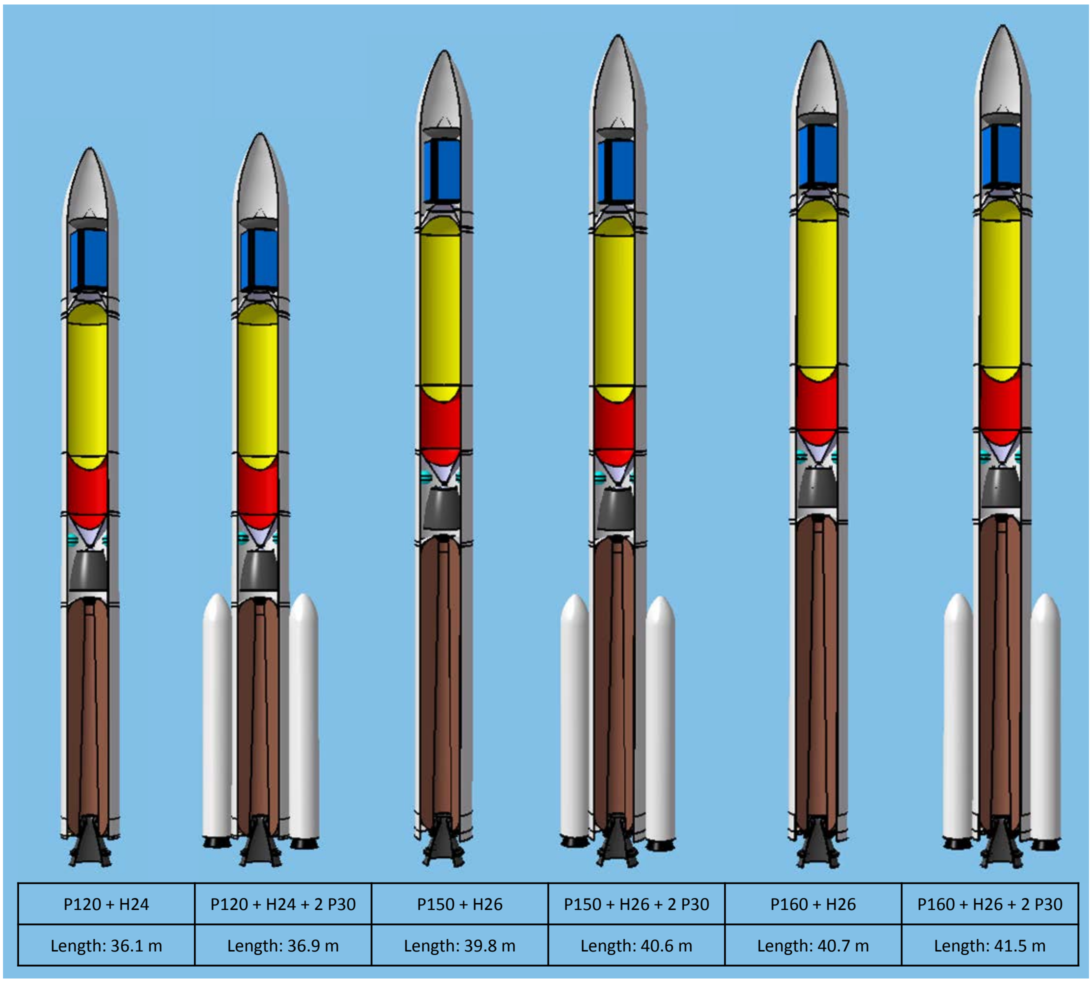

Figure 9: 3D view and total length of the selected configurations

6. Performances

Based on the masses presented in Section 5, the payload performances of the different advanced TSTO launch vehicles were assessed. An own aerodynamic model is considered for each of them. The main characteristics of the ascent trajectory for each of the considered configurations are summed up in Table 4. As expected the best performances are reached by the launch vehicles based on P160. With a maximum payload capacity of almost 1.6 tons in GTO the version without boosters has the capability to achieve Galileo satellite replacement single launch missions. Indeed the payload performance in MTO is slightly higher than the one to GTO. The maximum acceleration and maximum dynamic pressure are in the range of what is nowadays achieved by most launchers and therefore considered as reasonable. If two P30 boosters are added this advanced TSTO launch vehicle would be able to inject up to 2800 kg in GTO. With 4.5 g, the maximum acceleration is in the same range as what is achieved for a reference mission of Vega. Only the maximum dynamic pressure is a bit high, but the structure and the fairing have been sized to withstand the corresponding loads. The separation of the boosters is taking place at a dynamic pressure of about 1.6 kPa, that is to say in a range where this is considered as feasible. The corresponding ascent trajectory is plotted in Figure 10. The configurations based on P150 also have interesting performances, though there are slightly below those of launch vehicles based on P160. However the maximum acceleration and maximum dynamic pressure are considered as high. A new adaptation of the thrust law of the P150 solid rocket motor would be required, especially if a boosted version is foreseen. A reduction of the maximum dynamic pressure and of the maximum acceleration would have, however, probably a negative impact on the performances due to an increase of the gravitational losses even if the structure of the second stage could be slightly lighter. Finally the configurations based on P120 feature performances deemed too low. Indeed the version without booster would not be able to achieve Galileo satellite replacement single launch missions.

