Weights and sizing assessment in the context of small launcher design

Alexandru-Iulian ONEL*[,1,2], Teodor-Viorel CHELARU[2]

*Corresponding author
*[,1]INCAS – National Institute for Aerospace Research "Elie Carafoli",
B-dul Iuliu Maniu 220, 061126, Bucharest, Romania,
onel.alexandru@incas.ro
[2]"POLITEHNICA" University of Bucharest, Faculty of Aerospace Engineering,
Str. Gheorghe Polizu 1-7, 011061, Bucharest, Romania,
teodor.chelaru@upb.ro

DOI: 10.13111/2066-8201.2020.12.3.11

Received: 24 March 2020/ Accepted: 31 July 2020/ Published: September 2020

***Abstract**: The paper presents mathematical models that can be used to quickly define preliminary key aspects regarding the sizing and weight characteristics of studied small launchers. The tool developed based on the proposed mathematical models can be used for standalone liquid propelled stage design or it can be integrated in an iterative multidisciplinary optimisation design scheme (MDO) for a preliminary small launcher design, able to insert the desired payload into a predefined orbit.*

***Key Words:** mathematical models, weight and sizing, small launcher, multidisciplinary optimisation, mass breakdown*

1. INTRODUCTION

In the context of small launcher preliminary design, the most efficient way to obtain an overall competitive launcher at the end of the process is by using a multidisciplinary design optimisation (MDO) approach. The current paper continues the work previously elaborated in [1], [2], [3], [4] and [5] by using the same MDO tool architecture, as depicted in Figure 1.

The developed MDO tool consists of four main modules that are assessed in a cascade order (Weights and Sizing, Propulsion, Aerodynamics and Trajectory) and several auxiliary modules (Inputs, Optimisation variables, Variable constraints, Objection function). Several mathematical models that can be used for quick component weight and sizing evaluations are detailed in this paper.

As observed in [1] and [6], MDO solution convergence usually occurs after several hundreds of thousands iterations; therefore a reduction in the complexity of the mathematical models implemented is desired. The main approach in reducing MDO complexity is by using a 3DOF dynamic model, which has a great impact on the number of key output data needed from each primary module.

In the Weight & Sizing module the main objective is to define preliminary key aspects regarding the sizing and weight characteristics of the analysed small launcher based on a reduced number of optimisation variables and input data.

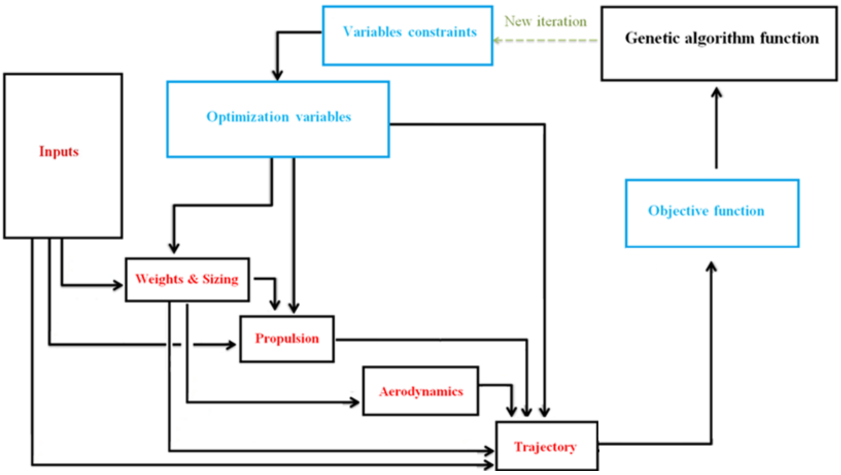

Figure 1 - Block scheme of small launcher MDO tool [1], [2], [3], [4], [5]

2. WEIGHTS AND SIZING ASSESMENT

A bottom-up strategy will be employed, the dimensions and masses of the launcher components being individually calculated. As depicted in Figure 2, this ensures that in the end, by summing the individual contributions, one can determine the dimensions and mass of each stage, and then of the entire *n*-stage launcher. The launcher breakdown scheme is valid for both sizing and weights assessment.

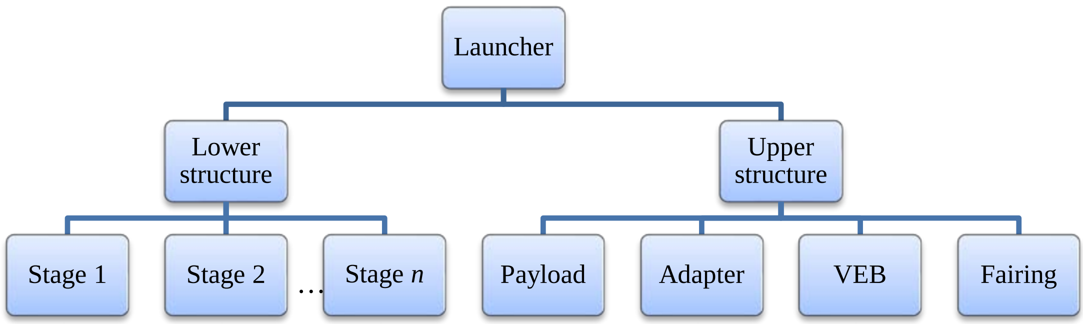

Figure 2 – Launcher breakdown scheme

For the upper structure, the dimensions and masses of each of the 4 individual components must be estimated in order to be integrated into the final small launcher architecture. In the context of a small launcher design optimisation (to insert a desired satellite into a predefined orbit), the payload dimensions and mass are known. Thus, from the ones listed above, the mathematical model for the last 3 components remains to be defined.

