Small Solid Propellant Launch Vehicle Mixed Design Optimization Approach

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ABSTRACT: For a small country with limited research budget and lack of advanced space technology, it is imperative to find new approaches for the development of low-cost launch vehicles (LV), which is, among all possibilities, an interesting option for rapid access to space, focused on integration of acquired components complemented with indigenously developed subsystems. This approach requires the cooperation of developed countries with huge experience and knowledge in LV development and operations. The main objective is to develop a small three stage solid propellant LV capable of delivering a payload of 100 kg to a circular low earth orbit of 600 km altitude, with the first and second stage solid rocket motors (SRM) hypothetically acquired from different countries and the third one designed and produced domestically in accordance with the production and technological capability. This approach provides main advantages such as: reduction in total time to access the space and to master the basic knowledge of launch operations. For this purpose, an integer continuous genetic algorithm global optimization method was selected and implemented, the SRM characteristics of the first and second stage were considered as integer variables, whereas the design variables of the third stage SRM and the trajectory variable were considered as continuous. A multi discipline feasible (MDF) framework was implemented along with the propulsion, aerodynamic, mass and trajectory models. Despite their particular characteristics and constraints, the results show highly acceptable values, and the approach proved to be reliable for conceptual design level.

KEYWORDS: Launch vehicle, Mixed design optimization, Solid propellant.

INTRODUCTION

The last decade may be characterized by an increased number of small satellites delivered into the low earth orbit (LEO), and this tendency will be dominant in the coming years.

Small satellites have a reduced manufacturing cost, and are relatively easy to operate and maintain. Furthermore, the miniaturization of technology makes possible its delivery into space by using small cost effective launch vehicles (LV).

Small countries generally have a limited research budget oriented to space technology development, however, nowadays it is possible to deliver a small satellite into orbit with a reasonable budget, considering the cooperation with technologically more advanced countries.

This research was focused on finding a way to have rapid access into space and to master the basic knowledge of space development and operations. In such a way, several options had been analyzed, among them the most suitable alternative in terms of economic investment and development time resulted in a small solid propellant LV with mixed design configuration, involving a strong cooperation with different countries.

The strategy considered here prioritizes the technology integration over expensive and time consuming new development, this means that complex and advanced devices were acquired and complemented with indigenous manufactured devices using available resources and technology.

As a result, a three stage solid propellant LV was configured, where the first and second stage solid rocket motors (SRM) were acquired from different providers, complemented with a locally developed third stage SRM, which was designed and optimized to accomplish the specific mission.

In our research, a mixed integer continuous variables genetic algorithm (GA) method has been used in order to optimize the overall configuration of the LV.

LAUNCH VEHICLE MODEL

LAUNCH VEHICLE DEFINITION

A small three stage solid propellant LV in tandem configuration is considered for this research. The mission is to deliver a 100 kg payload to a circular LEO of 600 km of altitude. The payload volume requirements and the instrument module weight were specified beforehand in mission definition analysis and are shown in Table 1.

Table 1. Launch Vehicle data.

Variables	Units	Value
Payload	kg	100
Fairing mass	kg	50
Instrument module	kg	50
Payload deployment module	kg	50

CONSIDERED SOLID ROCKET MOTORS

The considered SRM are listed in the Table 2 and were selected based on the variety of design characteristics. However, it is possible to add additional parameters such as cost, availability, technology complexity, country of origin among others.

PROPULSION ANALYSIS

The propulsion analysis has been conducted for all three stages of the LV, using the classical approach presented in Sutton and Biblarz (2001) and He (2004a; 2004b). For the third stage SRM, a detailed analysis was conducted, considering the properties of the propellant. In this analysis, the burning surface is considered constant by introducing a grain geometry shape coefficient, k_s, the burning surface of the grain S_b can be calculated as:

$$S_b = k_s \, D_m \, L_m \tag{1}$$

where, L_m is the rocket motor cylindrical length and D_m the diameter.

The burning time t_b, grain mass m_{gn}, and mass flow rate $\dot{m}_{gn}$ of the grain are calculated as:

$$m_{gn} = \frac{\pi}{4} \rho_{gn} \eta_v \lambda_{gn} D_m^3 \tag{2}$$

Table 2. Selected solid rocket motors.

