Liquid rocket engine performance assessment in the context of small launcher optimisation

Alexandru-Iulian ONEL*,[1], Oana-Iuliana POPESCU[1], Ana-Maria NECULAESCU[1], Tudorel-Petronel AFILIPOAE[1], 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*, popescu.oana@incas.ro, neculaescu.ana@incas.ro,
afilipoae.tudorel@incas.ro
[2]"POLITEHNICA" University of Bucharest, Faculty of Aerospace Engineering,
Str. Gheorghe Polizu 1-7, 011061, Bucharest, Romania,
teodor.chelaru@upb.ro

DOI: 10.13111/2066-8201.2019.11.3.12

Received: 26 June 2019/ Accepted: 09 August 2019/ Published: September 2019

***Abstract**: The paper presents a fast mathematical model that can be used to quickly asses the propulsive characteristics of liquid propelled rocket engines. The main propulsive parameters are computed using combustion surfaces obtained after a nonlinear data fitting analysis. This approach is much more time efficient than using standard codes which rely on frequent calls of the Fuel Combustion Charts and interpolating their data. The tool developed based on the proposed mathematical model can be used separately or it can be integrated in a multidisciplinary optimisation algorithm for a preliminary microlauncher design.*

Key Words: *rocket propulsion, specific impulse, nonlinear surface fitting, small launchers, multidisciplinary optimisation, mathematical model*

1. INTRODUCTION

Designing a competitive and efficient small launcher, even in the preliminary conceptual phase, is a great challenge due to the fact that the complexity of the system does not significantly scale down with the decrease in payload mass [1]. A preliminary design for such a small launcher is of interest due to the increased recently demand and increase in the small satellites market, which has been booming in the last years.

A multidisciplinary approach must be used to successfully obtain a preliminary design of the microlauncher, which is often realised with the aid of a multidisciplinary design optimisation (MDO) algorithm. The MDO algorithm can vary from author to author, the block scheme for the MDO algorithm used in papers [1] and [2] can be seen in Figure 1, as an example. Here, the MDO employs an intrinsic Matlab genetic algorithm in order to search for the global optimum for the launch vehicle design problem.

The MDO tool developed core is constituted by the four main disciplines that are assessed in a cascade order: Weights & Sizing, Propulsion, Aerodynamics and Trajectory.

The objective of this paper is to present the mathematical model that can be used in the assessment of the small launchers propulsive performance, which takes place in the Propulsion block of the MDO scheme presented in Figure 1. For solution convergence, it can take up to several hundred thousand iterations [2]. Reducing the complexity of the mathematical models used is thus a very important aspect to consider.

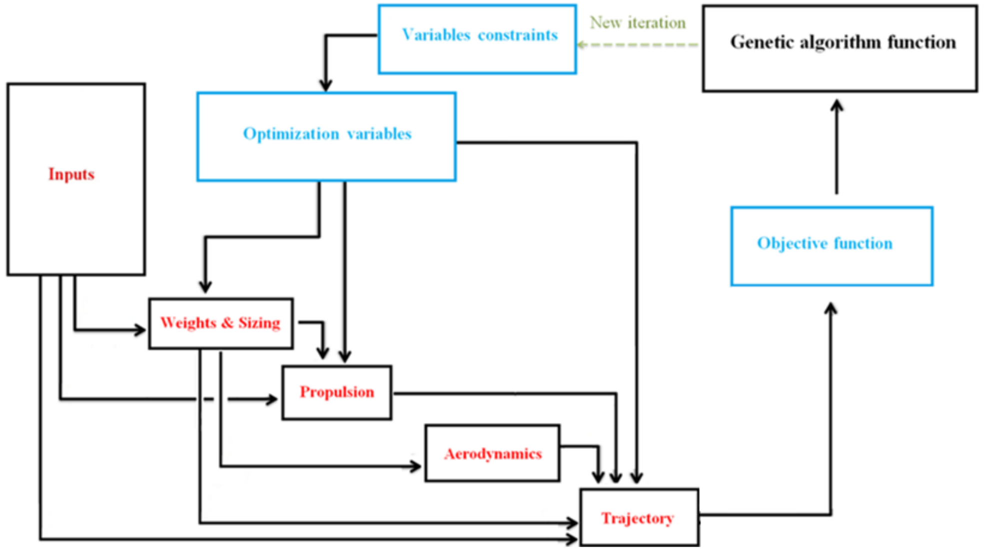

Figure 1 - Block scheme of a MDO algorithm

2. PROPULSION ANALYSIS

Thrust curve evaluations at different altitude regimes must be accurately computed using simple mathematical models to ensure that the MDO tool can successfully asses the overall launcher performance and later optimise its configuration. For this study a total of 4 liquid propellant combinations are assessed, with oxygen as oxidizer and kerosene, methane, hydrogen and ethyl alcohol/ethanol as fuel.

