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An Optimization Approach for Conceptual and Preliminary Design of Scramjet Engines

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Abstract

Hypersonic air-breathing propulsion and supersonic combustion is a research field that has been explored either for military and civilian applications.Developing technologies to achieve airbreathing hypersonic flight involves challenges in different areas. Three problems related to scramjet engine design are addressed in the present paper. Stochastic computational techniques are utilised to optimise the total pressure recovery, considering two different inlet geometries, 2D planar and Bussenann geometry, the uninstalled thrust, and the geometry of the combustion chamber to keep constant the internal pressure considering heat addition. Partial results are presented for the three problems considered. It is an ongoing study to optimise the performance of a scramjet at the conceptual and pre-design phase using metaheuristics and semi-empirical methodologies, leading to a more robust result to be explored in CFD computations.

Keywords : *hypersonic, scramjet, metaheuristics*

1. **Introduction**

Studies on hypersonic air-breathing propulsion and supersonic combustion started in the late 1950's [1], and only in 2002 the supersonic combustion was viable as a propulsion system in the flight of Hyshot II [2]. The benefits for its development had been discussed and there are many possible applications for this kind of technology, for either military or civilian applications [3]. From a military point of view with this technology would be possible to increase the capabilities of versatility, response time, survivability, and unfueled range [3]. The civilian benefits are long-range rapid commercial transportation and safe, affordable, reliable, and flexible transportation to low-earth orbit. Hypersonic air-breathing propulsion and supersonic combustion can become a game changing in space access with fully or partially reusable single or two-stage-to-orbit launch systems since this kind of air-breathing

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launch system improves safety, vehicle design robustness, increases mission flexibility, and reduces operating costs [3].

However, the advances in hypersonic technology are not easy and to have a completely developed hypersonic system would be necessary advances in the four Knowledge Management Space, which was developed by Matsch and McMasters, as explained in [4], and is shown in Fig 1.

Fig 1. Knowledge Management Space [4].

This development is necessary because it is a disruptive technology, so there are not many experiences about the challenges that this kind of vehicle will confront, and the quadrant about "Traps and Surprises" of Fig. 1 is responsible for a great amount of uncertainty to the project.

Thus, the effort is in the sense of minimizing these development risks, increasing the knowledge in the quadrant of Targeted Research (Fig. 1), and addressing the capabilities that must be developed to give robustness to the projects.

Russia demonstrated supersonic combustion in atmospheric flight in 1991 [5]. Later, in cooperation with ONERA and NASA, the combined cycle engine of subsonic and supersonic combustion technologies was demonstrated [5, 6]. Australia became the second country to realize supersonic combustion in flight with the scramjet HyShot II, developed by the University of Queensland [7, 8]. NASA demonstrated supersonic combustion in two flights of the X-43A in 2004, when for the first time, a vehicle powered by scramjet technology and producing lift by waverider technology proved the supersonic combustion decoupled from rocket engines [9, 10, 11, 12, 13, 14].

The achievement of supersonic combustion is significant since it illustrates the feasibility of using a supersonic combustion ramjet (scramjet), which despite the usual alternatives such as turbojets and ramjets, promotes a better specific impulse for higher Mach numbers [15]. In general, scramjet engines demonstrate better performance for Mach numbers greater than 6, considering that ramjet engines have limitations in their operating speeds due to the presence of extremely high temperatures and pressures in the combustion chamber by reducing the flow velocity for the subsonic regime. One of the major drawbacks is that it is incapable of providing a thrust at low speeds, which requires an initial booster.

This article presents an optimization approach utilizing metaheuristics to solve three problems related to the scramjet engine. It is a step in developing a framework for optimizing hypersonic vehicles.

2. **Bibliographic Review**

Conceptually, hypersonic air-breathing vehicles that use scramjet technology are vehicles without rotating components and appropriately use vehicle surfaces to compress the flow due the oblique shock waves. Scramjet could be divided into three main sections, inlet, combustion chamber, and expansion, where the flow is always supersonic along the entire vehicle/engine. Furthermore, they operate based on the Brayton cycle (Fig. 2) and are designed for hypersonic flight, i.e., Mach number above 5 [16].