If the payload adapter specifications are not known before-hand, a semi-empirical mathematical model based on a total of 10 adapters from the launcher families Ariane, Vega, Soyuz and Atlas is implemented [7]. Thus, the mass of the adapter will be estimated with the help of the following relation:

$$M_{adapter} = 4.77536 \cdot 10^{-2} \cdot M_{payload}{}^{1.01317} \quad (1)$$

with $M_{payload}$ measured in kg.

The length of the adapter is approximated using:

$$L_{adapter} = \mathrm{k} \cdot D_{adapter} \tag{2}$$

with $D_{adapter}$ being equal to $D_{payload}$ and k depending on the upper structure complexity (for small launchers the value k =0.15 is proposed).

The avionics and electrical power system (EPS) required for the launcher to successfully accomplish the desired mission are located in its upper region, being part of either the last stage (upper stage) or the upper structure. They are located in the vehicle equipment bay (VEB), which can sometimes be integrated inside the satellite adapter. Such of an architecture can be considered feasible for small launchers, the length of the VEB area being thus negligible because it resides inside the payload adapter. However, the VEB mass cannot be neglected and is approximated, according to [8]:

$$M_{VEB} = 0.3672 \cdot M_{dry}{}^{0.6798} \tag{3}$$

with M_{dry} is the launcher dry mass and measured in kg.

The fairing geometry is defined based on the input data (such as preferred fairing profile, L/D ratios) and also the interior space required to house the payload (with known dimensions). For the fairing mass estimation, the following formula is proposed:

$$M_{fairing} = 7.12 \cdot S_{wet} \tag{4}$$

with S_{wet} is the fairing wetted area, measured in m^2.

A clear representation of the upper structure component breakdown is shown in Figure 3.

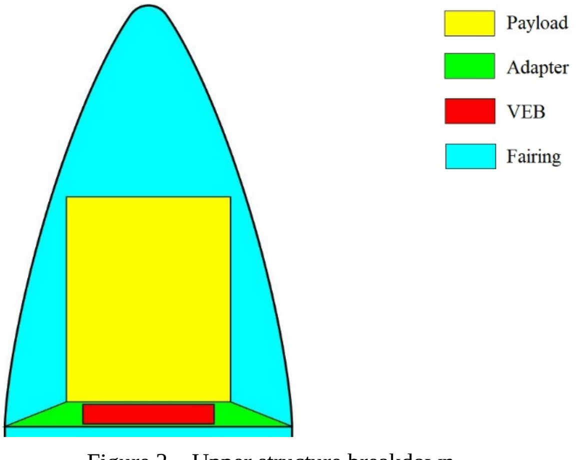

Figure 3 – Upper structure breakdown

For the lower structure, the dimensions and masses of each stage are computed separately and later added up. In this paper, liquid propelled stages are analysed, having the following component breakdown, as shown in Figure 4. The following data are considered to be part of the optimisation variables vector (for each stage): propellant mass M_{prop}, stage diameter D_{stage}, combustion chamber pressure P_c, exhaust pressure P_e, stage thrust to weight ratio $\frac{T}{W}$.

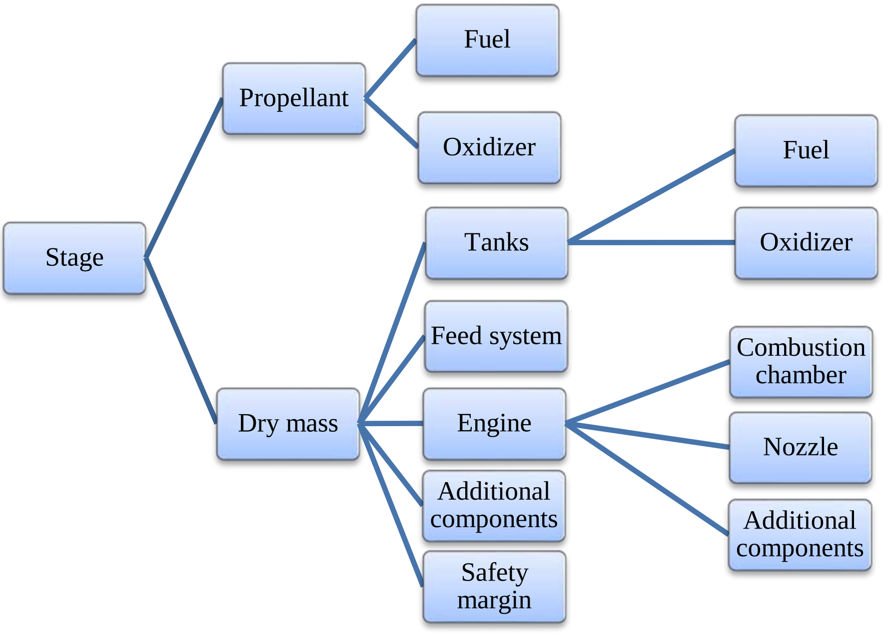

Figure 4 – Stage breakdown scheme

The oxidizer and fuel masses are computed with the aid of the following formulae:

$$M_{oxidizer} = \frac{R_m}{R_m + 1} M_{propellant}$$
$$M_{fuel} = \frac{1}{R_m + 1} M_{propellant} \tag{5}$$

where the optimal mixture ratio R_m can be obtained from [4]:

$$R_m = a + b \cdot {P_c}^c + d \cdot {P_e}^e \tag{6}$$

and (a, b, c, d, e) depend on the propellant combination used and detailed in [4].

The dry mass components are modelled based on empirical data and those from literature ([8], [9], [10]).

Both tanks are estimated with the same mathematical model, here being presented only the procedure for the oxidizer tank. First, the oxidizer volume and tank pressure are computed using:

$$V_{oxidizer} = \frac{M_{oxidizer}}{\rho_{oxidizer}}$$
$$P_{tank_{oxidizer}} = \left(10^{-0.1068\log(V_{oxidizer}) - 0.258810^6}\right) \cdot 10^6 \tag{7}$$

where: $\rho_{oxidizer}$ is the oxidizer density.

Then the thickness of the tank is computed using:

$$t_{tank} = F_s \frac{P_{tank} \cdot D_{tank}}{2\sigma_{tank}} \tag{8}$$

where: F_s is a safety factor (1.25), D_{tank} is the tank diameter (proposed 95% of stage diameter) and σ_{tank} is the ultimate strength of tank material.

For long stages, the proposed tank geometry is the one shown in Figure 5, while for short stages, a better geometry is found in Figure 6.

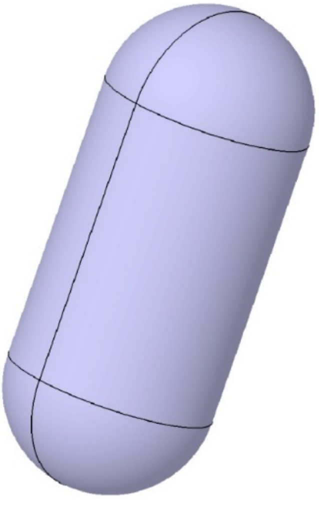

Figure 5 – Large tank

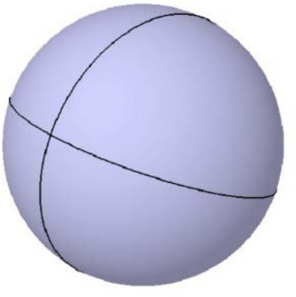

Figure 6 – Small tank

Based on the large tank (Figure 5) volume formula, one can obtain the length of the cylindrical part:

$$V_{tank} = \pi \left(\frac{D_{tank}}{2}\right)^2 L_{cyl_{tank}} + \frac{4}{3}\pi \left(\frac{D_{tank}}{2}\right)^3 \tag{9}$$

and if the length of the cylindrical part $L_{cyl_{tank}} < 0$, then a spherical tank is used.

After computing the tank surface with:

$$S_{tank} = \pi D_{tank} L_{cyl_{tank}} + 4\pi \left(\frac{D_{tank}}{2}\right)^2 \tag{10}$$

one can finally obtain the mass of the tank:

$$M_{tank} = t_{tank} \cdot S_{tank} \cdot F_{tank} \cdot \rho_{tank} \tag{11}$$

where: F_{tank} is a safety factor (1.2) and ρ_{tank} is the density of tank material.

The role of the feed system is to raise the propellant pressure to that required in the combustion chamber, but also to transport it to the combustion zone.

Feeding the oxidant and fuel to the combustion chamber can be achieved using one of the following 3 technical solutions: pressure fed systems, mechanical turbopumps or electrical pumps.

The simplest technical solution is the pressure fed system, where an additional pressuring tank (usually filled with Helium) is used.

This solution is viable for small upper stages, where the propellant mass is low. A similar procedure to that presented for the propellant tanks is used, with some modifications to include an interior liner. Details can be observed in [9].

The baseline feed system solution is that of mechanical turbopumps which can be seen on most of the small, medium and large launchers.

Based on papers such as [9] and [10], a semi-empirical model can be imagined, separate fuel and oxidizer pumps being modelled. First, the oxidizer and fuel mass flows are computed using:

$$q_{oxidizer} = \frac{R_m}{R_m + 1} q_{propellant}$$
$$q_{fuel} = \frac{1}{R_m + 1} q_{propellant} \tag{12}$$

The propellant mass flow $q_{propellant}$ of the liquid rocket engine is considered to be an input data of the Wights & Sizing module, being derived from the optimization variable $\frac{T}{W}$. The necessary pump power is computed using:

$$P_{nec_{fuel}} = \frac{q_{fuel}\left(P_c - P_{tank_{fuel}}\right)}{\rho_{fuel} \cdot \eta_p}$$
$$P_{nec_{oxidizer}} = \frac{q_{oxidizer}\left(P_c - P_{tank_{oxidizer}}\right)}{\rho_{oxidizer} \cdot \eta_p} \tag{13}$$

where: P_c is the combustion chamber pressure, η_p is the pump efficiency (75% - 80%).