	SRM	Grain mass (kg)	SRM mass (kg)	Diameter (m)	Length (m)	Specific impulse (N.s/kg)	Burn time (s)	Mass flow (kg/s)	Thrust (N)
Stage1	11	18400	20791	1990	4.80	2364	65	283.08	669194
	12	15000	16779	1390	7.25	2314	74	202.70	469054
	13	9950	11281	1390	5.20	2280	62	160.48	365903
	14	4530	5207	0.98	4.60	2265	70	64.71	146578
Stage2	21	9800	10950	1990	3.62	2805	65	150.77	422908
	22	5080	5607	1390	3.10	2776	64	79.38	220345
	23	4138	4412	1390	2.86	2746	65	63.66	174815
	24	3700	4190	1390	2.40	2754	68	54.41	149850
	25	3300	3650	1390	2.63	2824	55	60.00	169440
	26	1760	1949	0.98	1.85	2776	46	38.26	106212
	27	650	719	0.85	1.50	2849	43	15.12	43066

$$t_b = \frac{\pi \, \eta_v \, D_m}{4 \, u \, k_s} \tag{3}$$

$$\dot{m}_{gn} = \rho_{gn} u \, S_b = \rho_{gn} u \, k_s \lambda_{gn} D_m^2 \tag{4}$$

$$\lambda_{gn} = \frac{L_{gn}}{D_{gn}} \tag{5}$$

where, u is the burning rate of propellant, ρ_{gn} density of the grain, $L_{gn} = L_m + 0.314 D_m$ length of the grain, $D_{gn} = D_m$ diameter of the grain, λ_{gn} fineness ratio of the grain (grain length/diameter), and η_v the grain volumetric loading fraction.

The expansion ratio ε, nozzle throat area A_t, and nozzle exit area A_e are calculated as:

$$A_t = \frac{\rho_{gn} u \, S_b}{\Gamma_0 \, p_{c\max}} \sqrt{R_c \, T_c} \tag{6}$$

$$\varepsilon = \frac{\Gamma_0}{\left(\frac{p_e}{p_c}\right)^{\frac{1}{k}} \sqrt{\frac{2k}{k-1}\left[1 - \left(\frac{p_e}{p_c}\right)^{\frac{k-1}{k}}\right]}} \tag{7}$$

$$A_e = A_t \varepsilon \tag{8}$$

$$\Gamma_0 = \sqrt{k}\left(\frac{2}{k+1}\right)^{\frac{k+1}{2(k-1)}} \tag{9}$$

where, P_c is the chamber pressure, p_e exit pressure, $R_c = 296 \, J/(kg.K)$ gas constant, $T_c = 3300 \, K$ temperature in the combustion chamber, $P_{c\max} = 1.1 P_c$ maximum value of chamber pressure, and $k = 1.2$ the specific heat ratio of gas.

The specific impulse I_{sp}, and the thrust T can be calculated as:

$$I_{sp} = I_{sp}^a + \left(\frac{p_e}{p_c}\right)^{\frac{k-1}{k}} \frac{R_c \, T_c}{g^2 \, I_{sp}^a} \tag{10}$$

$$T = I_{sp} \dot{m}_{gn} - p_a A_e \tag{11}$$

where, p_a is the atmospheric pressure, I_{sp}^a average specific impulse, g acceleration due to gravity, and A_e the nozzle exit area.

MASS ANALYSIS

The mass analysis was conducted for the entire LV, and is represented by the following equations:

$$m_{LV} = m_{01} + m_{02} + m_{03} + m_{Inert} \tag{12}$$

$$m_{03} = m_{st} + m_{gn} \tag{13}$$

$$m_{Inert} = m_{IM} + m_{PDM} + m_{PAY} \tag{14}$$

where, m_{LV} is the LV gross mass, m_{01} first stage mass, m_{02} second stage mass, m_{03} third stage solid rocket mass, m_{IM} instrument module mass, m_{PDM} payload deployment module mass, m_{PAY} payload mass, and m_{st} the structural mass of the third stage SRM.

He (2004a; 2004b) provided a methodology and a detailed calculation of the third stage SRM structural mass. This design consisted in a classical metallic case made of high strength steel, ethylene propylene diene monomer (EPDM) for chamber insulation, and carbon phenolic for the nozzle.