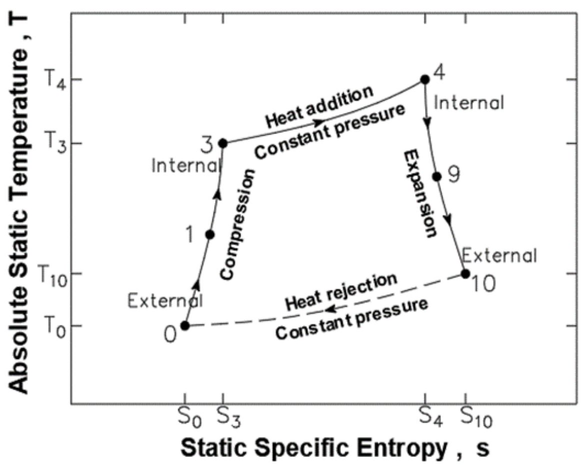

Fig 2. Ideal Brayton's cycle. Adapted from [16].

The fluid compression is accomplished by shock waves propagation in the engine´s surface (0-3), Fig. 3. During this process, the inlet flow is decelerated and heated into the combustion chamber (3-4) where the favorable pressure and temperature result in the ignition of the air and fuel mixture. Finally the nozzle accelerates the high-temperature and high-pressure flow resulting from the combustion (4-9) [16].

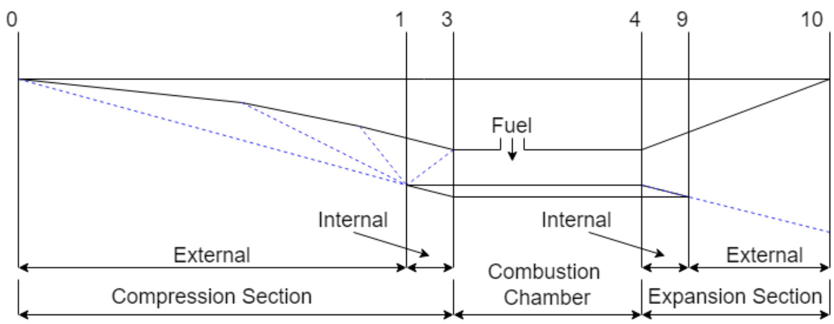

Fig 3. Scramjet Terminology.

The airflow is compressed by oblique shock waves at the compression section (inlet), reaching the required conditions to burn the fuel in the combustion chamber. The supersonic airflow exiting the combustion chamber is accelerated at the expansion section to generate thrust [16]. The process of combustion inside the scramjet occurs under constant pressure (Fig. 2), avoiding the boundary layer separation inside the combustion chamber and peak of pressure, for this purpose it could be designed a combustion chamber with variable area. Unlike the case of heat addition with constant area, when the pressure is kept constant, the flow kinetic energy is also kept constant in the frictionless case, which allows greater heat addition without choked flow, as explained in [16]. Also, it permits the scramjet to be a natural extension of the turbojet and ramjet engines.

The inlet has a crucial key to the performance of the scramjet engine. The airflow quality at the inlet exit affects the supersonic combustion and, consequently, the thrust. Heiser et al. [16] states that specific impulse will increase by 3%-5% for every 1% increase in compressive efficiency of the inlet. Smart [17] proposed the optimization of a two-dimensional scramjet inlet, with n shock waves, based on some manipulation of gas dynamics relations to maximize the total pressure recovery using Lagrange multipliers. The optimization solves a set of (3n + 2) equations to obtain the deflection angles, which maximize the objective function by applying a reasonable first guess using the Oswatitsch criterion

[18]. Raj and Venkatasubbaiah [19] presented a new optimization methodology to design the mixed compression scramjet inlet based on two criteria, the required Mach number at the inlet exit and maximum total pressure recovery on the external compression, which satisfy the shock on-lip condition. The Oswatitsch criterion, i.e., shocks with equal strength at the external compression, was applied to increase the efficiency of the inlet. Computational Fluid Dynamics (CFD) analysis was conducted to design the scramjet inlet considering the viscous effects. The correction factor was proposed to adjust the inviscid scramjet inlet design and achieve the shock-on-lip condition in the CFD analysis considering the boundary-layer development. Araújo et al. [20] optimizes the scramjet inlet using the Oswatitsch criterion to maximize the total pressure ratio reaching the temperature requirements at the inlet exit to burn hydrogen spontaneously in the combustion chamber. The influence of both the number of ramps and the flight Mach number were taken into account. The analyses cover the entropy generation, adverse pressure gradient, heat addition, and uninstalled thrust.