Finally, the mass of the turbopumps can be estimated by:

$$M_{turbopump_{fuel}} = 1.5\left(\frac{P_{nec_{fuel}}}{\frac{2N\pi}{60}}\right)^{0.6}$$
$$M_{turbopump_{oxidizer}} = 1.5\left(\frac{P_{nec_{oxidizer}}}{\frac{2N\pi}{60}}\right)^{0.6} \tag{14}$$

with N being the pump rotational speed (12400 $\frac{rot}{min}$).

Some progress has been realised in the past years towards the use of electric pump feed systems, but the technical solution is not yet fully developed. If a feed system based on electric pumps is desired, then it is necessary to use at least 3 main components: electric motor, inverter and batteries.

A model similar to that used for the calculation of turbopumps can be a starting point, making an analogy with turbopump power supply for calculating the power required to feed the combustion chamber with fuel and oxidant.

Possible mathematical models for electric pumps are presented in works such as [11] or [12].

The liquid rocket engine or motor is a very complex component of the stage assembly. Depending on the combustion chamber pressure, the mass of the engine sub-assembly (combustion chamber + nozzle) is increased by a correction factor $\frac{1}{\xi}$ between 2.5 and 5, according to [9] and [10], to include additional components, such as the injector, the ablative heat shield, but also other smaller components. Thus, the mass of the rocket motor assembly is estimated using:

$$M_{engine} = \frac{M_{combustion_{chamber}} + M_{nozzle}}{\xi} \tag{15}$$

with:

$$\xi = \begin{cases} 0.2 & , if\ P_c \leq 20\ bar \\ \dfrac{0.2P_c + 2}{30} & , if\ 20\ bar < P_c < 50\ bar \\ 0.4 & , if\ P_c \geq 50\ bar \end{cases} \tag{16}$$

To obtain the combustion chamber characteristics, the mathematical model from [9] and [10] is used. The nozzle throat cross-sectional area A_c is computed with the aid of:

$$A_c = \frac{q_{propellant}\sqrt{\gamma R T_c}}{\eta_c \cdot P_c \cdot \gamma \cdot \left(\frac{2}{\gamma + 1}\right)^{\frac{\gamma+1}{2\gamma-2}}} \tag{17}$$

where: R = mixture gas constant, T_c = flame tempretaure, η_c = combustion efficienc (96% - 99%), P_c = combusiton chamber pressure, γ = isentropic coefficient.

For typical liquid propellant combinations, data can be obtained from [4], with simple two-variable power functions expresions such as equation (6).

After obtaining the nozzle throat diameter D_c, the combustion chamber cross-sectional area is computed using:

$$A_{c_{ch}} = (8 \cdot (100D_c)^{-0.6} + 1.25)\pi \frac{{D_c}^2}{4} \tag{18}$$

After obtaining the combustion chamber diameter $D_{c_{ch}}$, its length can be approximated using:

$$L_{c_{ch}} = L^* \left(\frac{D_c}{D_{c_{ch}}}\right)^2 \tag{19}$$

The characteristics length L^* depends on the liquid propellant combination used. Typical values can be found in Table 1 and for conservative reasons the maximum values are proposed to be used.

Table 1 – Characteristic length for different liquid propellant combinations

Oxidizer	**Fuel**	**L^* [m]**	**References**
LOX	Kerosene	1.02 - 1.27	[11], [13], [14]
LOX	Methane	0.84 – 1.51	[15]
LOX	Hydrogen	0.76 – 1.02	[9], [11], [13], [14]
LOX	Ethanol	2.5 - 3	[16], [17], [18]

The thickness of the combustion chamber can now be calculated with the following formula:

$$t_{c_{ch}} = P_c f_c \frac{D_{c_{ch}}}{2\sigma_{ch}} \tag{20}$$

where: f_c is a safety factor (proposed 2) and σ_{ch} is the ultimate strength of chamber material.

Finally, the mass of the combustion chamber can be approximated with:

$$M_{combustion_{chamber}} = \left(2A_{c_{ch}} + \pi D_{c_{ch}} L_{c_{ch}}\right) \cdot \mathrm{t}_{c_{ch}} \cdot \rho_{c_{ch}} \tag{21}$$

with $\rho_{c_{ch}}$ being the density of combustion chamber material.

Numerous types of nozzles concepts can be used depending on the launch application such as: conical nozzles, bell nozzles, dual bell nozzles, expansion-deflection nozzles, multi-position nozzles aerospike nozzles, each with its own advantages and disadvantages. A comprehensive list of possible nozzle configurations can be found in [19].

