Design and Analysis of an Airborne, solid Propelled, Nanosatellite Launch Vehicle using Multidisciplinary Design Optimization

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Abstract
The work focusses on the use of multidisciplinary optimization to design a cost optimized airborne nanosatellite launch vehicle capable of bringing a 10 kg payload into low earth orbit (LEO). Piggyback or shared launch options currently available for nanosatellites are relatively low cost (~45,000 €/kg) but have as serious disadvantage a limited mission flexibility due to a limited range in attainable orbits and the launch schedule being connected to that of its fellow passengers. An alternative option, providing increased mission flexibility, is through the use of a dedicated launch vehicle, be it at a higher launch cost. An interesting option to limit the increase in launch cost is by air-launch to orbit using an already existing aircraft as carrier vehicle, i.e. first stage. This is considered beneficial especially for small launch vehicles as many potential carrier vehicles are available and because of the relatively high drag loss that is associated with ground-launch to orbit for small launch vehicles.
The work presented here addresses the use of multidisciplinary optimization (MDO) methods to the design of solid rocket propelled launch vehicles, thereby taking into account both air- and ground-launch as well as the addition of lifting devices (use of wings). The method combines both vehicle and trajectory design in a sequential approach. Analysis modules included address issues concerning vehicle geometry, aerodynamics, solid rocket propulsion and vehicle mass, size and cost.
The tools developed have been used to design a low-cost, solid propelled vehicle that is launched from an F-16 aircraft. Main design variables are release altitude, -velocity and -flight path angle as well as number of rocket stages, stage thrust and stage burn time. The results show an optimized, three stage launch vehicle that fits within the contours of the F-16's 370 gallon external fuel tank and with a gross take-off mass that is up to 70% lower than that of a comparable 10-kg to LEO ground launched vehicle. The vehicle's launch costs are estimated at 1.9 million euro per launch based on a total of 120 launches over a period of 20 years. This is a 30% reduction as compared to the cost of a comparable 10-kg to orbit optimized ground launched vehicle.

1. Introduction

The emerging market for nano- and microsatellites with a total mass in range 1-50 kg is considered a prime market for airborne launch vehicles [1]. Currently, microsatellites are often sharing a ride with larger satellites, because opportunities for a dedicated launch are lacking. However, this piggybacking or shared launch brings for the small satellite as a disadvantage that the main paying customer dictates the final destination and launch date. This makes it difficult to fully exploit the full potential of the small satellite's mission. Another concern for nano- and microsatellites is to find a suitable piggyback ride or a spot on a shared launch [2].
Recent projections indicate continued growth of the nano- and microsatellite market, with an estimated number of 121 to 188 nano- and microsatellites ready for launch in 2020 [3]. Experts from industry and governmental organizations have identified a need for a dedicated launch vehicle for nanosatellites [4] and indicate that to make a dedicated nano-or microsatellite launch vehicle competitive with ride share, cost per flight should be of the order of 1-2 $M per launch. For a 10 kg nanosatellite, this comes down to a specific launch cost of 100,000-200,000 $/kg. This should be compared with a specific launch cost in range 50,000-55,000 $/kg (45,000 €/kg) for a ride share [5]. Using an airborne launch platform, i.e. air launch instead of ground launch, is considered a viable option by various researchers to reach the desired cost level [6-16].

Also in the Netherlands air launch has attained some interest, partially based on NASA's Horizontal Launch Study (HLS) wherein a modified fighter jet is considered as an option for carrying a small multistage solid rocket to launch a small satellites (payload mass less than 250 kg) [10]. Hence in 2011, the National Aerospace Laboratory (NLR) conducted a study entitled "Affordable Launch Opportunities for Small Satellites (ALOSS)" focusing on the use of an airborne platform for a dedicated launch (air launch) for nano- and microsatellites in the mass range 1-20 kg [6]. A conceptual design of a near-term (1-3 years) three stage launcher concept capable of launching a 10 kg satellite in low Earth Orbit (LEO) was performed using the F-16 fighter jet as an airborne launch platform. An important limitation in the study by NLR is no comparison was made with a comparable ground launch vehicle, which makes it difficult to quantify the relative improvement of air launch versus ground launch in a fair way.
In this study the cost and performance advantages of air launch rockets compared to ground launch rockets are addressed for a 10 kg satellite to LEO. Like for the NLR-study, this study will be limited to the use of solid rockets only. The first reason is that solid rockets are cheaper than liquid rockets even though liquids outperform solids in terms of specific impulse [18]. Next, the higher acceleration levels of solid rockets in comparison to liquid rockets increase the significance of drag and, therefore, solid rockets would have more benefit from air launch than liquid rockets [10], [11]. Moreover, because of the high mass density of the solid propellant, they allow for a high thrust-to-frontal area ratio, thereby allowing for short burn time and hence a reduced gravity loss. A final reason is that, solid rockets have shown to be able to withstand the sideways g-forces and the high aerodynamic pressure of a horizontal air launch with little increase in mass [6]. This is due to that the motor case of a solid rocket must be sized to withstand the internal pressure of combustion, which adds to the rocket's structural strength [6]. An important element of this study will be the comparison with the vehicle from the ALOSS study. But before discussing the methods and models used as well as the results, we will first discuss air- versus ground-launch in a more general sense to provide an overview on the issues involved.