For best hypersonic air-breathing engine performance, the inlet must provide a large efficiency of compression. This inlet should be self-starting and for most applications should have, at the exit plane, a uniform velocity parallel to the freestream [21]. [22] first proposed an axisymmetric internal flow, consisting of an isentropic compression followed by a conical shock wave. The compression is initiated at the freestream Mach angle, and it is isentropic and conically symmetric, significantly increasing compression efficiency.

Some authors consider the design of a combustion chamber with a divergent section, aiming to maintain the pressure constant in the combustion chamber [12]. There are some numerical methodologies to calculate the thermodynamics properties with a generic geometry of a combustion chamber [23, 24, 25, 26]. However, the methods explored so far in the literature are too complex and involve numerical solutions. In [27], the authors apply a simplified method for calculation of thermodynamics properties in a flow inside a combustion chamber with variable area with an adapted form of the classical theories explained in [28], which are area ratio expansion, Rayleigh's flow and fanno flow.

Also, for the control of the pressure, were utilized metaheuristic's process to define the contour of the combustion chamber aim to optimize the geometry of the chamber and achieve a minimal root mean square error for the pressure between the stations 3 and 4 with respect to the pressure on the beginning of the combustion chamber.

According to [29], a metaheuristic is a technique that permits the user to achieve acceptable solutions in a reasonable time for solving hard and complex problems in science and engineering, however, it does not guarantee the optimality of the obtained solutions.

The metaheuristics can be subdivided into two groups: single-based (S-metaheuristic) and population-based (P-metaheuristic) solutions. The S-metaheuristic, e.g., tabu search [30, 31], simulated annealing [32] and vortex search [33], focuses on improving a single current solution at each iteration aiming to return the better result found at the end of the process. The P-metaheuristic, e.g., black hole [34], evolutionary algorithm [35], gravitational search [36], modified vortex search [37], particle swarm [38] and sine-cosine algorithm [39], focuses on improving a current population of solutions at each iteration aiming to return the better result found at the end of the process.

3. **Objectives**

The main proposal of this work is to apply methodologies for the development of systems based on scramjet vehicles and to optimize them from the use of metaheuristics. Three different cases are addressed:

- Case 1: Optimize the planar 2D scramjet inlet by maximizing the total pressure recovery based on temperature requirement at the inlet exit.
- Case 2: Optimize the uninstalled thrust of internally compressed axisymmetric scramjet vehicle based on Busemann geometry.

- Case 3: Minimize the root mean square error for the pressure inside the combustion chamber when compared with pressure in its entrance, aiming to promote a combustion process under constant pressure.

4. **Methodology**

The optimization problem was divided into three independent but related cases as follows:

4.1. **Case 1**

To optimize a planar 2D scramjet inlet with *n* shock waves at the compression section, *m* internal and n-m external shock waves, the mathematical modeling presented by [17] is used to maximize the total pressure recovery. For a minimization problem, the objective function is given by:

$$F = -\prod_{i=0}^{n-1} f_i^{\gamma/(\gamma-1)} g_i^{1/(\gamma-1)} \tag{1}$$

Where γ is the heat capacity ratio, f_i and g_i are the density and inverse pressure ratios across the ith oblique shock wave, respectively. The constraint associated with the requirements at the inlet exit is the temperature ratio [20, 40], different from the pressure ratio used in [17].

$$TR = \prod_i (f_i \ g_i)^{-1} \tag{2}$$

Also, there are two more constraints: All n shocks must satisfy the Rankine-Hugoniot relations and the flow enters the combustion chamber parallel with the freestream. The problem has 2n variables with n+2 constraints. Otherwise, only n-1 variables are defined as decision variables, while the others can be defined from constraints.

Metaheuristics were used to solve the optimization problem, and the results were compared with data available in [20].

4.2. **Case 2**

The Busemann geometry consists of a conical, axisymmetric, and internal isentropic compression that is followed by a conical shock wave that is canceled at the end of the compression section [39]. The symmetrical nature of the geometry produces a uniform flow at the entrance to the combustion chamber and the presence of a single shock wave translates into minimal entropy production and high performance, Fig. 4.

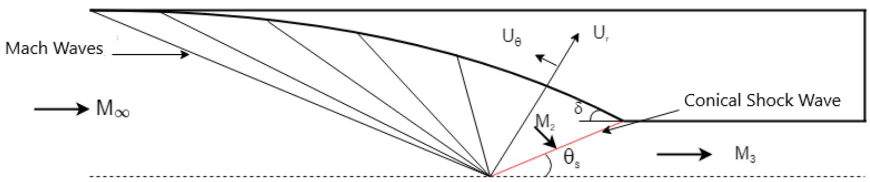

Fig 4. Parameters of the Busemann's geometry.