With small launchers, when choosing the optimal nozzle architecture to implement, the usual deciding criteria is its cost and simplicity. The simplest architecture from a technical point of view, but also with a very low production cost corresponds to the standard conical nozzle. In this paper, a conical nozzle will be used, and the proposed mathematical model for weight and sizing evaluations presented. The nozzle expansion ratio ε is computed using:

$$\varepsilon = \frac{\left(\frac{2}{\gamma+1}\right)^{\frac{1}{\gamma-1}} \cdot \frac{P_c}{P_e}^{\frac{1}{\gamma}}}{\sqrt{\left(\frac{\gamma+1}{\gamma-1}\right) \cdot \left(1 - \left(\frac{P_e}{P_c}\right)^{\frac{\gamma-1}{\gamma}}\right)}} \tag{22}$$

Then, the cross-sectional area A_n and diameter D_n of the nozzle exhaust is computed using:

$$\begin{gathered} A_n = A_c \varepsilon \\ D_n = 2\sqrt{\frac{A_n}{\pi}} \end{gathered} \tag{23}$$

The nozzle length L_n is estimated with:

$$L_n = \frac{D_n - D_c}{2 \cdot \tan \lambda} \tag{24}$$

with λ the nozzle half angle (considered 15°).

The thickness of the nozzle can now be estimated with the aid of the following formula:

$$t_n = \frac{0.5 \cdot P_c \cdot D_n \cdot f_n}{2\sigma_n} \tag{25}$$

where: f_n is a safety factor (proposed 2) and σ_n is the ultimate strength of nozzle material.

Finally, the nozzle mass is obtained using:

$$M_{nozzle} = \frac{\pi(D_n + D_c) \cdot L_n \cdot \mathrm{t}_n \cdot \rho_n}{2} \tag{26}$$

In addition to the dry mass obtained so far from the major components (tanks, feed system, engine) the smaller components which exist on board of the launcher stages must be also taken into account.

This additional mass includes components such as: TVC (thrust vector control), guidance system, aerodynamic casing, inner pipes but also other small components. The additional mass on board a launcher stage with liquid propulsion, can be approximated by the relation from [9]:

$$M_{add} = M_{propellant}\left(-2.3 \cdot 10^{-7} M_{propellant} + 0.07\right) \tag{27}$$

The above relationship is valid for propellant masses up to 300 t, few stages being outside the validity of the formula. The stages with a higher propellant masses correspond to the very large launchers, not being of interest for the current work. A mass safety margin of 5% and a length safety margin of 10% are also proposed to be implemented to boost the overall confidence of the MDO assessment (can be lowered or set to 0 in the detailed phase of small launcher design).

3. TEST CASES AND RESULTS

The results obtained with the proposed mathematical models will be validated with the aid of already existing launcher constructive solutions (which have been used in the past or are currently used). In order to demonstrate the flexibility of the proposed model, both small and large stages will be analysed. A total of 8 test cases are used for which reference data could be gathered. Propellant mass and stage diameter are part of the input data. Stage length and dry mass are part of the output data.

The first 2 stages analysed are those corresponding to the first and second liquid propelled stages of the Atlas V launcher. The Atlas V boosters were based on HTPB solid propellant and cannot be approximated with the current model. The first stage, also known as the Atlas CCB, is built around a RD-180 engine which uses kerosene as fuel. Propellant mass is 284 t, the burn time is 253 seconds and any other additional data are gathered from [20], [21], [22].

A very good accuracy of the results, compared with the reference data can be seen in Table 2. The second stage of the Atlas V launcher is much smaller than the previous one with about 20.8 t of fuel on board. The upper stage is called Centaur, based on the RL-10A-4-2 engine that uses the combination of liquid oxygen and liquid hydrogen to generate thrust. The comparison of the reference results ([20], [21]) with those obtained using the proposed model is observed in Table 3. The errors are significantly larger compared to the first stage, and an 11% error can be observed in the estimation of the structural mass. Being an overestimation of the mass, it is considered that the proposed model has generated a conservative value.

Table 2 – Atlas CCB data comparison

	Reference values	Obtained values
Stage mass [t]	305.14	304.59
Propellant mass [t]	284.09	284.09
Dry mass [t]	21.05	20.50
Stage length [m]	32.46	32.19
Stage diameter [m]	3.81	3.81

Table 3 – Atlas Centaur data comparison

	Reference values	Obtained values
Stage mass [t]	23.15	23.40
Propellant mass [t]	20.83	20.83
Dry mass [t]	2.32	2.57
Stage length [m]	12.68	13.29
Stage diameter [m]	3.05	3.05

The next launcher used to validate the proposed mathematical model is Ariane 5. Numerous variants of Ariane 5 have been used over time. The first one studied is Ariane 5, version G. Again, excluding the solid propellant boosters, the first stage with liquid propellant is known as EPC H158, being made up of a Vulcain type engine, with liquid hydrogen as fuel. The comparison of the reference results ([23], [24]) with those obtained with the proposed model is observed in Table 4. An average error of about 7% can be observed in estimating the structural mass and the stage length. The second version of Ariane 5 proposed for the validation of the results is Ariane 5, ES. The first stage with liquid fuel is called EPC E (H173), being developed on the Vulcain 2 engine architecture. The comparison of reference results ([23], [25]) with those obtained using the proposed model is observed in Table 5. The maximum error is about 6% in the estimation of the structural mass, the length of the stage being estimated with a very high accuracy.