2. Air launch versus ground launch

The most obvious benefit of air launch over ground launch is that the required ΔV to achieve orbit is reduced. In [9], it is stated that air launch provides a reduction in ΔV to orbit of ~300-950 m/s in case of subsonic launch conditions. It is also concluded that the higher the launch altitude the better [9]. The reasons for this reduction are multifold. First of all, an air launched rocket obtains the velocity at release from the airborne launch platform for free. Second, with increasing release altitude, the time the rocket has to fly through the denser layers of the atmosphere decreases, thereby decreasing the drag loss [7]. According to [6], air launch is more beneficial for small rockets than for larger ones as for the former drag loss is more significant. Thirdly, air launch limits the gravity loss because the time that an air launched vehicle needs for the ascent is shorter than for a ground launched vehicle [7]. In addition, the shorter flight time of an air launched rocket will lead to a reduction in steering losses [6]. A fifth reason is that for an air-launched rocket a more efficient nozzle design (higher specific impulse) can be utilized because of the lower ambient pressure at launch altitude [5]. Every rocket motor is designed for a certain altitude. At this design altitude the ambient pressure equals the exit pressure of the nozzle and ideal expansion will occur [19]. The first stage nozzle design of a ground launched vehicle is typically compromised due to the large range of altitudes it needs to cover during ascent. The nozzle design of the first stage for an air launched vehicle is less compromised, as it operates over a smaller range of pressures. In [7] it is claimed that a more efficient nozzle expansion can bring down the required amount of ΔV to orbit with up to 105 m/s.
Next to the advantage in terms of ΔV to orbit, air launch also holds other benefits. According to [6], [20] air launch also reduces the aerodynamic loads on the launch vehicle, which allows for a simplification of the structural design and hence a lower structural mass for the launch vehicle. Another advantage of air launch is a reduction in acoustic loads compared with ground launch [14], which again may lead to a reduction in structural mass. Air launch furthermore allows for aircraft-like operations, including the ability to conduct launch operations over open ocean areas, far from populated areas, which can significantly reduce range safety concerns. Due to the mobility of the carrier aircraft also a wide range of orbital inclinations can be obtained. This operational benefit is actually also a performance benefit because the flexibility of launch latitude and azimuth removes the need of expensive (high ΔV) dogleg maneuvers. A further operational advantage of air launch is that it does not require complex launch facilities and that air launch is not restricted by the extremely demanding weather conditions that are imposed to a ground launch because it can fly to another location or can be launched above the weather.
Another advantage for air launch is in terms of cost. Since the rocket can be made smaller, it is expected this leads to a lower launch cost, be it that some of the cost gain may be off-set by the cost of the carrier vehicle and the additional operations cost.
A major problem of air launch is certification. The ignition of a rocket motor in the proximity of a manned aircraft is the main reason for flight clearance authorities to reluctantly certify air launched vehicles. A problem with the certification was one of the primary reasons for the cancellation of AirLaunch LLC's QuickReach launch vehicle [13]. The president of SpaceWorks Inc. (involved in the GOLauncher concept) expressed that certification issues are

a major threat for the program [15], [16]. A further downside of air launch is that the gross takeoff weight (GTOW) and the geometry of an air launched vehicle are restricted by the limitations of the carrier aircraft. Therefore, the growth potential for air launched vehicles is also limited. A third disadvantage is the risky separation of the launch vehicle from the carrier aircraft.

Another problem is the additional steering and loads that depend on the separation conditions. According to [9], the optimal release flight path angle is around 30° but has for subsonic release velocities a range of almost ±15°. This requires for the launch vehicle to perform a pull-up maneuver to gain altitude quickly. This increases the steering needed and may also lead to higher structural loads thereby offsetting the gain in structural mass from launching at a high altitude. Also according to [9] there is no ΔV benefit to be expected of having wings in case the release flight angle is close to optimal.

3. Method

For the optimum design of both air- and ground-launch solid propelled rockets, with or without lifting devices, it is considered to design and optimize the launch vehicle and its ascent flight simultaneously. For this Multidisciplinary Design Optimization (MDO) is deemed the most suitable approach. MDO allows designers to incorporate all relevant disciplines of an engineering problem simultaneously. A top level overview of the tool developed for the present study, the disciplines involved and the dependencies of the various disciplines is shown in Figure 1. The method combines both vehicle and trajectory design. Available analysis modules (Multi-Disciplinary Analysis; MDA) address issues concerning vehicle geometry, aerodynamics, rocket propulsion performance and vehicle mass, size and cost as well as ascent flight (trajectory) including flight loads estimation depending on the flight environment (atmosphere and gravity). As design variables are considered amongst release altitude, release velocity, initial flight path angle after release and for each stage: motor chamber pressure, nozzle exit pressure, motor case diameter, nozzle exit diameter and motor burn time. The search space for these variables is based on the values of actual Solid Rocket Motors (SRMs). Where necessary, the models are adapted to allow for accurate modelling of small (solid) rockets. As inputs are considered design constraints, payload mass, target orbit, etc.

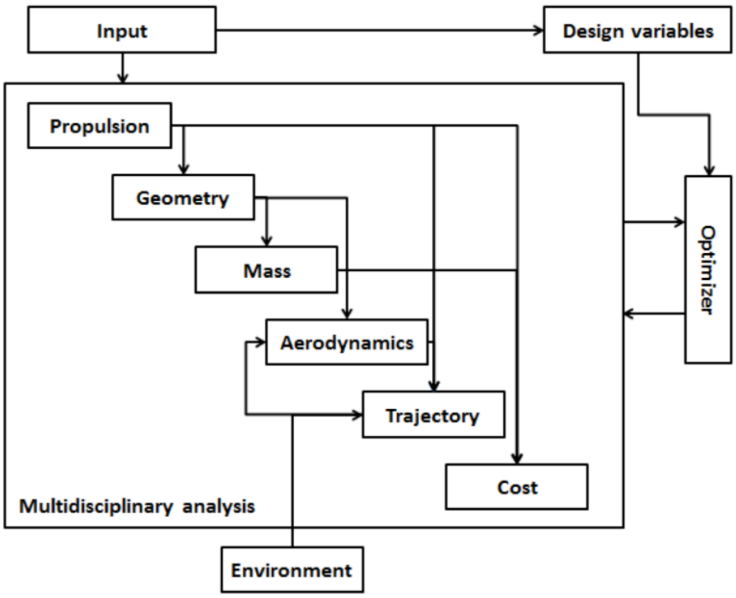

Figure 1: Top level overview MDO architecture

Starting point for the tool has been the work performed in [21], [22] wherein a rocket launch vehicle ascent simulation has been developed based on the TU Delft Astrodynamics Toolbox (TUDAT) and combined with an MDO tool. TUDAT is a C++ library that originally provided functionality to perform astrodynamics simulations [23]. TUDAT is developed within the Space Engineering department at TU Delft, faculty of Aerospace Engineering. It is set up with particular focus on code modularity and robustness. The library contains various environmental models, reference frames and numerical integrators. Based on the work conducted in [21] and [22], routines have been added for modelling rocket ascent, rocket aerodynamics (Missile Datcom) including lifting surfaces, liquid rocket propulsion, rocket mass and rocket geometry and to allow for multi-objective optimization using the ESA developed Parallel Global Multi-objective Optimizer (PaGMO) [24]. This is a C++ platform that contains a number of optimization algorithms.