AERODYNAMIC ANALYSIS

The aerodynamic coefficients were estimated using the Missile DATCOM 1997 digital (Blake, 1998). This software is easy to use and implemented, and accurate enough for the conceptual design phase. Qazi and He (2005) and Villanueva *et al.* (2013) applied DATCOM in LV aerodynamics analysis. The lift and drag forces were calculated using the following relations:

$$L = C_L q S_{ref} \tag{15}$$

$$D = C_D q S_{ref} \tag{16}$$

where, q is the dynamic pressure, D drag force, L lift force, S_{ref} vehicle reference area, C_L lift coefficient, and C_D the drag coefficient.

The aerodynamic coefficients were calculated repeatedly for each LV configuration, the selected Mach ranged from 0 to 8 and the angle of attack from −8 to +1 degrees.

TRAJECTORY ANALYSIS

The trajectory analysis considers a 3 degree of freedom (3DOF) model, which has been modeled in SIMULINK (Zipfel, 2007; Fleeman, 2001). The previously calculated aerodynamic coefficients, the mass and the propulsion are the input parameters. In order to obtain a quick result, a 2D coordinate system was adopted, the LV flies as a point mass in a non rotating earth model. Figure 1 illustrates the forces acting on a LV and below a set of governing equations of motion (Xiao, 2001). The LV is flying in an inertial reference coordinate system XOY, with
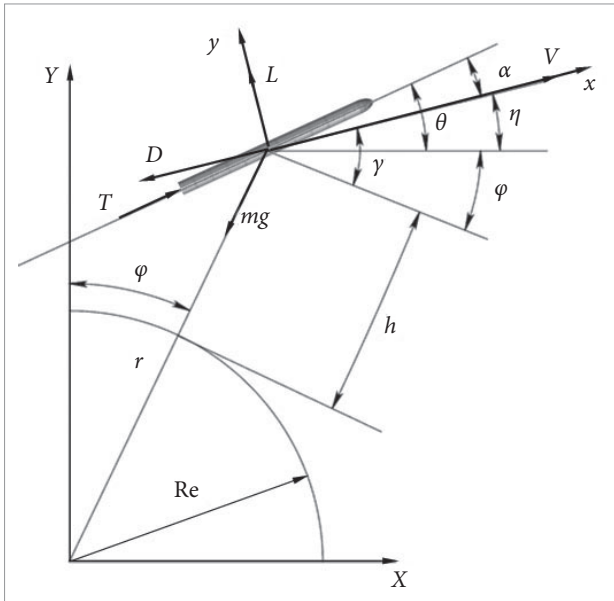
Figure 1. Forces acting on a launch vehicle.

its origin located in the center of the earth. Furthermore, all forces applied to the LV were considered in relation to the body centered velocity coordinate systems xoy as shown in Fig. 1.

$$\frac{dV}{dt} = \frac{T\cos\alpha - D}{m} - g\sin\gamma$$

$$\frac{d\gamma}{dt} = \frac{T\sin\alpha + L}{mV} - \frac{g\cos\gamma}{V} + \frac{V\cos\gamma}{r}$$

$$\frac{dh}{dt} = V\sin\gamma$$

$$\frac{d\varphi}{dt} = \frac{V\cos\gamma}{r} \qquad (17)$$

$$\alpha = \theta + \varphi - \gamma$$

$$\frac{dm}{dt} = -\frac{T}{I_{sp}}$$

$$\alpha = \alpha_{prog}(t)$$

where, V is the velocity, m vehicle mass, θ pitch angle, η trajectory angle, γ flight path angle, φ range angle, h height above ground, α angle of attack, and $\alpha_{prog}(t)$ is the programmed angle of attack.

The axial and normal overload coefficients ensure the integrity of the LV in all phases of flight, and were calculated in a body centered velocity coordinate systems (xoy), as follows:

$$n_x = \frac{T + L\sin\alpha - D\cos\alpha}{mg} \leq n_{x\max} \qquad (18)$$

$$n_y = \frac{T\sin\alpha + L}{mg} \leq n_{y\max} \qquad (19)$$

The thrust to weight ratio gives an important value to evaluate the liftoff characteristics of the LV:

$$\nu = \frac{T}{m\,g} \qquad (20)$$

The density variation with altitude can be calculated as:

$$\rho = \rho_0\, e^{(-h/\beta)} \qquad (21)$$

The gravity varies with altitude and can be represented as:

$$g = \frac{\mu}{(R_e + h)^2} \qquad (22)$$

where, ρ_0 is the sea level density, R_e radius of earth, β density scale height, and μ the earth gravitational parameter.