Table 4 – Ariane 5 EPC H158 data comparison

	Reference values	Obtained values
Stage mass [t]	170.5	171.27
Propellant mass [t]	158.3	158.3
Dry mass [t]	12.2	12.97
Stage length [m]	30.5	28.16
Stage diameter [m]	5.4	5.4

Table 5 – Ariane 5 EPC H173 data comparison

	Reference values	Obtained values
Stage mass [t]	184.7	183.82
Propellant mass [t]	170	170
Dry mass [t]	14.7	13.82
Stage length [m]	30.5	30.14
Stage diameter [m]	5.4	5.4

Another very important launcher used in Europe is Ariane 4, being the predecessor of Ariane 5 and recording a total of 113 successful missions out of a total of 116 launches. The only stage of interest for the current work is the upper one, known as the H10 stage, the version analysed being the final one, H10-3, based on a HM7-B engine type. The comparison of the reference results [26] with those obtained using the proposed model is presented in Table 6. A maximum error of about 8% can be observed in the estimation of the structural mass, the length of the stage being estimated again with a very good accuracy.

One commonly used stage family is the DCSS (Delta Cryogenic Second Stage). DCSS-type stages have been used on-board the Delta III and Delta IV launcher and will continue to be used on the SLS (Space Launch System) as the upper stage. The thrust is generated by a Pratt & Whitney RL10B-2 engine, its technical details being obtained from [27]. Analysing the upper stage of the Delta III launcher, the comparison of the reference results ([28], [29]) with those obtained using the proposed model is observed in Table 7. A maximum error of less than 3% can be observed in the dry mass estimation, but an error of about 1m in length estimation.

Table 6 – Ariane 4 H10-3 data comparison

	Reference values	Obtained values
Stage mass [t]	13.5	13.36
Propellant mass [t]	11.8	11.8
Dry mass [t]	1.7	1.56
Stage length [m]	11.05	11.14
Stage diameter [m]	2.6	2.6

Table 7 – Delta III DCSS data comparison

	Reference values	Obtained values
Stage mass [t]	19.3	19.23
Propellant mass [t]	16.82	16.82
Dry mass [t]	2.48	2.41
Stage length [m]	8.8	9.96
Stage diameter [m]	4	4

For the Delta IV launcher, two versions of the DCSS stage have been used so far, the first having a diameter of 4m and the other having a diameter of 5m.

For the DCSS 4m version, the results obtained from the literature ([28], [29]) and those obtained with the proposed model are presented in Table 8. A very good estimate of the dry mass is observed.

For the DCSS 5m version, the results obtained from the literature ([28], [29]) and those obtained with the proposed model are presented in Table 9. For this stage, a maximum error of 11% can be observed in estimating the structural (dry) mass and about 6% in estimating its length.

Table 8 – Delta IV DCSS 4m data comparison

	Reference values	Obtained values
Stage mass [t]	24.17	24.19
Propellant mass [t]	21.32	21.32
Dry mass [t]	2.85	2.87
Stage length [m]	12.2	11.15
Stage diameter [m]	4	4

Table 9 – Delta IV DCSS 5m data comparison

	Reference values	Obtained values
Stage mass [t]	30.71	31.11
Propellant mass [t]	27.22	27.22
Dry mass [t]	3.49	3.89
Stage length [m]	13.7	12.83
Stage diameter [m]	5	5

The absolute errors between the reference and computed data for these 8 stages are tabulated in Table 10, based on the data presented in Table 2 - Table 9.

An average error of 6.1% was found in stage dry mass estimation and 5.4% in stage length estimation.

For a better visualization of the differences between the reference values and those obtained with the proposed model, the stages dry masses are observed in Figure 7, and their lengths in Figure 8.

Based on the data presented in this paper, the proposed model for the Weight & Sizing module of the MDO algorithm is validated, the average mean error being about 5.8%.

Table 10 – Proposed model absolute errors

Launcher	Stage	Dry mass error [%]	Length error [%]	Average error [%]
Atlas V	Atlas CCB	2.61	0.83	1.72
Atlas V	Centaur	11.01	4.81	7.91
Ariane 5, G	EPC H158	6.31	7.67	6.99
Ariane 5, ES	EPC H173	5.99	1.18	3.58
Ariane 4	H10-3	8.24	0.81	4.52
Delta III	DCSS	2.82	13.18	8.00
Delta IV	DCSS, 4m	0.70	8.61	4.65
Delta IV	DCSS, 5m	11.46	6.35	8.91
All stages		**6.14**	**5.43**	**5.79**