New in the present study are the inclusion of solid rockets in the propulsion model as well as in the geometry and mass model. In addition, a module has been added to allow handling multiple staging. Also an extensive cost model capable of estimating development, production and operations cost for small rocket stages has been added. This model also allows for taking into account production series size as well as a decrease in cost because of the learning effect. This allows the vehicle not only to be optimized for maximum payload mass, minimum Gross Take-Off Weight (GTOW), but also minimum life cycle cost. The various models will be described in the next section in more detail with focus on the additions made in this study.

3.1 Environmental models

A central gravity field model and the 1976 U.S. Standard Atmosphere in tabulated form have been used for this study [21].

3.2 Propulsion

Propulsion performances and more specifically specific impulse and thrust are modelled using ideal rocket theory taken from [19]. Thermodynamic characteristics of the propellants (input to the tool) are determined using NASA's Glenn Equilibrium Program (CEA) [25], [26] under the assumption of chemical equilibrium in the combustion chamber. Using ideal rocket theory, specific impulse and thrust are known to be overestimated. Hence, in this study, we have used data of 14 Solid Rocket Motors, taken from [27], to determine a correction that allows for reducing the difference between actual and predicted performances. In Figure 2, results are shown for the uncorrected (original) model and the corrected (adjusted) model.

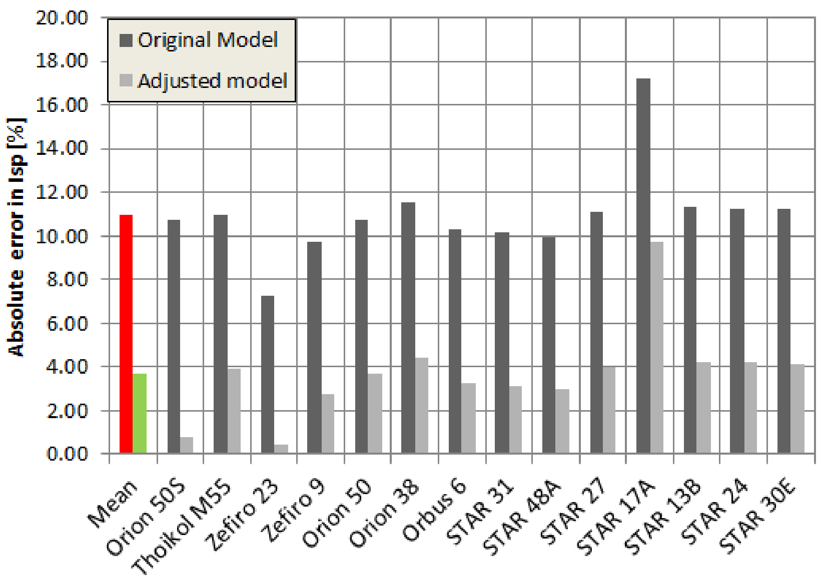

Figure 2: Absolute error [%] in calculated versus actual vacuum specific impulse (Isp)

Also shown in the figure is the absolute mean error E, which is defined mathematically as:

$$E = \frac{100\%}{N} \sum_{i=1}^{N} \frac{\left| y_i - \bar{y} \right|}{y_i} \tag{1}$$

Here N is the number of data points; y_i is the data point value and $\bar{y}$ is the average value of the parameter y. Results showed that the remaining mean error (μ), from Eq. (2), in thrust, specific impulse and propellant mass after correction for the 14 SRMs investigated could be limited to less than 0.5% for thrust and less than 2.5% for specific impulse. Sample standard deviation (σ), as defined in Eq. (3) was found to be about 10% for thrust and less than 2.5% for specific impulse.

$$\mu = \frac{100\%}{N} \sum_{i=1}^{N} \frac{y_i - \bar{y}}{y_i} \tag{2}$$

$$\sigma = 100\% \sqrt{\frac{\sum_{i=1}^{N}\left(\mu - \left(\frac{y_i - \bar{y}}{y_i}\right)\right)^2}{N-1}} \tag{3}$$

3.3 Geometry model

For this study only serial staged multi-stage launch vehicle stages with each stage having a single motor only (no bundles of SRMs) and with the stages burning sequentially. The reason is that almost all solid propelled launch vehicles have this configuration [27], [28], [29].

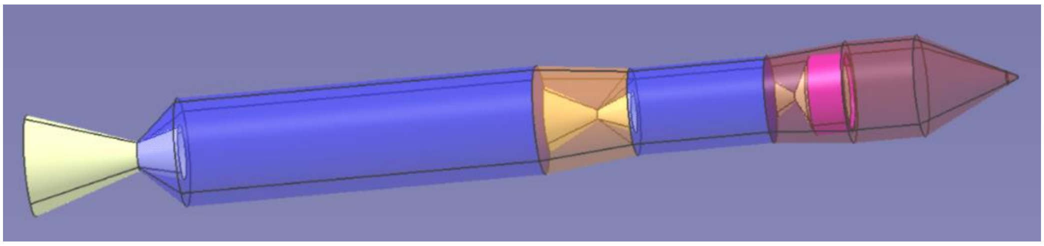

Figure 3: Impression of the launch vehicle model.

Separate geometry models have been generated for stage, fairing and the wing and tail surfaces. For instance, the rocket stage is considered to consist of the length of the motor case plus the length of the con-di nozzle. Length relations have been determined using outputs from ideal rocket theory on nozzle geometry. To simplify the model, the intricate relation between regression rate, burn time, grain shape and stage diameter is neglected. For more information, see the section on sensitivity analysis. Typical results for length are shown in Figure 4. Here also a correction factor has been introduced to minimize differences between predicted an actual results. The figure clearly shows that the absolute mean error reduces to less than 10% after correction.