The mission requires to deliver the payload to an altitude h_f with a circular orbital insertion velocity V_f:

$$V_f = \sqrt{\frac{\mu}{h_f + R_e}} \qquad (23)$$

TRAJECTORY PHASES

The trajectory of the LV can be described as a composition of several phases, as presented by He (2004a), Qazi and He (2005) and Villanueva *et al.* (2013). For the present research, the trajectory was sectioned in seven phases, as shown in Fig. 2. Each phase has a specific flight characteristic as described next:

- Vertical launch phase: This phase starts from the time of ignition of the first stage SRM until the end of vertical flight time t_v ($t_v = t_1$ in Fig. 3), during this time the LV flies vertically with a flight path angle equal to 90 degrees.
- Pitch over phase: During this phase, the LV maneuver with a negative angle of attack until the transonic velocity is reached. In this point, the angle of attack should approaches zero degrees.
- Powered first stage phase: This phase lasts until the end of the burning time of the first stage SRM. The angle of attack should be kept at zero during the stage separation process.
- Coasting phase 1: The LV flies with no thrust until the second stage ignites.
- Powered second stage phase: The duration of this phase starts with the ignition of the second stage SRM and is equal to its burning time.

- Coasting phase 2: This phase is characterized by a prolonged ballistic free flight approaching the target altitude.
- Kick phase: This phase starts with the ignition of the third stage SRM until the insertion altitude at the required orbital velocity and flight path angle.

FLIGHT PROGRAM FORMULATION

The flight profile defines the performance and loads acting on the LV Consequently, its selection should be integrated in the optimization process. Figure 3 explains the variation of the angle of attack during the pitch over phase (He, 2004a; Xiao, 2001):

$$\alpha_{prog}(t) = -\alpha_{max} \sin^2 f(t) \tag{24}$$

$$f(t) = \frac{\pi(t - t_1)}{a_m(t_2 - t) + (t - t_1)} \tag{25}$$

$$a_m = \frac{t_a - t_1}{t_2 - t_a} \tag{26}$$

where, α_{max} is the maximum angle of attack, a_m launch maneuver variable, t_a time corresponding to maximum angle of attack, t time of flight, and t_1 the time of start of pitch over phase, coincident in value with time t_v.
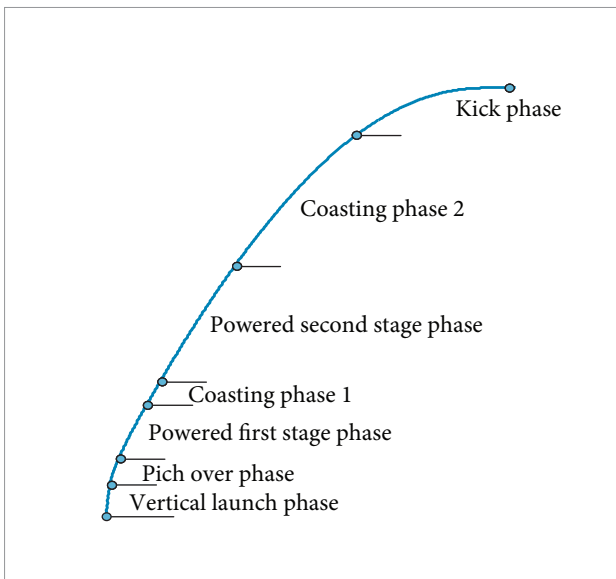

Figure 2. Trajectory phases of launch vehicle.

DESIGN OPTIMIZATION PROBLEM

OBJECTIVE FUNCTION

There can be different objective functions for LV optimization problem, such as minimization of the LV cost, which can be obtained knowing the cost of the first and second stage SRMs and the development cost of the third stage, and also the minimization of the development time, knowing the availability of the first and second stage SRM and the development time of the third stage SRM. However, this analysis considers the minimization of the gross launch mass (m_{LV}). The mathematical description of design objective is as follows:

$$\min m_{LV} = f(X) \tag{27}$$

$$g_j(X) \leq 0 \tag{28}$$

$$h_k(X) = 0 \tag{29}$$

$$X_{lb} \leq X_i \leq X_{ub} \tag{30}$$

where, g_j is the inequality constraints, h_k the equality constraints, X the set of variables, X_{lb} the lower bound of variables and X_{ub} the upper bound of variables.