The thrust of the rocket engine T is computed with the following formula:

$$T = q \cdot g_0 \cdot I_{sp} \tag{1}$$

The specific impulse I_{sp} is computed from papers [2] , [3], [4] and [5] with:

$$I_{sp} = \eta_n \cdot \frac{C^*}{g_0} \left(\gamma_t \sqrt{\left(\frac{2}{\gamma_t - 1}\right) \cdot \left(\frac{2}{\gamma_t + 1}\right)^{\left(\frac{\gamma_t+1}{\gamma_t-1}\right)} \cdot \left(1 - \left(\frac{P_e}{P_c}\right)^{\left(\frac{\gamma_t-1}{\gamma_t}\right)}\right)} + \frac{\varepsilon}{P_c}\left(P_e - P_a\right) \right) \tag{2}$$

Where: q, g_0 are the propellant mass flow rate and the gravitational acceleration at sea level; η_n is the nozzle efficiency, C^* is the propellants characteristic velocity, γ_t is the isentropic coefficient (also known as specific heat ratio) at the throat and ε is the nozzle expansion ratio; P_c represents the chamber pressure, P_e represents the exhaust pressure and P_a represents the atmospheric pressure.

The last term in equation 136 represents the correction with altitude of the specific impulse. At low altitudes, the atmospheric pressure is high and thus the thrust generated by the rocket engine is significantly lower than of that obtained at high altitudes, near vacuum conditions.

The nozzle area ratio or expansion ratio is computed with the formula given below:

$$\varepsilon = \frac{\left(\frac{2}{\gamma_t + 1}\right)^{\frac{1}{\gamma_t - 1}} \cdot \left(\frac{P_c}{P_e}\right)^{\frac{1}{\gamma_t}}}{\sqrt{\left(\frac{\gamma_t + 1}{\gamma_t - 1}\right) \cdot \left[1 - \left(\frac{P_e}{P_c}\right)^{\frac{\gamma_t - 1}{\gamma_t}}\right]}} \quad (3)$$

The propellant characteristic velocity is computed based on the following formula:

$$C^* = \eta_c \cdot \frac{\sqrt{\gamma_t \cdot R \cdot T_f}}{\gamma_t \cdot \left(\frac{2}{\gamma_t + 1}\right)^{\frac{\gamma_t + 1}{2(\gamma_t - 1)}}} \quad (4)$$

Where: η_c is the combustion efficiency, R is the exhaust gas constant and T_f is the adiabatic flame temperature.

The exhaust gas constant is considered to be ratio between the universal gas constant R_u and the gas molecular weight M_w. The adiabatic flame temperature is the temperature achieved by a combustion reaction that takes place adiabatically for the given reactants, measured in Kelvin. The exhaust gas molecular weight is the average molar weight of the combustion products, being the ratio between the mass of the exhaust gas divided by the number of moles, measured in kg/kmol.

The set of data shown in Table 1 is used for some of the parameters that appear in the presented model.

Table 1 – Key model parameters

Key parameter	Value
g_0	$9.80665\ \frac{m}{s^2}$
P_a	0 atm for vacuum conditions 1 atm for sea level conditions
η_n	98%
η_c	96% for LOX/ethanol; 96% for LOX/kerosene; 98% for LOX/methane; 99% for LOX/hydrogen
R_u	$8314.3\ \frac{J}{kmol \cdot K}$

The propellant mass flow, together with the chamber and exhaust pressures are considered to be optimisation variables in the multidisciplinary optimisation context. Based on these 3 parameters and the flight regime, thrust generated by the liquid rocket engine can be computed. This is possible because the four main propulsive parameters (mixture ratio, adiabatic flame temperature, specific heat ratio at the throat and the molecular mass at combustion) can be approximated with the aid of propellant combustion charts [6].The mixture ratio R_m does not explicitly appear in the mathematical model, but the other 3 propulsive parameters are computed with respect to it.

The combustion charts are generated after a thermochemical equilibrium analysis with the aid of STANJAN code [7]. The thermochemical data are based on the JANAF Thermochemical Tables [8]. For specific impulse computations only the chamber pressure P_c and exhaust pressure P_e are needed.