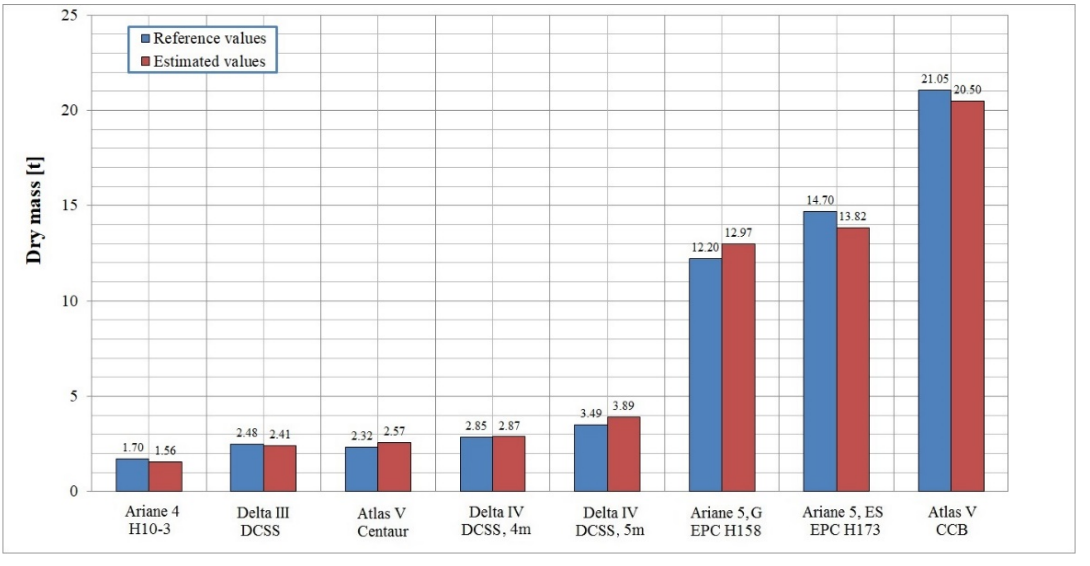

Figure 7 – Weights estimation

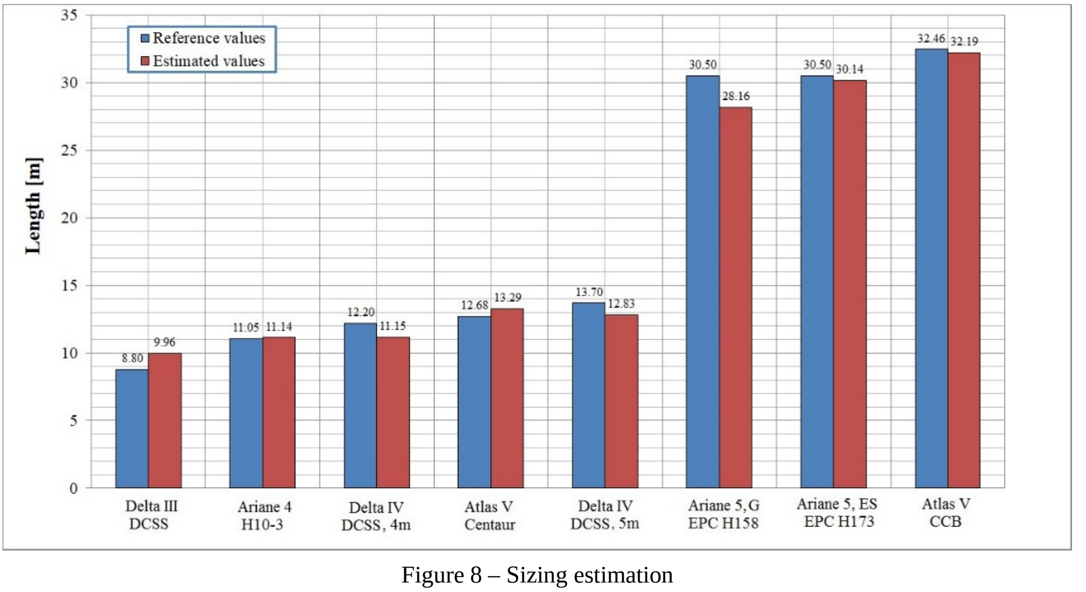

Figure 8 – Sizing estimation

4. CONCLUSIONS

The paper continues the work previous done in [1], [2], [3], [4] and [5] extending the capabilities of a multidisciplinary optimisation tool for small launchers design. Mathematical models for weights and sizing assessment are presented, results being validated with the aid of 8 launcher stages of interest reference data. An average error of 6.1% was found in stage dry mass estimation and 5.4% in stage length estimation, corresponding to a mean average model error of 5.8%.

REFERENCES

[1] A. I. Onel, T. P. Afilipoae, A. M. Neculăescu, M. V. Pricop, MDO approach for a two-stage microlauncher, *INCAS Bulletin*, ISSN 2247–4528, vol. **10**, issue 3, pp. 127-138, 2018, https://doi.org/10.13111/2066-8201.2018.10.3.11.

[2] A. I. Onel, T. P. Afilipoae, A. M. Neculăescu, M. V. Pricop, Drag coefficient modelling in the context of small launcher optimisation, *INCAS Bulletin*, ISSN 2247–4528, vol. **10**, issue 4, pp. 103-116, 2018, https://doi.org/10.13111/2066-8201.2018.10.4.10.