The discretization of the flow along the compression section obeys the Taylor-Maccoll equation and for the development of the geometry, the steps present in [41] are followed. The flight Mach number is a result of the optimization process of the choice of the Mach number M_3 and the total pressure ratio of the inlet.

For the combustion section, a simplified variable area combustion chamber was considered, that is, a chamber that keeps constant flow pressure throughout the combustion process.

The Area Ratio Theory is considered [28] for the expansion section, which relates the exit area of the combustion chamber with the final area of the nozzle, thus estimating the Mach number at the end of the vehicle expansion.

The design variables used and their limits are described in Table 1**.**

Table 1. Design variables of Case 2.

Name	Meaning	Inferior Limit	Superior Limit	Unity
TPR	Total Pressure Recovery	0.4	0.99	-
M_3	Mach Number at combustor entrance	1.5	5	-
f	Fuel ratio	0.005	0.1	-
r_2	Shock wave radius	0.1	1	m
r_{ext}	Nozzle radius	0.1	0.5	m
M_0	Flight Mach Number	7	7	-
H	Flight altitude	30000	30000	m

The objective function is directly related to the uninstalled thrust and the optimization problem tries to maximize this thrust. In addition, we want to obtain the highest value of the total pressure recovery and penalize the objective function as the calculated flight Mach number deviates from the desired flight Mach number. Thus, the objective function is defined as follows:

$$F_{Obj} = F_{Uninstalled} \cdot TPR \cdot \left[1 - ABS\left({M_0}_{Calculated} - M_0\right)\right] \tag{3}$$

To transform the problem into a minimization, the objective function is defined as:

$$F = 1/F_{Obj} \tag{4}$$

The restrictions for this case refer to temperature limits (entrance and exit of the combustor), dimensional limits and the calculated Mach number.

4.3. **Case 3**

The optimization problem is maintaining the pressure constant during heat addition in the combustion chamber. The objective is to find an arbitrary contour $f(x)$, as illustrated in Fig. 5, that better promotes that effect.

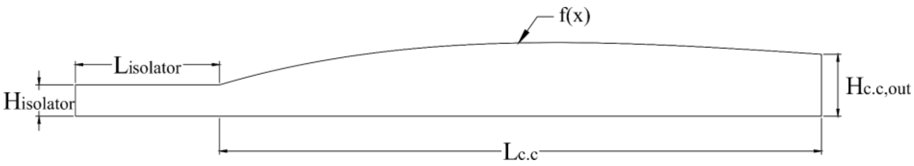

Fig 5. Generic combustion chamber with an isolator.

To address the optimization problem, the combustion chamber is divided in segments each one with a fixed angle of inclination. Such segments are called combustors, Fig. 6.

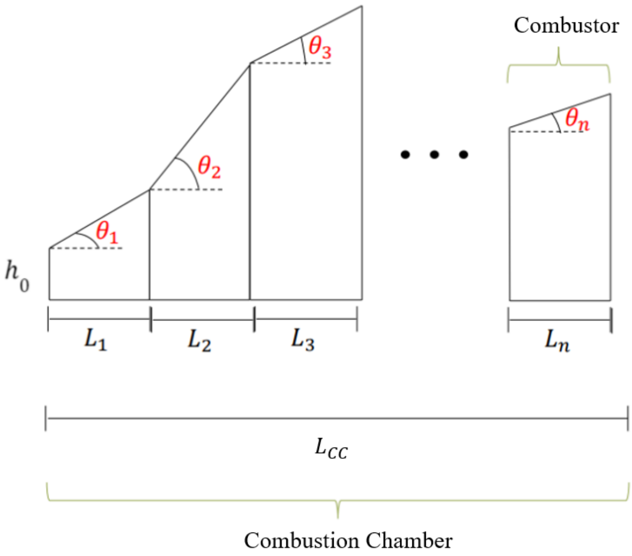

Fig 6. Combustion chamber subdivided in combustors.

Then, with a number of combustors large enough, it is possible to approximate the combustion chamber's contour with virtually any curve.

Due to this approach it is possible to define the input of the problem, shown in Table 2**,** and the results of the optimization are the angles θ_i.

For the calculus of the thermodynamics properties over the combustion chamber, each combustor is discretized in elements of $dx << L_i$ and, for each element, is applied the area ratio expansion, adapted fanno flow, and Rayleigh's theory for heat addition, respectively, in a superposition way.