3. NONLINEAR APPROXIMATION FUNCTION

The typical approach to obtain the main four propulsive parameters is by directly calling the combustion charts and interpolating their data multiple times. Doing this several million times until the launcher configuration converges, the need for a simpler model arises.

Thus, this paper proposes to determine a two-variable nonlinear function that provides accurate results for the pressure range of interest.

Figure 2 shows the combustion charts for one of the four bipropellant combinations studied (LOX/kerosene), as presented in [6]. The data were extracted from these charts using a plot digitizer tool [9].

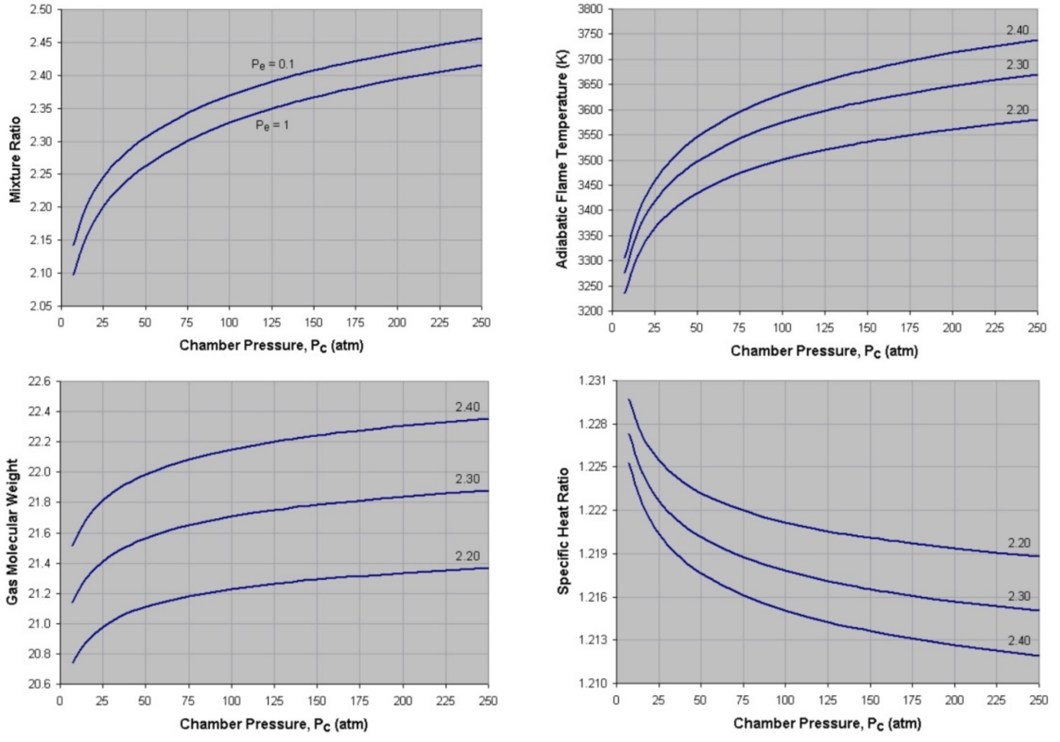

Figure 2 – LOX/kerosene combustion charts [6]

For the nonlinear surface generation, a two-variable power function has been chosen having the following form:

$$f(x, y) = a + b \cdot x^c + d \cdot y^e \tag{5}$$

The aim of the data fitting analysis is to find the optimum values of the 5 coefficients (a, b, c, d, e) such that the surface $z = f(x, y)$ accurately represents the data in the propellant combustion charts.

The nonlinear regression is attempted with the Trust-region algorithm detailed in [10], [11], [12] and the Levenberg-Marquardt algorithm detailed in [13], [14] which minimize the summed square of residuals. The main disadvantage of least-squares fitting is its sensitivity to outliers. The outliers are the extreme values that can appear in the input data and usually tend to be at the range frontiers. Two robust methods for outlier influence minimisation are also used, these being the LAR and Bisquare methods, with the aid of the Matlab surface fitting tool [15].