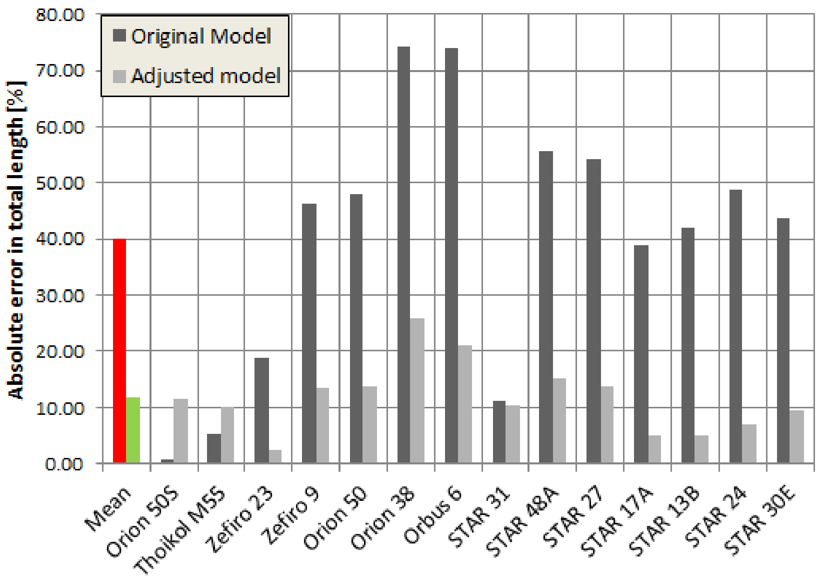

Figure 4: Estimated stage length versus actual stage length for existing SRMs

The payload fairing is modelled as an ogive shape that encompasses a user-described volume. Its base is taken to be of an identical diameter as the upper stage carrying the fairing. The inter-stage between two stages is modelled as a frustum with the length of the nozzle of the stage on top plus a fixed distance of 0.1 m.

3.4 Mass model

The mass model of a rocket without wings and tail surfaces is determined using the breakdown as shown in Figure 5. The approach followed is essentially identical to the approach presented in [30], but differs in that the mass of the thrust vector control (TVC) system is included in the nozzle mass. Mass estimation relationships used are summarized in Table 1.

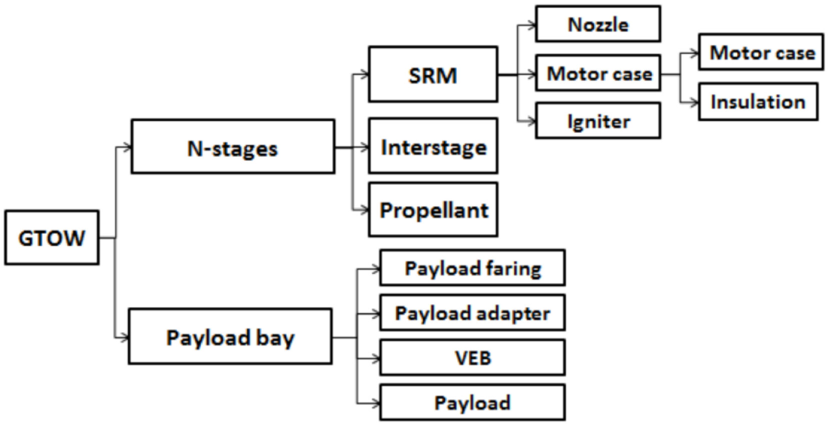

Figure 5: Mass breakdown structure of launch vehicle (no aerodynamic surfaces included)

Table 1: Mass estimation relationships for small solid rocket launch vehicles

Mass item	Relationship	R-squared/RSE	Remark	Ref.
SRM nozzle	$M = 0.0006*F^2-0.3214*F+263.8$	0.9991/4.5%	F > 200 kN	This study
	$M = -0.0018*F^2+1.004*F-1.9$	0.9425/14.8%	F < 200 kN, with TVC	This study
	$M = 0.1605 *F^{1.2466}$	0.9456/29.2%	F < 200 kN, no TVC	This study
Igniter	$M = 20.62 * (V_{cavity})^{0.7368}$	-/13.2%		[30]
Inter-stage	$M = K * 7.717 (S_{int}) ((D_{int})^{3.3208})^{0.4856}$		K = 1.0 for aluminum case and 0.7 for composite case.	[29]; Lower stages only
Payload adapter	$M = 0.004775 \cdot(M_{payload})^{1.0132}$	-/38.2%		[30]
Fairing	$M = 12.2\ S_{fairing}$	NA		[30]
VEB	$M = 0.3672*(M_{dry})^{0.6798}$	0.9527/25.1%		This study
All mass values in kg		S_{int} is inter-stage surface area in m^2		
F = thrust in kN		D_{int} is inter-stage diameter in m		
V_{cavity} = Motor free volume in m^3		$S_{fairing}$ is fairing surface area in m^2		
$M_{payload}$ = payload mass in kg		M_{dry} is launcher dry mass in kg		

Table 1 does not provide a mass relationship for the motor case as this is determined based on taking into account the thickness of the case material that results from the Maximum Expected Operating Pressure (MEOP, a design variable), case diameter (D_{case}), material strength (σ; material is user input) and some suitable safety factor (also user input), using (cylindrical geometry):

$$t = \frac{MEOP \cdot D_{case}}{2\sigma} f_{safety} \tag{4}$$

In the above table also the R-squared value and the relative standard error of estimate are given. An R-squared value close to 1 is a good fit, whereas a value close to zero indicates a bad fit or no dependency of the dependent parameter on the independent parameter. Relative standard error is calculated using:

$$RSE = \sum_{j=1}^{n} \left(\frac{y_j}{f(x_j)} - 1 \right)^2 \times 100\% \tag{5}$$

It is expressed as a percent of the estimate. For example, if the estimate of VEB mass is 100 kg and the RSE is 25%, than there is a probability of 68% that the real VEB mass is in range 75-125 kg.
In Figure 6 absolute error for motor inert mass (SRM mass excluding propellant mass) is given for a number of existing stages are given as well as the mean absolute error. Some correction has also been applied to allow for reducing the mean error. Results indicate an average error of -2.76% (mass is underestimated on average with 2.76%) with a standard deviation of 13.8%.