DESIGN VARIABLES

The design variables are composed from integer (first and second stage SRMs), and continuous third stage SRM and trajectory variables. They are listed in Table 3 and can be represented as:

$$X = [X_{SRM}, X_{SRM3}, X_{Trajectory}] \tag{31}$$
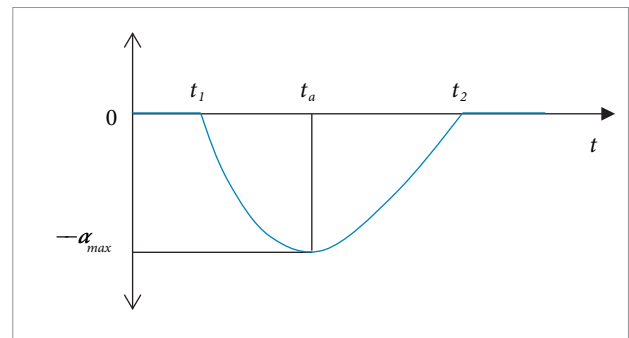
$$X_{SRM} = [srm_1, srm_2] \tag{32}$$

Figure 3. Pitch over ascent phase of launch vehicle.

$$X_{SRM3} = [L_{m3}, D_{m3}, p_{c3}, p_{e3}, k_{s3}, u_3, \rho_{gn3}] \qquad (33)$$

$$X_{Trajectory} = [t_v, t_m, t_{c1}, t_{c2}, \alpha_{max}, a_m] \qquad (34)$$

DESIGN CONSTRAINTS

The selections of constraints were oriented in order to satisfy the mission, to prevent any failure during flight, and to consider the limitation of the third stage manufacturing technology. They are listed in Table 4:

OPTIMIZATION STRATEGY

INTEGER CONTINUOUS OPTIMIZATION APPROACH

The particularity of our problem deals with integer and continuous variables simultaneously. The selection of SRM type for first and second stages were considered as integer variables.

Meanwhile, the trajectory and design parameters of the third stage SRM were considered as continuous.

Several engineering applications of mixed integer continuous optimization approach were presented by Haupt *et al.* (2009), Faustino *et al.* (2006), as well as detailed explanation in Yeniay (2005) and Gantovnik *et al.* (2005).

Garfield and Allen (1995) used integer optimization applied to the configuration of LVs, Johnson (2002) conducted a screening process of booster for hypersonic vehicles, Calabro *et al.* (2002) presented the optimization of the propulsion for multistage LVs, and Bhatnagar *et al.* (2012) solved the mass distribution problem under restrictive condition.

Hartfield *et al.* (2004) have shown the application of GA in finding the global optimum in ramjet propulsion. Bayley and Hartfield (2007) used GA for LV multidisciplinary design optimization with emphasis on minimum cost.

GA has been effectively applied to solve the problem of liquid propellant based LV (Riddle *et al.* 2007), as well as

Table 3. Design variables.

		Variables	Symbol	Units
X1		Stage 1	srm_1	
X2		Stage 2	srm_2	
X3		Rocket motor cylindrical length	L_{m3}	m
X4		Rocket motor diameter	D_{m3}	m
X5		Chamber pressure	P_{c3}	Pa
X6		Nozzle exit pressure	P_{e3}	Pa
X7		Coefficient of grain shape	k_{s3}	
X8		Grain burning rate	u_3	m/s
X9		Grain density	ρ_{gn3}	kg/m³
X10		Vertical flight time	t_v	s
X11		Time to pitch over	t_m	s
X12		Coasting time 1 (between 1st and 2nd stage)	t_{c1}	s
X13		Coasting time 2 (between 2nd and 3rd stage)	t_{c2}	s
X14		Maximum angle of attack (absolute)	α_{max}	deg
X15		Launch maneuver variable	a_m	

Table 4. Design constraints.