The following regression metrics were computed to quantify the accuracy of data approximation:

Table 2 – Regression metrics

Regression metric	**Formula**
Mean Squared Error (MSE)	$MSE = \frac{1}{N}\sum_{i=1}^{N}(y_i - \hat{y}_i)^2$
Root Mean Squared Error (RMSE)	$RMSE = \sqrt{\frac{1}{N}\sum_{i=1}^{N}(y_i - \hat{y}_i)^2} = \sqrt{MSE}$
Mean Absolute Error (MAE)	$MAE = \frac{1}{N}\sum_{i=1}^{N}\lvert y_i - \hat{y}_i\rvert$
Mean Absolute Percentage Error [%] (MAPE)	$MAPE = \frac{1}{N}\sum_{i=1}^{N}\frac{\lvert y_i - \hat{y}_i\rvert}{y_i}\cdot 100$
Max Absolute Percentage Error [%] (MaxAPE)	$MaxAPE = \max\left(\frac{\lvert y_i - \hat{y}_i\rvert}{y_i}\cdot 100\right)$

In Table 2: N is the number of data input points (ranging from 50 to 75 for all the combustion graphs); y_i is the actual expected output and $\hat{y}_i$ is the model's prediction.

From all nonlinear regression methods earlier presented, the one which has yielded the lowest Mean Absolute Percentage Error has been chosen as the optimum solution. For the LOX/kerosene propellant combination, the optimum values of the 5 desired coefficients are shown in Table 3.

The validity range of the model is: $P_c \in [10 - 250]$atm and $P_e \in [0.1 - 1]$atm, which corresponds to $R_m \in [2.2 - 2.4]$. A couple of LOX/kerosene combustion surfaces together with the residuals are shown in Figure 3, while the corresponding regression metrics are presented in Table 4.

Table 3 – LOX/kerosene approximation function coefficients

Propulsive parameter	**Function variable**		**Coefficient**				
	x	y	a	b	c	d	e
Mixture ratio	P_c	P_e	0.40488	1.80306	0.04244	-0.27005	0.07216
Flame temperature	P_c	R_m	-96657.5664	100008.738	0.00111	-20471.89	-5.10454

Gas molecular weight	P_c	R_m	-61.87059	35.50626	0.00568	39.03287	0.22586
Specific heat ratio	P_c	R_m	2.83175	-1.6644	0.00204	0.16136	-1.06601

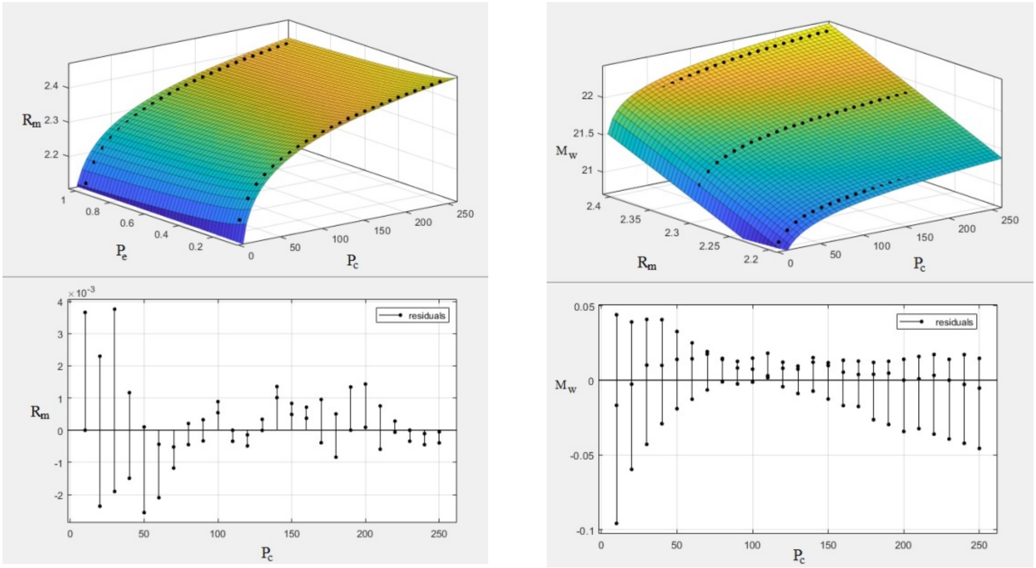

Figure 3 – LOX/kerosene mixture ratio and gas molecular weight approximations

Table 4 – LOX/kerosene regression metrics

Propulsive parameter	**Approximation errors**				
	MSE	RSME	MAE	MAPE [%]	MaxAPE [%]
Mixture ratio	1.443E-06	1.201E-03	8.185E-04	0.036	0.169
Flame temperature	9.994E+01	9.997E+00	7.185E+00	0.205	1.099
Gas molecular weight	5.583E-04	2.363E-02	1.739E-02	0.081	0.444
Specific heat ratio	9.962E-08	3.156E-04	2.373E-04	0.019	0.107

It can be clearly seen that the proposed two-variable power function accurately predicts the propulsive data of interest for the LOX/kerosene liquid propellant combination. The residuals are low for the majority of the models validity range, being slightly above average towards the end of the chamber pressure limits. Overall, the proposed power function yields very good results, as seen in Table 4.