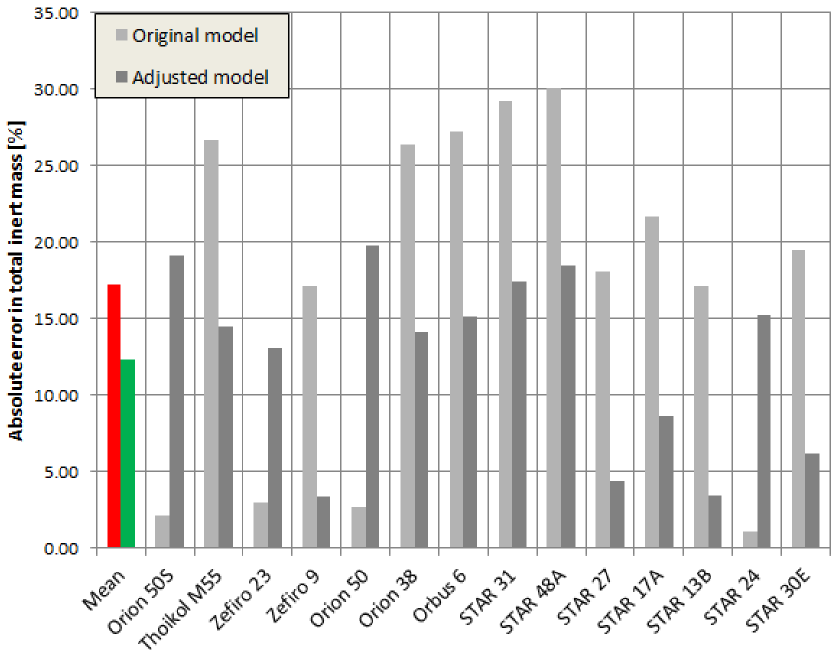

Figure 6: Absolute error [%] in calculated total inert mass compared with actual inert mass

3.5 Aerodynamics

Missile Datcom 1999 is used for the generation of aerodynamic coefficients for the launch vehicles [31]. The reason is that there is relevant experience with the use of Missile Datcom at TU Delft [21], [22], but its use is also widespread elsewhere [9], [30], [31]. This decision was based on the availability of the tool and because in [21] a routine to implement Missile Datcom in the TUDAT framework was developed. According to [31], [33] Missile Datcom has errors of ±20% for the axial and normal force coefficients. In [30] errors of 20% in the Missile Datcom predicted aerodynamic coefficients for Ariane V and Vega are reported.

3.6 Trajectory

In a three dimensional space the motion of a rigid body can be described by a combination of translational and rotational equations. For trajectory simulation the main interest is in the motion of the vehicle's center of mass, which essentially reduces the problem to 3 Degrees of Freedom (DoF). For our study the vehicle is assumed to be a point mass, thereby assuming that the rotational motion does not significantly affect the results of the study. This holds in it the presumption that the vehicle control system can generate the required moments to change the attitude of the vehicle. A derivation of the equations of motion for a mass-varying body can amongst others be found in [22].
To steer the vehicle, a parametric control law is used that is determined by defining discretization points all along the trajectory. A number of pitch angles is selected for each stage and the interval is divided over the burn time of the stage. These points are optimized in order to satisfy the optimality conditions of the problem. The control law is defined by linear interpolation between the discretization points. To reduce the computation time, piece-wise linear functions are used to interpolate the control law [36]. Also the coasting time between the stages is a variable that can be varied to investigate its effect on launch performance. Important constraints that have been included are maximum dynamic pressure, maximum acceleration, maximum bending load and heat flux due to aerodynamic heating.

3.7 Cost

Initially, we set out to implement the TransCost cost model [18] to estimate the life cycle cost of the launch vehicle. This model is considered standard for medium heavy and heavy launch vehicles. However, during the process, it was found that TransCost has the tendency to significantly overestimate the cost of small solid rocket stages. This was confirmed by [30] and [37], with the latter already providing an adapted TransCost model for small rocket stages. This adapted cost model was taken as the baseline cost model to be implemented. However, further modifications have been made to further improve the production cost estimation of small solid rocket stages (propellant mass < 10,000 kg). Using data provided by ESA [38] the cost estimation relationship for small SRM propelled rocket stages has been adapted. The adapted model, like TransCost itself, has a variety of correction factors taking into account the degree of engineering required. Most correction factors are considered of similar value for either a ground launch or am air launch. However, correction factors relating to the complexity and the integration of the launch vehicle were given a slightly higher value for air launched vehicles. Also the operations costs have been adapted to allow for taking into account the different operations for an air launch vehicle and the cost of the airborne launch platform. For this the approach taken in [39] was followed. The cost of developing the hardware and certifying the launch vehicle to be launched from under an F16 is estimated at 1-5 M€. In addition the cost of using the F16 for a typical mission are estimated at 10-50keuro, which is considered negligible when compared to the total operations life cycle cost. Full details of the cost estimation model can be found in [40]. Here only an overview is provided of the estimation relationships used for solid rocket stage production cost, see Table 2.

Table 2: Estimation relationships for production cost of SRM propelled rocket stages

SRM	CER	Remark	Ref.
Small SRM	$PC = 0.2422\,(M_p)^{0.2962}$	RSE = 21.7%	This study
Intermediate SRM	$PC = 3.12 + 0.068\,(M_p/1000)$	NA	[37]
Large SRM	$2.3\left(\dfrac{M_p}{11-5\left(\dfrac{M_p-40000}{200000}\right)}\right)^{0.399} - 38.25$	NA	[37]

PC is production cost in [€M], Fiscal Year 2009
M_p is propellant mass in kg
NA is Not Available

Typical estimated life cycle cost estimates versus the observed cost are provided in Figure 7. Results confirm the overestimation that results from the use of TransCost. In contrast, Martino's model and the model developed in this study give much better comparison with the observed value. Based on the numbers provided in the above figure, it follows an average over-prediction of the cost per flight of 19% with a standard deviation of about 14%.