		Constraints	Value	Units
C1		Orbit insertion velocity	$V_f \geq 7560$	m/s
C2		Final altitude	$V_f \geq 600$	km
C3		Axial overload	$n_x \leq 14$	
C4		Normal overload	$n_y \leq 2$	
C5		Maximum dynamic pressure	$q_{max} \leq 85$	kPa
C6		Angle of attack $(0.8 \leq M \leq 1.3)$	$\alpha = 0$	deg
C7		Orbit insertion angle	$\gamma = 0 \pm 0.2$	deg
C8		Rocket motor diameter	$D_{m1} \geq D_{m2}$	m
C9		Rocket motor mass	$m_{SRM1} \geq m_{SRM2}$	kg
C10		Total LV length	$L_{LV} \leq 18$	m
C11		Grain fineness	$\lambda_{gn3} < 2$	
C12		Thrust to weight ratio	$v_3 \geq 1.8$	
C13		Nozzle exit diameter	$d_{e3} \leq 0.9 D_{m3}$	m
C14		Burning time	$t_{b3} \leq 65$	s

solid propellant LVs (Bayley *et al.* 2008). Rafique *et al.* (2009) and Goldberg (1989) provides detailed and comprehensive implementation of GA in solving complex problems.

OPTIMIZATION METHOD

The adopted and implemented GA optimization method is shown in Fig. 4, where a set of input design variables (SRM type, trajectory and third stage), as well as the lower and upper bounds, are passed to the main loop, where an initial population is randomly created. Furthermore, the selection, the crossover and the mutation operations are performed until the stopping criteria is achieved. The constraints were calculated and handled by external penalty function, as presented in Deb (2000) and detailed and explained in Coello (1999) and Kramer (2010). At each routine, the propulsion, mass, aerodynamics and trajectory analysis were performed.

The main characteristics of GA are presented in Table 5.

OPTIMIZATION FRAMEWORK

The optimization framework considered for this research is based on the multi-discipline feasible (MDF) design, which allows an easy and accurate result (Qazi and He L, 2006), as shown in Fig. 5.
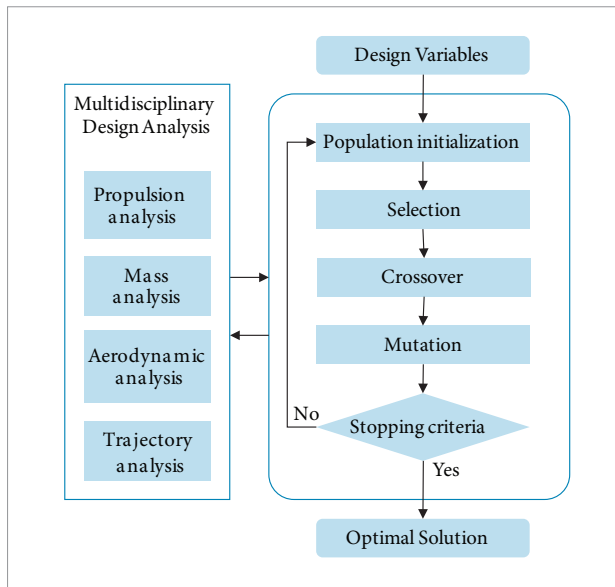

Figure 4. Genetic algorithm optimization approach.

Table 5. Genetic algorithm characteristics.

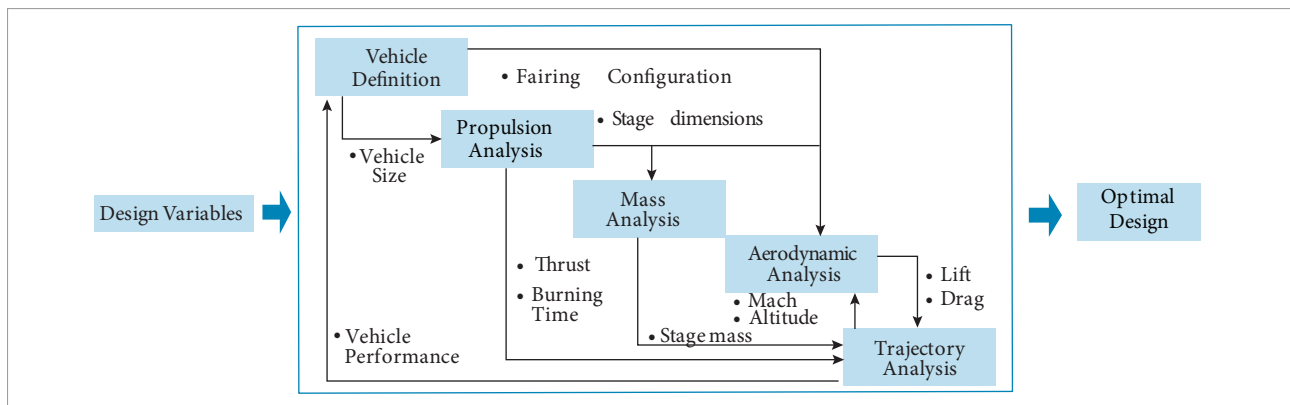| Variables	Characteristics
Generations	200
Population size	100
Stopping criteria	Function tolerance 10e-6
Population type	Double vector
Selection	Stochastic uniform
Crossover	Single point pc = 0.8
Mutation	Uniform pm = 0.2564
Reproduction	Elite count = 2
Function evaluation	2000

Figure 5. Multidisciplinary design optimization.