Similar to the LOX/kerosene, the other 3 propellant combinations have been analysed and the results are presented in Table 5, Table 6 and Table 7. The validity range of the model is still: $P_c \in [10-250]$atm and $P_e \in [0.1-1]$atm, which corresponds to $R_m \in [2.7-2.9]$ for LOX/methane, $R_m \in [5-6]$ for LOX/hydrogen and $R_m \in [1.25-1.33]$ for LOX/ethanol.

Table 5 – LOX/methane approximation function coefficients

Propulsive parameter	**Function variable**		**Coefficient**				
	x	y	a	b	c	d	e
Mixture ratio	P_c	P_e	-0.88149	3.17521	0.03058	0.03777	-0.34131
Flame temperature	P_c	R_m	-283663.22149	200294.10397	0.00045	84639.97654	0.01976

Gas molecular weight	P_c	R_m	-332.49854	372.70801	0.00045	-36.22978	-0.51958
Specific heat ratio	P_c	R_m	2.1738	-2.66329	0.00105	1.78059	-0.0395

Table 6 – LOX/hydrogen approximation function coefficients

Propulsive parameter	**Function variable**		**Coefficient**				
	x	y	a	b	c	d	e
Mixture ratio	P_c	P_e	6.13493	-91.65097	-1.26518	0	0
Flame temperature	P_c	R_m	-30566.91615	12651.74602	0.00705	18970.35078	0.05811
Gas molecular weight	P_c	R_m	-11.04871	-1.59875	-0.11377	13.13317	0.37008
Specific heat ratio	P_c	R_m	6.18754	-4.99755	0.00058	0.52031	-1.78262

Table 7 – LOX/ethanol approximation function coefficients

Propulsive parameter	**Function variable**		**Coefficient**				
	x	y	a	b	c	d	e
Mixture ratio	P_c	P_e	-45.27899	46.40622	0.00084	0.00198	-0.81443
Flame temperature	P_c	R_m	-33290.54725	26893.55167	0.00297	8940.82886	0.12003
Gas molecular weight	P_c	R_m	-101.38495	66.57454	0.00253	55.2925	0.15025
Specific heat ratio	P_c	R_m	3.6356	-5.26427	0.00047	2.83722	-0.0138

For a better overall understanding of the regression metrics of the proposed functions the global MAPE and MaxAPE of all 4 propulsive parameters are shown in Table 8. It can be seen that the global mean absolute percentage errors (of all 4 propulsive parameters) are very small, the maximum being of almost 0.5%, while the global maximum absolute percentage error occurs for the LOX/hydrogen combination and is around 4.4%.

Table 8 – Global average errors – proposed two-variable power function

Propellant combination	**Global average errors (of all 4 propulsive parameters)**	
	MAPE [%]	MaxAPE [%]
LOX/kerosene	0.085	0.455
LOX/methane	0.074	0.607
LOX/hydrogen	0.451	4.391
LOX/ethanol	0.072	0.431

4. RESULTS

A total of 11 liquid propelled rocket engines have been analysed, ranging from 30kN to 7.7MN. All four bipropellant liquid combinations earlier presented have been analysed to validate the proposed mathematical model.

The two-variable power function with the earlier obtained coefficients has been implemented for the propulsive parameters of interest computation. Specific impulse and thrust values for sea and vacuum conditions obtained from the tool developed have been compared with the reference values and the errors computed. A list of all studied configurations is presented in Table 9.

Table 9 – Engine test configurations

Test configuration	Engine	Fuel	Chamber pressure [atm]	Exhaust pressure [atm]	Mass flow [kg/s]	Expansion ratio	References
1	Rocketdyne F-1	Kerosene	65.66	0.441	2577.8	16	[16], [17]
2	Merlin 1C-F9	Kerosene	65.26	0.173	161.5	14.5	[18], [19], [20]
3	Merlin 1D	Kerosene	95.92	0.421	236.6	21.4	[20]
4	Rocketdyne J-2	Hydrogen	51.92	0.473	240.7	27.5	[21]
5	Rocketdyne RS-25	Hydrogen	203.72	0.218	472.9	69	[22], [23], [24]
6a	JAXA 30 kN class	Methane	11.84	2.017	-	1.6	[25], [26]
6b	JAXA 30 kN class	Methane	9.87	0.015	9.1	49	[25], [26]
7	JAXA 100kN class	Methane	51.32	0.020	28.2	150	[25], [26]
8	DLR SE-12	Methane	252.95	0.591	1215.4	36.4	[27]
9	DLR L75	Ethanol	57.74	0.025	24.3	147	[28]
10	Glushko RD-101	Ethanol	21.31	1.174	172.9	3.42	[29], [30], [31]
11	Glushko RD-103	Ethanol	24.08	1.029	196.4	4.1	[29], [30], [31], [32]