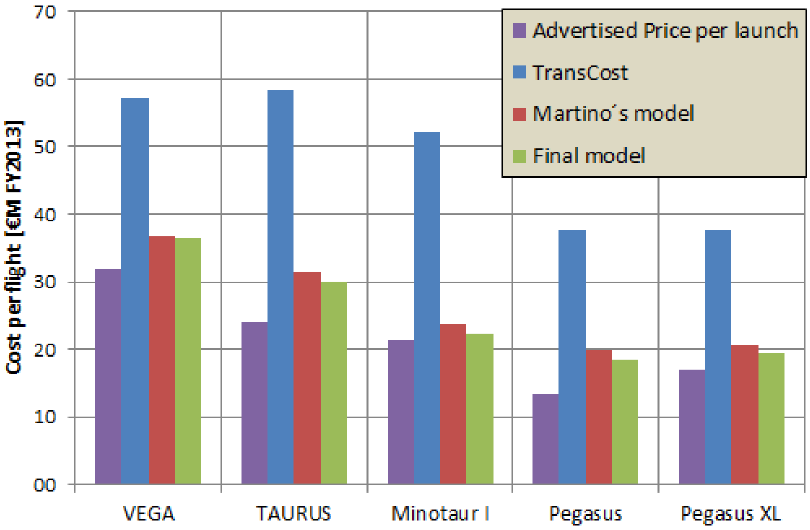

Figure 7: Error in estimated cost per flight for TransCost, Martino's model and the model resulting from this study

4. Validation

The various sub-models discussed in the previous section all have some inaccuracies in the estimated parameter value. An overview of inaccuracies in some main parameters is given in Table 3. The statistical figures used for this work are the mean error, μ, the absolute mean error, E, and the standard deviation of the error; σ.

Table 3: Statistical figures of the errors [%] in the parameters estimated

Parameter estimated	E	μ	σ
Vacuum thrust	6.46	-0.41	8.52
Specific impulse	3.68	+3.68	2.14
Propellant mass	6.83	+0.06	10.2
Stage length	11.7	+7.15	11.5
Stage inert mass	12.4	-2.76	13.8
VEB mass	30.9	+6.66	40.9
Fairing mass	14.5	-0.27	17.5
Drag coefficient	40.0	0.00	20.0
Lift coefficient	40.0	0.00	20.0
Cost per flight	19.1	+19.1	14.3

A preliminary overall tool validation has been performed using flight and design data obtained for the Taurus and Pegasus XL launch vehicle. Unfortunately, no data could be obtained for launch vehicles with a payload mass as targeted for in the present study. The inputs used for the Taurus rocket are shown in Table 4.

Table 4: Inputs for Taurus rocket analysis and optimization

Parameter				
Target orbit [km]	691			
Payload mass [kg]	1050			
Stage parameters		**Stage**		
	1	**2**	**3**	**4**
Propellant	HTPB/Al	HTPB/Al	HTPB/Al	HTPB/Al
Chamber pressure [bar]	85.9	58.6	55.8	39.4
Nozzle exit pressure [bar]	0.718	0.268	0.114	0.0866
Case diameter [m]	2.36	1.27	1.27	0.965
Nozzle exit diameter [m]	1.52	1.21	0.86	0.526
Burn time [s]	79.5	75	75.6	67.7
TVC [-]	Yes	Yes	Yes	Yes

Typical results as obtained for the Taurus and Pegasus XL are shown in Table 5. Results show that the tool results compare reasonably well with the observed (actual result). Largest difference is in the inert mass estimation of the Pegasus XL launch vehicle which differs roughly 40% with the observed value. This is mostly attributed to the inert mass estimation of the first stage which is about 50% too low. The reason for this is that in this study only composite motor cases have been taken into account, whereas the actual Taurus and Pegasus vehicles have aluminum cases. Optimizing the results furthermore showed that design changes (as of optimization) remained close to the actual design with only a marginally lower life cycle cost (~5%). Hence, this confirms that both Taurus and Pegasus XL are well designed vehicles.

Table 5: Observed versus estimated results (non-optimized) for two launch vehicles

	Taurus		**Pegasus XL**	
Parameter	**Observed**	**Estimated**	**Observed**	**Estimated**
Orbital altitude [km]	691	705	741	736
Maximum dynamic pressure [kPa]	NA	89.9	57.5	75.0
GTOW [kg]	72156	72292	23701	22727
Propellant mass stage 1 [kg]	48809	49358	15014	15451
Propellant mass stage 2 [kg]	12154	12374	3925	4155
Propellant mass stage 3 [kg]	3025	3086	770	772
Propellant mass stage 4 [kg]	782	772	-	-
Payload mass [kg	1050	1050	227	227
Fairing mass [kg]	400	524	194	181
Total inert mass [kg]	7386	6728	3992	2348
Total length	29.0	30.7	17.4	18.9
Cost per flight	24.0	30.1	16.9	19.3

5. Results

In this section, the results are presented for the design of an ALOSS-type of small launch vehicle; capable of launching a payload mass of 10 kg in a circular target orbit with orbital altitude 780 km. Results will be presented for a ground launch design as well as for an air launch design both designed for a launch from the equator in eastward direction. Like for ALOSS, it is assumed that the launch vehicle follows a direct ascent trajectory to orbit [34]. This means that the burnout conditions of the final stage are identical to the injection conditions of the desired orbit. No consideration is given to an alternative ascent trajectory where the satellite is first launched into a parking orbit of approximately 200 km altitude before the satellite is injected into the final orbit (Hohmann Transfer Ascent; HTA). Although this approach would allow further minimizing the required energy to reach final orbit [34], but given that this study only evaluates SRMs that are not re-ignitable makes the application of a HTA impractical. As airborne launch platform, the F-16 fighter aircraft is used. To limit the effect of the launch vehicle on the aircraft, it is required that the launch vehicle contour fits within the contours of an existing 370-gallon external fuel tank [17] with the total launch mass not exceeding 1,450 kg. In agreement with the results reported in [9], no attempt has been made of incorporating a wing. It is considered that an agile aircraft like the F-16 can achieve optimum release flight path angle, which negates the effect of a wing.