To measure the pressure variation along the combustion chamber the root mean square error is used as the objective function and the objective of the problem is to minimize:

$$\sigma = \sqrt{\frac{\sum_{j=0}^{N} \left(p_j - p_{in}\right)^2}{N}} \tag{5}$$

The objective function is subjected to following constraints, which are derived from the adopted approach:

$$0° \leq \theta_i \leq 5° \tag{6}$$

Table 2. Design variable of Case 3.

Name	Meaning	Unity
h_0	Height of isolator	mm
w	Width of combustion chamber	mm
L_i	Length of each combustor	mm
n	Number of combustors	-
Q	Heat added	kJ/s
ρ	Density of the air in the isolator's entrance	kg/m^3
p	Pressure of the air in the isolator's entrance	Pa
T	Temperature of the air in the isolator's entrance	K
M	Mach number of the air in the isolator's entrance	-

5. **Results**

In case 1, the scramjet inlet was optimized via metaheuristics, using seven P-metaheuristic and one S-metaheuristic, considering five ramps at the compression section (6 shock waves), temperature ratio, TR, of 4.73, and flight at Mach number 7, which is one of the cases analyzed in [20]. In case 2, the scramjet vehicle was optimized for flight Mach 7 at 30 km altitude and a maximum dimensional radius of 0.5 m. Ten metaheuristics were utilized. In case 3, the optimization goal is to maintain constant pressure in a combustion chamber with four ramps. The thermodynamic conditions at the entrance were minimal, enough to promote the auto ignition of a hydrogen's mass flow equal to 0.0479915 kg/s. The metaheuristics convergence curve for all three cases is presented in Fig. 7.

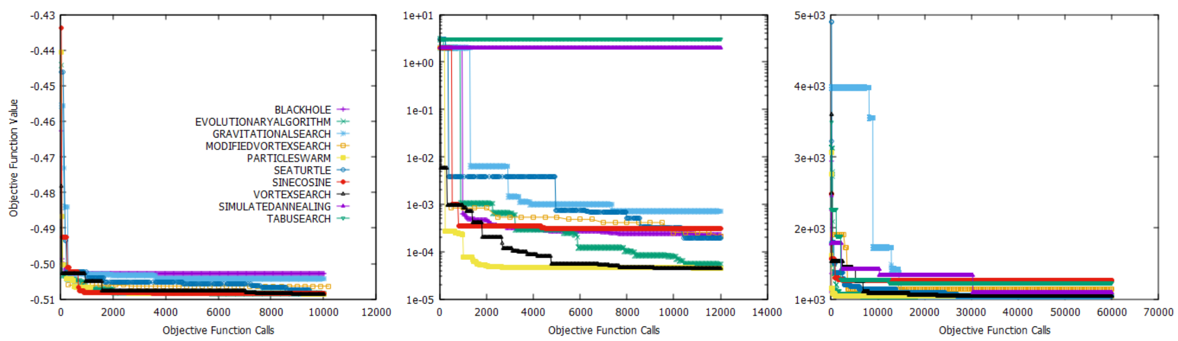

Fig 7. Convergence curve considering all metaheuristics used for all three cases. Case 1 on the left side, case 2 in the center, and case 3 on the right side.

5.1. **Case 1**

A set of metaheuristics, which uses 10000 objective function calls as stop criteria and ten runs for a population (for p-based metaheuristics) or neighborhood (for S-based metaheuristics) size of 30, were used to solve the 2D scramjet inlet optimization. All metaheuristics for case 1 are close to each other, not including tabu search and simulated annealing (Fig. 7). The three best were the particle swarm, vortex search, and sine-cosine algorithms. The results from the three better metaheuristics were compared with the data in [20] for a scramjet with five ramps, freestream Mach number 7 at 30 km altitude (TR = 4.73).

The deflection angles and temperature ratio results presented a slight variation, less than 4.3% compared with the baseline. Still, the temperature ratio reached on the inlet exit for particle swarm and vortex search algorithms was smaller than in [20] by about 0.02% (Table 3). Hence, the total pressure recovery, TPR, was approximately 0.2% greater than the baseline results [20] (Table 4)

Table 3. Ramp angles and temperature ratio from metaheuristics compared with [20].