Table 4: Main characteristics of the ascent trajectory to GTO for the selected configurations

	P120 + H24		**P150 + H26**		**P160 + H26**	
Booster	**-**	**2 P30**	**-**	**2 P30**	**-**	**2 P30**
Acc_max [g0]	4.1	5.2	4.1	4.9	4.2	4.5
q_max [kPa]	54.2	69.1	55.8	71.7	46.1	60.3
q @ booster separation [kPa]	-	1.3	-	1.3	-	1.6
Payload to GTO [kg]	891	2031	1376	2588	1598	2792
GLO mass [kg]	162010	229598	197440	265120	208450	276110

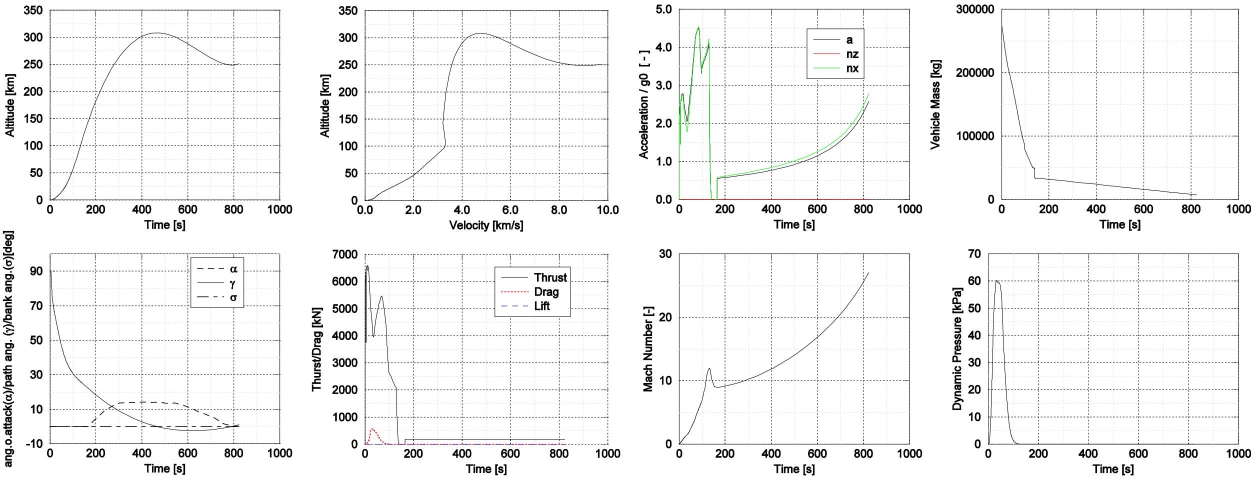

Figure 10: Ascent trajectory of the P160 + H26 + 2 P30 configuration

7. Conclusion

An advanced two-stage to orbit launch vehicle based on technologies and components already available in Europe and exploiting synergies with other on-going European programs, like a P160 + H26 configuration could inject up to 1600 kg in a geostationary orbit and a little bit more in a MTO. Adding boosters, it was calculated that the payload performance to GTO could be increased to inject up to 2800 kg. It appears that at least a P150 first stage will be however required to reach, without strap-on boosters, the targeted performance; namely a Galileo satellite replacement single launch mission. The performances of variants based on P120 are deemed too low. The present configurations are promising, but the controllability might be a problem due to the length of the launchers.

In the coming months, this study will be continued by the DLR's group for Space Launcher Systems Analysis (SART). A modification of the diameter of the first stage and second stage will be considered to better match with the probable diameter required by the first stage and to reduce the overall length of the launch vehicle. The influence on the structural index will be assessed. The behaviour of the launcher for different missions will also be studied and particular attention will be paid to the effect of the ballistic phase on the upper stage propellant between boosts. A study of the controllability of the selected launch vehicle is also planned.

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