Some of the input data, such as chamber pressure and expansion ratio are obtained from the presented references, and in most cases the other data (exhaust pressure and mass flow) are mathematically derived based on reference data.

Table 10 – Proposed model results vs. reference values

Engine	Reference values				Obtained values			
	I_{sp} [s] s.l.	I_{sp} [s] vac.	T [kN] s.l.	T [kN] vac.	I_{sp} [s] s.l.	I_{sp} [s] vac.	T [kN] s.l.	T [kN] vac.
Rocketdyne F-1	263	304	6770	7770	263.66	305.40	6665.14	7720.35
Merlin 1C-F9	263	302	409.24	469.29	265.56	303.63	420.59	480.88
Merlin 1D	282	320	654.33	742.41	273.49	311.87	634.57	723.62
Rocketdyne J-2	-	424	-	1023.09	-	420.74	-	993.23
Rocketdyne RS-25	366	452.3	1705.83	2090.66	362.25	439.25	1679.94	2037.04
JAXA 30 kN class	234	335	-	30	225.59	332.73	-	29.79
JAXA 100kN class	-	356	-	98	-	353.34	-	97.78
DLR SE-12	322.5	348.3	3844	4152	315.85	342.11	3764.72	4077.76
DLR L75	-	315	-	75	-	313.75	-	74.64
Glushko RD-101	214	240	363	402	221.04	246.02	374.79	417.14
Glushko RD-103	220	251	432	500	224.95	251.49	433.27	484.37

The specific impulse and thrust values for sea level (s.l.) and vacuum (vac.) conditions are obtained for the input data from Table 9. The comparison between the reference values and the ones obtained with the proposed model are shown in Table 10.

Table 11 and Figure 4 present the errors between the results obtained with the Matlab tool developed based on the proposed mathematical model and the reference data. It can be seen that the model predicts real-life engine data with a high order of accuracy, the maximum absolute percentage error being of 3.2%, while the global mean absolute percentage error is at 1.65%.

Table 11 – Proposed model absolute errors

Engine	**Absolute error [%]**				**MAPE [%]**
	I_{sp} **s.l.**	I_{sp} **vac.**	T **s.l.**	T **vac.**	
Rocketdyne F-1	0.25	0.46	1.55	0.64	*0.73*
Merlin 1C-F9	0.97	0.54	2.77	2.47	*1.69*
Merlin 1D	3.02	2.54	3.02	2.53	*2.78*
Rocketdyne J-2	-	0.77	-	2.92	*1.84*
Rocketdyne RS-25	1.03	2.89	1.52	2.56	*2.00*
JAXA 30 kN class	3.59	0.68	-	0.69	*1.65*
JAXA 100kN class	-	0.75	-	0.22	*0.48*
DLR SE-12	2.06	1.78	2.06	1.79	*1.92*
DLR L75	-	0.40	-	0.47	*0.43*
Glushko RD-101	3.29	2.51	3.25	3.77	*3.20*
Glushko RD-103	2.25	0.19	0.29	3.13	*1.47*
Global model prediction error [%]					**1.65**