OPTIMIZATION RESULT

The results show that the considered mixed integer-continuous GA optimization approach successfully reached the objective function. The optimized LV has a total mass of 23,530 kg and a 16.12 m of length. Table 6 shows the optimized value of variables and in Table 7 the main parameters of the LV third stage are listed.

The first and second stages SRM design type (SRM 12 and SRM 22), had been optimized and selected from Table 2. Both SRMs have the same diameter but different length. As it is represented in Fig. 6, the shroud design is configured with the same diameter as the third stage, in order to reduce the aerodynamics forces and interferences.

The performance characteristics of the LV, shown in Fig. 7, demonstrates the capability of the three stage solid propellant LV to place a small payload into the LEO orbit maintaining its main parameters inside its limit values, furthermore, the overall design configuration facilitates its launch operations.

CONCLUSION

A small three-stage solid propellant LV was configured and optimized using a mixed integer-continuous GA optimization method. The first and second stages SRM types were considered as integer variables, whereas the third stage SRM and trajectory as continuous. The main advantage of using GA relies on its independency of initial values to start the optimization, and the ability to handle integer variables. The propulsion, mass, aerodynamic and dynamic models were developed and integrated in a MDF framework.

Table 6. Optimum values of variables.

Variables		Lower Bound	Upper Bound	Optimized Value
X1	srm_1	11	14	12
X2	srm_2	21	27	22
X3	L_{m3} (m)	0.80	1.20	0.90
X4	D_{m3} (m)	0.80	1.20	0.83
X5	P_{c3} (Pa)	70e5	80e5	77.42e5
X6	P_{e3} (Pa)	0.05e5	0.15e5	0.133e5
X7	k_{s3}	1.10	1.60	1.14
X8	u_3 (m/s)	6.0e-3	8.0e-3	6.71e-3
X9	ρ_{gn3} (kg/m³)	1650	1740	1683.1
X10	t_v (s)	3.0	6.0	3.01
X11	t_m (s)	18.0	25.0	21.57
X12	t_{c1} (s)	2.0	8.0	4.46
X13	t_{c2} (s)	360	400	372.61
X14	α_{max} (deg)	3.0	6.0	5.731
X15	a_m	0.28	0.42	0.319

Table 7. Parameters of launch vehicle third stage.

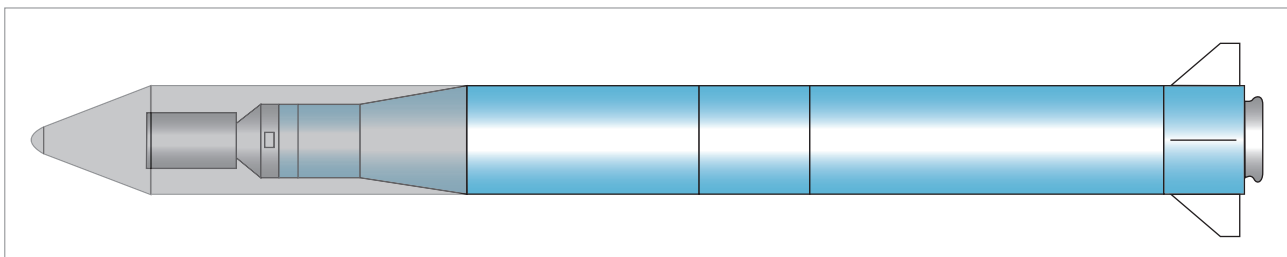| Parameters	Value (Units)
Stage mass	893.6 (kg)
Propellant mass	762.1 (kg)
Stage dry mass	131.2 (kg)
Propellant mass fraction	0.854
Average Thrust	33.82 (kN)
Specific Impulse vac	2702.8 (N.s/kg)
Nozzle expansion ratio	48.01
Thrust to weight ratio	3.69
Burning time	60.9 (s)