[3] A. I. Onel, T. V. Chelaru, Aerodynamic assessment of axisymmetric launchers in the context of multidisciplinary optimisation, INCAS Bulletin, ISSN 2247–4528, vol. **12**, issue 1, pp. 135-144, 2020, https://doi.org/10.13111/2066-8201.2020.12.1.13.

[4] A. I. Onel, O. I. Popescu, A. M. Neculăescu, T. P. Afilipoae, T. V. Chelaru, Liquid rocket engine performance assessment in the context of small launcher optimisation, *INCAS Bulletin*, ISSN 2247–4528, vol. **11**, issue 3, pp. 135-145, 2019, https://doi.org/10.13111/2066-8201.2019.11.3.12.

[5] A. I. Onel, T. V. Chelaru, Trajectory assessment and optimisation in the context of small launcher design, INCAS Bulletin, ISSN 2247–4528, vol. **12**, issue 2, 2020, https://doi.org/10.13111/2066-8201.2020.12.2.10.

[6] T. P. Afilipoae, A. M. Neculăescu, A. I. Onel, M. V. Pricop, A. Marin, A. G. Perșinaru, A. M. Cișmilianu, I. C. Oncescu, A. Toader, A. Sirbi, S. Bennani, T. V. Chelaru, Launch Vehicle - MDO in the development of a Microlauncher, *Transportation Research Procedia*, vol. **29**, pp. 1-11, 2018.

[7] F. Castellini, *Multidisciplinary design optimization for expendable launch vehicles*, Politecnico Di Milano Aerospace Engineering Department Doctoral Program In Aerospace Engineering, Milano, 2012.

[8] M. W. van Kesteren, B. T. C. Zandbergen, Design and Analysis of an Airborne, solid Propelled, Nanosatellite Launch Vehicle using Multidisciplinary Design Optimization, *6th European Conference For Aeronautics And Space Sciences (Eucass)*, 2015.

[9] M. Balesdent, *Multidisciplinary Design Optimization of Launch Vehicles*, Ecole Centrale de Nantes, 2011.

[10] R. Humble, G. Henry, W. Larson, *Space Propulsion Analysis and Design*, Mc-Graw Hill, 1995.

[11] P. Rachov, *Electric Feed Systems for Liquid Propellant Rocket Engines*, Buenos Aires, December 6th, 2010.

[12] G. Waxenegger-Wilfing, R. H. S. Hahn, J. Deeken, *Studies on electric pump-fed liquid rocket engines for micro-launchers*, April 2018.

[13] D. K. Huzel, D. H. Huang, *Modern Engineering for Design of Liquid-Propellant Rocket Engines*, 1st ed., Washington DC: AIAA, 1992.

[14] D. K. Huzel, D. H. Huang, *Design of Liquid-Propellant Rocket Engines*, Washington DC: Rocketdyne Division, North American Aviation Inc., 1967.

[15] A. Preuss et al, *LOX/Methane Technology Efforts for Future Liquid Rocket Engines*, 5th International Spacecraft Propulsion Conference, DLR.

[16] M. Barrčre et al, *Raketenantriebe*, Elsevier Publishing Company, 1961.

[17] J. Sieder et al, *Proceedings of the smart rockets project: design development and first measurement results of a 500N lox/ethanol combustion chamber*, Deutscher Luft- und Raumfahrtkongress, 2014.

[18] K. Sakaki et al, Fundamental Combustion Characteristics of Ethanol/Liquid Oxygen Rocket Engine Combustor with Planar Pintle-type Injector, *Trans. Japan Soc. Aero. Space Sci*., vol. **58**, nr. 1, 2015.

[19] P. Rachov, *Sizing and design of a liquid propellants rocket engine*, Buenos Aires, December 2011

[20] * * * Wikipedia, *Atlas V*, https://en.wikipedia.org/wiki/Atlas_V.

[21] * * * United Launch Alliance, *Atlas V Launch Services User's Guide*, 2010, https://www.ulalaunch.com/docs/default-source/rockets/atlasvusersguide2010.pdf.

[22] * * * Space Launch Report, *Atlas 5 Data Sheet*, https://www.spacelaunchreport.com/atlas5.html

[23] * * * Wikipedia, *Ariane 5,* https://en.wikipedia.org/wiki/Ariane_5.

[24] * * * Space Launch Report, *Ariane 5 Data Sheet*, http://www.spacelaunchreport.com/ariane5.html.
[25] * * * Space Flight 101 Team, *Ariane 5 ES*, http://www.spaceflight101.net/ariane-5-es.html.
[26] * * * N. Brügge, *Russian space-rocket and missile liquid-propellant engines*, http://www.b14643.de/Spacerockets/Specials/Russian_Rocket_engines/engines.htm.
[27] * * * M. Wade, Encyclopedia Astronautica, *RL-10B-2*, 2011
[28] * * * R. A. Braeunig, *Space Launchers-Delta*, http://www.braeunig.us/space/specs/delta.htm.
[29] * * * Wikipedia, *DCSS*, https://en.wikipedia.org/wiki/Delta_Cryogenic_Second_Stage.