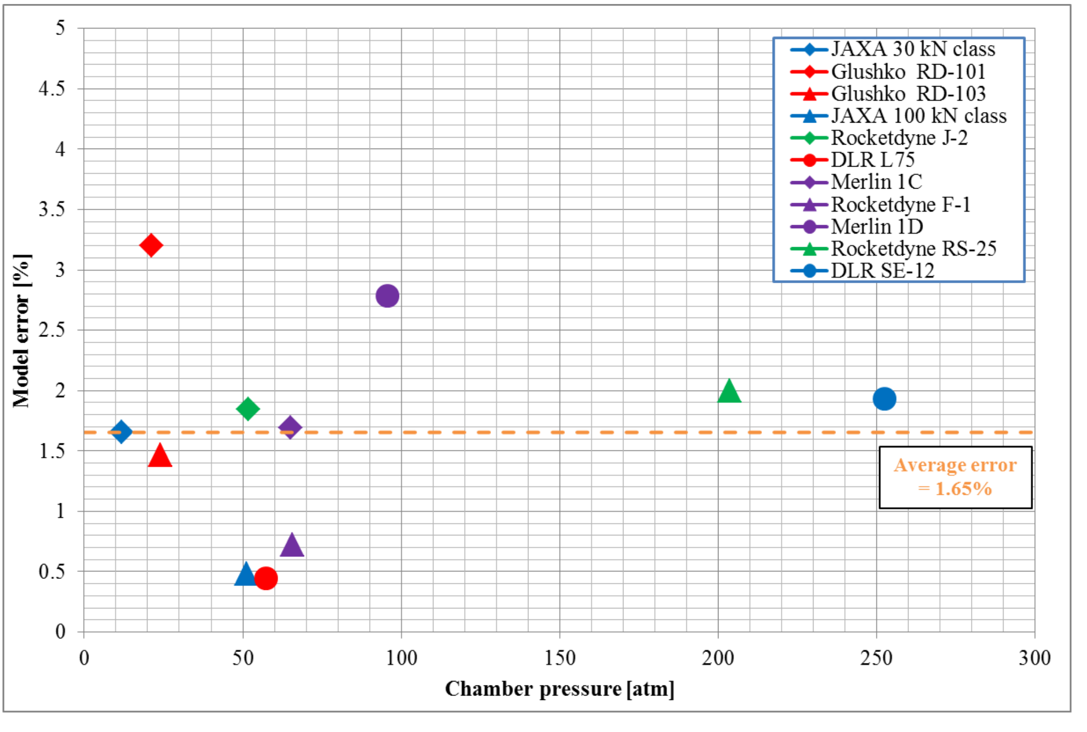

Figure 4 – Proposed model accuracy vs. combustion chamber pressure

5. CONCLUSIONS

The proposed model can be used to quickly assess the propulsive performance for most small launcher liquid engines. The main propulsive parameters are computed using combustion surfaces obtained after a nonlinear data fitting analysis. The optimum values of the 5 coefficients used in the nonlinear two-variable power function are given in Table 3, Table 5, Table 6 and Table 7.

In the preliminary phases of a launcher design, when the propulsive assessment of a high number of configurations must be realised, models that provide fast approximations are preferred. In this paper, the specific impulse of the engine is modelled based on the combustion chamber and the exhaust pressures, together with the atmospheric conditions at the desired flight regime (sea level or vacuum). For thrust computations, the propellant mass flow is also needed as input data.

The results provide a very good approximation, as seen in the comparison with the engines reference values, the proposed model global error being under 2%. The tool developed based on this mathematical model can be used separately or it can be integrated in a more complex, multidisciplinary optimisation. Because of the low computational time needed, especially due to the nonlinear approximation functions used, the proposed mathematical model is suitable to be used in a full loop MDO algorithm.

REFERENCES

[1] 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.

[2] 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.

[3] M. Balesdent, *Multidisciplinary Design Optimization of Launch Vehicles, Optimization and Control*, [math.OC], Ecole Centrale de Nantes (ECN) (ECN) (ECN), 2011, English, HAL Id: tel-00659362, https://tel.archives-ouvertes.fr/tel-00659362, submitted on 12 Jan 2012.

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

[5] G. Sutton, O. Biblarz, *Rocket Propulsion Elements*, 8th Edition, John Wiley & Sons, New Jersey, 2010.

[6] * * * R. A. Braeunig, Rocket and Space Technology, *Propellant Combustion Charts*, Available at http://www.braeunig.us/space/comb.htm.

[7] W. C. Reynolds, *STANJAN: Interactive computer programs for chemical equilibrium analysis*, Stanford University, Thermosciences Division, 1981.

[8] * * * National Institute of Standards and Technology (NIST), *NIST-JANAF Thermochemical Tables – SRD 13*, NIST Standard Reference Database 13, 1998, DOI: 10.18434/T42S31, Available at https://janaf.nist.gov/.

[9] * * * A. Rohatgi, *Web based tool to extract data from plots, images and maps*, WebPlotDigitizer, Version: 4.2, San Francisco, USA, Available at https://automeris.io/WebPlotDigitizer.

[10] J. J. Moré, D. C. Sorensen, Computing a Trust Region Step, *SIAM Journal on Scientific and Statistical Computing*, vol. **3**, pp. 553–572, 1983.