5.1 Constraints

Due to size and mass restriction related to the F-16's 370-gallon external fuel tank, the length and diameter of the launch vehicle are restricted to 5.5 m and 0.66 m, respectively and the maximum mass of the launch vehicle to 1,450 kg. Nominal release conditions are taken equal to those used for the ALOSS study; Release of the launch vehicle taking place at a flight velocity of 250 m/s at 15 km altitude under a flight path angle 50°. The following trajectory constraints are included: maximum acceleration (100 m/s^2), dynamic pressure (90 kPa) and Q·α limit (4,170 Pa·rad). In order to estimate the cost of a launch, a total of 120 launches over a period of 20 years (6 launches per year) are considered. This number of launches is considered a representative number based on the projected number of nanosatellites that require a launch in the near future, see earlier in this paper.

5.2 Optimized ALOSS-type of launch vehicle

First, the ALOSS design was analyzed using the design inputs as resulted from the ALOSS study [17]. It was found that when using the MDA tool as developed for this study results shown only a marginally higher total mass estimate as compared to the result reported by the NLR (1451 kg versus 1380 kg). The main reason for this was found to be in the estimation of the mass of the vehicle equipment bay and the fairing. The small difference in total mass though shows that the NLR models and the MDA models developed for this study are in reasonable agreement.

Next, an optimized three-stage air launched ALOSS was designed and compared with the original ALOSS design by the NLR. The results are given in Table 6 and clearly show that the optimized design offers better performance than the original ALOSS vehicle in terms of the cost per flight (-13.4%), the GTOW (-32.6%) and the amount of ΔV required to orbit (-2.66%).

Table 6: Summary of the most important characteristics of the baseline ALOSS vehicle and the optimized ALOSS vehicle

Parameter	ALOSS baseline [17]	Optimized ALOSS
Payload mass [kg]	10	10
Number of stages [-]	3	3
Cost per flight [€M]	2.182	1.890
GTOW [kg]	1,380	931
Vehicle length [m]	~5.5	5.42
Vehicle diameter [m]	0.66	0.56
Gravity loss [m/s]	1,920	1,67
Drag loss [m/s]	166	176
Total ΔV [m/s]	8,793	8,633

Finally, we varied the number of stages. However, results showed that two-stage vehicles are not capable of reaching the target orbit within the given GTOW constraints. Four-stage vehicles are found to be capable of reaching the target orbit, but they showed to be longer than the maximum allowed length of 5.5 m and more costly too. So it was concluded that a three-stage rocket is the best option.

5.3 Effect of initial flight path angle, release altitude and release flight velocity

In this section results will be presented showing the effect of flight path angle at release as well as release altitude and flight velocity.

Table 7 presents the results for various initial (at release) flight path angles at otherwise nominal release conditions (release altitude and velocity are 15.0 km and 250 m/s, respectively). Results clearly show that it is favorable to have a high initial flight path angle as this reduces drag loss. A too high value though leads to an increase in gravity loss which at high values of flight path angle off-sets the reduction in drag loss. Another interesting observation is that the cost per flight and GTOW for the different release flight path angles only varies with 2.6% and 9.2%. This confirms the flat optimum as also observed in [9].

Table 7: Important design characteristics for optimized air-launch in relation to initial flight path angle (release altitude 15 km, release velocity 250 m/s).

	0^o	25^o	50^o	75^o	90^o
Cost per flight [€M]	2.011	1.930	1.890	1.902	1.940
GTOW [kg]	1,128	1,017	931	924	968
Vehicle length [m]	5.47	5.44	5.42	5.40	5.48
Vehicle diameter [m]	0.66	0.61	0.56	0.62	0.58
Gravity loss [m/s]	1,589	1,586	1,673	1,875	1,820
Drag loss [m/s]	461	347	176	125	148
Total ΔV [m/s]	8,851	8,687	8,633	8,718	8,716

The effects of a change in release altitude and release flight velocity on the required amount of ΔV are:

• Reducing release altitude from 15 km to 10 km: ΔV required: +271 m/s,
• Reducing release velocity from 250 m/s to 200 m/s: ΔV required: +53 m/s,
• Increasing release velocity from 250 to 400 m/s: ΔV required: -151 m/s.

Clearly both increasing release altitude and increasing flight velocity reduces the required ΔV. It is noted though that this must be considered in relation to the capabilities of the carrier vehicle.

5.4 Air launch versus ground launch

Results obtained in this study for an optimized air- versus ground launch are given in Table 8. It shows that air launch significantly reduces the GTOW (with about 70%) as compared to ground launch. The reason for this reduction is that air-launch as compared to ground launch allows for significantly reduced ΔV to orbit. For the optimized air launch the ΔV advantage is 1,225 m/s compared with the optimized ground launch case. This number is in reasonable agreement with the value of about 900 m/s reported in [9] for a Minotaur I launch vehicle at comparable release conditions. This is in part attributed to that in [9] the Minotaur LV is evaluated in its original configuration, thereby neglecting the advantage that can be gained by optimizing the nozzle for altitude operation. Also the larger size of the Minotaur (36,200 kg) reduces the drag loss as compared to smaller launch vehicles and hence the gain that can be obtained for smaller launch vehicles is higher.

Table 8: Summary of important design characteristics for the optimized ground- and air-launched vehicle for various flight path angles (release altitude 15 km, release velocity 250 m/s).

	Ground launch	50^o	75^o
Cost per flight [€M]	2.628	1.890	1.902
GTOW [kg]	3,087	931	924
Vehicle length [m]	6.75	5.42	5.40
Vehicle diameter [m]	0.91	0.56	0.62
Gravity loss [m/s]	2,225	1,673	1,875
Drag loss [m/s]	558.5	176	125
Total ΔV [m/s]	9,858	8,633	8,718

The reduction in GTOW also leads to a significant cost reduction of about 30% as compared to ground launch. That cost is reduced less than GTOW is associated with the increase in operational cost of the air-launch vehicle.