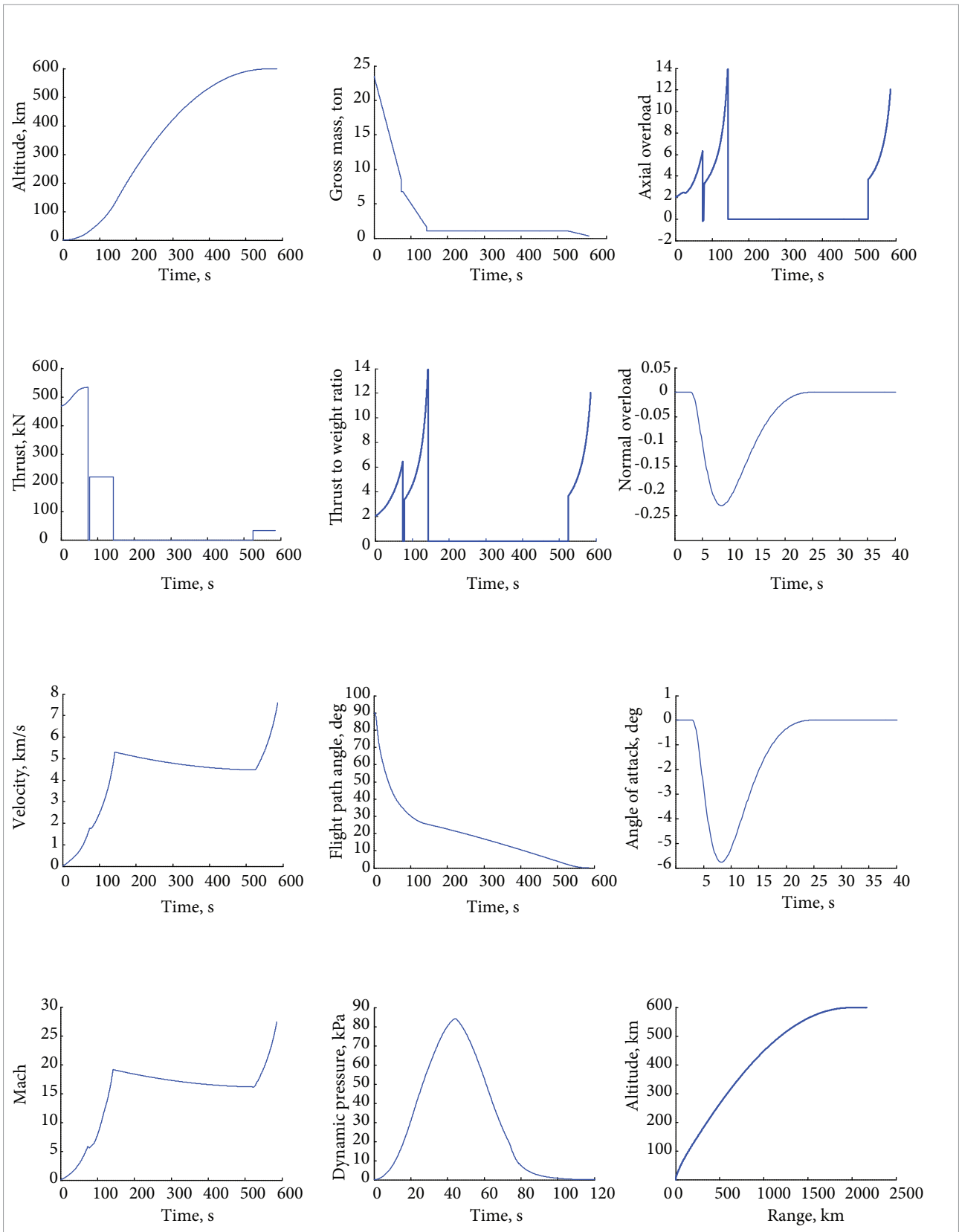

Figure 6. Optimized three stage launch vehicle.

Figure 7. Performance characteristics of launch vehicle.

An important contribution of this research is the approach in finding the best LV design configuration to rapid access to space with limited research budget, relied mainly on international cooperation and complemented with the indigenous aerospace manufacturing technology capability.

ACKNOWLEDGMENT

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