[11] M. A. Branch, T. F. Coleman, Y. Li, A Subspace, Interior, and Conjugate Gradient Method for Large-Scale Bound-Constrained Minimization Problems, *SIAM Journal on Scientific Computing*, vol. **21**, Number 1, pp. 1–23, 1999.

[12] R. H. Byrd, J. C. Gilbert, J. Nocedal, A Trust Region Method Based on Interior Point Techniques for Nonlinear Programming, *Mathematical Programming*, vol. **89**, No. 1, pp. 149–185, 2000

[13] K. Levenberg, A Method for the Solution of Certain Problems in Least Squares, *Quart. Appl. Math*. vol. **2**, pp. 164–168, 1944.

[14] D. Marquardt, An Algorithm for Least-Squares Estimation of Nonlinear Parameters, *SIAM J. Appl. Math*. vol. **11**, pp. 431–441, 1963.

[15] * * * Mathworks Matlab Documentation, *Curve and Surface Fitting Tool*, Available at https://www.mathworks.com/help/curvefit/curvefitting-app.html

[16] * * * NASA Marshall Space Flight Center, *F-1 Engine (chart)*, MSFC-980177.
[17] * * * National Aeronautics and Space Administration, *Saturn V News Reference: F-1 Engine Fact Sheet*, 1968, Available: https://history.msfc.nasa.gov/saturn_apollo/documents/F-1_Engine.pdf.
[18] B. Bjelde, P. Capozzoli, G. Shotwell, The SpaceX Falcon 1 Launch Vehicle Flight 3 Results, Future Developments, and Falcon 9 Evolution, *59th International Astronautical Congress, Paper nr. IAC-08.D2.1.3*.
[19] * * * Wikipedia, *Merlin engine*, Available: https://en.wikipedia.org/wiki/Merlin_(rocket_engine_family).
[20] N. Brügge, *Evolution of the SpaceX Merlin-1 engine (and parameter)*, Available at http://www.b14643.de/Spacerockets_2/United_States_1/Falcon-9/Merlin/index.htm.
[21] * * * National Aeronautics and Space Administration, *Saturn V News Reference: J-2 Engine Fact Sheet*, 1968, Available at https://www.nasa.gov/centers/marshall/pdf/499245main_J2_Engine_fs.pdf.
[22] * * * Pratt & Whitney Rocketdyne, *Space Shuttle Main Engine Performance*, Available at https://web.archive.org/web/20120208191620/http:/www.pw.utc.com/products/pwr/assets/pwr_SSME .pdf
[23] M. Wade, Encyclopedia Astronautica , *SSME*. Available at http://www.astronautix.com/s/ssme.html
[24] N. Brügge, *Space Shuttle Description*, Available at http://www.b14643.de/Spacerockets_2/ United_States_1/Space_Shuttle/Description/Frame.htm
[25] H. Asakawa, *The Status of the Research and Development of LNG Rocket Engines in Japan*, Chemical Rocket Propulsion: A Comprehensive Survey of Energetic Materials, Springer 2017.
[26] T. Kato et al, Subscale Firing Test for Regenerative Cooling LOX/Methane Rocket Engine, *7th European Conference For Aeronautics And Space Sciences (EUCASS)*, DOI: 10.13009/EUCASS2017-381.
[27] H. Burkhardt, M. Sippel, A. Herbertz, J. Klevanski, Comparative Study of Kerosene and Methane Propellant Engines for Reusable Liquid Booster Stages, *4th International Conference on Launcher Technology Space Launcher Liquid Propulsion*, Belgium, December 2002.
[28] L. Pfuetzenreuter et al, L75 LOx Ethanol Engine: Current Status of Thrust Chamber and Turbopump Cooperative Development, *AIAA Propulsion and Energy Forum 53rd AIAA/SAE/ASEE Joint Propulsion Conference*, Atlanta, GA, 10-12 July 2017.
[29] C. Lardier, S. Barensky, *The Soyuz Launch Vehicle: The Two Lives of an Engineering Triumph*, Springer, 2013.
[30] O. H. Przybilsk, The Germans and the Development of Rocket Engines in the USSR, *JBIS*, vol. **55**, pp. 404-427, 2002.
[31] N. Brügge, *Russian space-rocket and missile liquid-propellant engines*, Available at http://www.b14643.de/Spacerockets/Specials/Russian_Rocket_engines/engines.htm.
[32] * * * Liquid Propellant Rocket Engines, *Background to the creation of LRE RD-107/108*, Available at http://lpre.de/energomash/ED/index.htm