6. Sensitivity analysis

To find out how sensitive the optimized results are for small changes in the design inputs and /or uncertainties resulting from the used models, see Table 3, both a one at a time (OAT) approach as well as a Monte-Carlo (MC) analysis has been performed. The optimized ALOSS rocket as given in Table 6 is taken as baseline for the sensitivity analysis.
In the OAT approach, we vary one parameter at a time while keeping all others constant, thereby neglecting any interaction between the parameters. The parameter is changed up and down by a percentage equal to the absolute mean error, *E* (worst-case scenario) [30]. Next, the vehicle was optimized for maximum payload mass (identical target orbit and separation conditions) and change in payload mass and GTOW are recorded. Change in GTOW for all cases investigated remained below 1%. However, payload mass was found to change significantly (up to -50%). Results show that uncertainty in thrust and specific impulse and VEB mass and sliver mass estimation influence payload mass most and therefore should be considered candidate for future improvement. Little or no effect has been found on life cycle cost.
In the MC analysis approach all the parameters are randomly varied at the same time. A Gaussian distribution with a mean error, μ, and standard deviation of the error, σ, is used to vary the parameters. The MC analysis results, see Figure 8, show an average cost of all cases found of 2.25 €M with σ = 0.175 €M. Average payload mass is ~11 kg with σ = 3.48 kg. Average GTOW found is 938.1 kg with σ = 10.9 kg. So we find that uncertainty in the inputs may significantly affect the cost and the payload mass. Variations in GTOW are much more limited.

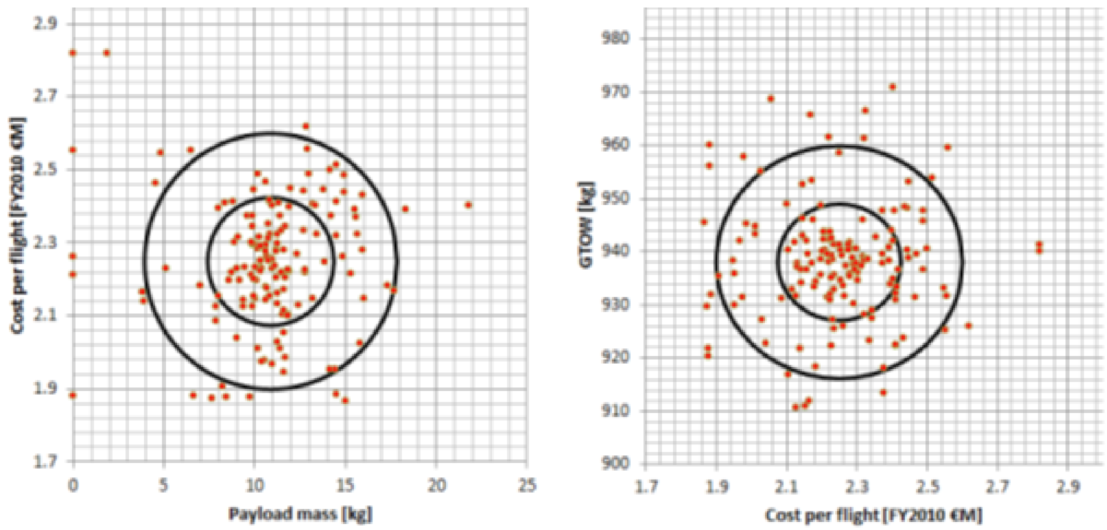

Figure 8: Scatter plot of results obtained for the Monte-Carlo analysis

As a final step in the sensitivity analysis, we performed a preliminary investigation on the effect of constraining web thickness in relation to case diameter assuming a cylindrical shaped propellant grain. The introduction of this constraint resulted in an increase in the cost per flight of ~2.5% and an increase of the GTOW of ~9% for both the optimized air and ground launched vehicle. This result thus shows that adding the constraint has no effect on the relative comparison of the two vehicles, but it shows that numbers reported in the results section are slightly optimistic.

7. Conclusions and future study

In this study, we have developed a MDA/MDO tool capable of analyzing/optimizing small solid rocket propelled multi-stage launch vehicles. The tool has been validated by comparing the tool outcomes with observed data for a number of existing small rocket launch vehicles. It is concluded that the tool's result are sufficiently close to reality to allow for meaningful optimization studies of small air- and ground launched vehicles. Still some differences do exist that require further study.
The tool developed in the study has been used to investigate an optimized air launched vehicle (ALOSS) with a payload mass of 10 kg released under high subsonic conditions comparable to those of the original ALOSS study conducted by NLR. Comparison of results shows good agreement between the tools results and the results reported in literature. It is also concluded that optimization shows potential for further reducing the launch costs and/or the GTOW.

We also investigated the difference between the performances of an air- and a ground-launched vehicle in the 10 kg payload mass range. It is concluded that air-launch holds high potential for reducing GTOW and cost per flight with a possible reduction in GTOW of 70% as compared to ground launch and 30% in cost per flight (from 2.6 M€ to 1.9 M€). Specific launch cost for a 10 kg payload air-launched payload are of order 190,000 €/kg, which is roughly a factor 3-4 higher than for a ride-share.
Uncertainties in the various sub-models have been determined and their effect investigated. It is found that the known uncertainties in the propulsion model (thrust and specific impulse estimation), and the sliver mass and VEB mass estimation models influences the results most. It is concluded though that the effect of the uncertainties remains limited. Moreover they apply to both ground- and air-launch and hence are not expected to affect the comparison of ground versus air-launch significantly. Still they do have an effect on the absolute cost and GTOW values. Some aspects, like the structural strengthening of a rocket associated with air launch and the trajectory constraints associated with separation, the actual steering of the rocket, and the burn program in relation to grain shape have not been considered in great detail yet. We recommend to continue validating the tool and where necessary to improve the modelling to increase the accuracy of the sub-models.
Furthermore, in this study only axisymmetric rockets and sequential staging has been considered. It is recommended to extend the tool so that both non-axisymmetric rockets as well as effects of parallel staging using multiple identical modules can be investigated in line with recommendations made in [32], [41]. Also the need for inclusion of small liquid/hybrid propellant propelled upper stages should be investigated as a means to allow for achieving improved injection accuracy and or higher payload performance within a given set of launch vehicle constraints. As a final recommendation, it is mentioned to include a risk module that would facilitate investigating the effects of differences in vehicle reliability on the trades performed in this study.

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