Ramp Angle	Particle Swarm	Vortex Search	Sine-Cosine	Araújo et al. [20]
θ_0 [deg]	3.547	3.411	3.539	3.564
θ_1 [deg]	3.876	3.848	3.888	3.919
θ_2 [deg]	4.210	4.370	4.279	4.316
θ_3 [deg]	4.912	4.706	4.714	4.762
θ_4 [deg]	5.282	5.490	5.409	5.269
θ_5 [deg]	21.827	21.825	21.830	21.830
TR	4.730	4.730	4.731	4.731

The total pressure ratio across all oblique shock waves at the external compression is constant for the baseline results presented by [20], which uses the Oswatitsch criterion [18]. For the case analyzed in the present study, the optimum scramjet inlet had external shock waves with a total pressure ratio differing from each other by less than 0.31% (Table 4).

Table 4. The ith total pressure ratio and total pressure recovery from metaheuristics compared with [20].

Pressure Recovery	Particle Swarm	Vortex Search	Sine-Cosine	Araújo et al. [20]
π_0	0.980	0.982	0.980	0.980
π_1	0.980	0.981	0.980	0.980
π_2	0.981	0.979	0.980	0.980
π_3	0.978	0.980	0.980	0.980
π_4	0.980	0.977	0.978	0.980
π_{ref}	0.563	0.563	0.563	0.563
TPR	0.509	0.509	0.508	0.508

5.2. **Case 2**

In the Busemann based scramjet optimization, the neighborhood size was considered equal to 50, and 12000 objective function calls were evaluated for each metaheuristic. Each metaheuristic was executed 20 times. The best results of each metaheuristic is in Table 5.

The metaheuristics Vortex Search, Particle Swarm, and Evolutionary Algorithm present the best results for this problem. Simulated Annealing and Tabu Search failed to exit restrictions, resulting in unfeasible solutions.

In an ideal case, based on a deterministic solution, for a flight Mach number equal to 7 at 30 km altitude, the total pressure ratio of the inlet is 0.91855. The values of uninstalled thrust, inlet total pressure ratio, calculated flight Mach number, and their deviations from the ideal case for the three best results are presented in Table 6.

Table 5. Performance of each metaheuristic in Case 2.

Performance	Metaheuristic	Best Objective Function Value
1º	Vortex Search	4.47463e-005
2º	Particle Swarm	4.54145e-005
3º	Evolutionary Algorithm	5.59527e-005
4º	Sea Turtle	0.000193605
5º	Modified Vortex Search	0.000207865
6º	Black Hole	0.00023251
7º	Sine Cosine	0.000313079
8º	Gravitational Search	0.000726257
9º	Simulated Annealing	2
10º	Tabu Search	3

Table 6. Three best results for Case 2 .

Metaheuristic	Best Objective Function Value	Uninstalled Thrust [N]	Inlet TPR (% Deviation from ideal case)	M_0 Calculated (% Deviation from ideal case)
Vortex Search	4.47463e-005	24328.4	0.920046 (0.1629 %)	6.99821 (-0.0256 %)
Particle Swarm	4.54145e-005	24561.4	0.900027 (-2.0165 %)	7.00392 (0.0560 %)
Evolutionary Algorithm	5.59527e-005	19793.9	0.910472 (-0.8794 %)	7.0083 (0.1186 %)

Notice that the Particle Swarm Optimization obtained the best-uninstalled thrust, but the objective function was penalized due to the calculated value of TPR and deviation of M0 from the ideal case (Eq.3). The Vortex Search Optimization found the best result.

5.3. **Case 3**

The optimization of the combustion chamber considered a vehicle flying at 30 km altitude in Mach 6.8 to calculate the air's thermodynamic properties at the isolator's entrance after a compression process in the inlet. After a compression process, the air which enters in combustion chamber must be able to promote the auto-ignition of a hydrogen mass flow of 0.0479915 kg/s at 249.5 K, that is, after the mixing of the air and the hydrogen, the final temperature must be greater than 845.15 K [16].

The air's thermodynamic properties at the entrance of the isolator, considering an air mass flow of 1.4756 kg/s, are shown in Table 7.

Table 7. Air's thermodynamic properties for Case 3.

Name	Value	Unity
ρ	0.41657	kg/m³
p	130594.20	Pa
T	1092.12	K
M	2.372	-

The heat addition of 5756.77 kJ/s is a consequence of hydrogen combustion, which is considered an ideal process.

The combustion chamber was assumed to have a width of 300 mm and an isolator's height capable of capturing all the air's flow. Only results for the combustion chamber of four ramps are presented. Preliminary tests have shown that only small gains are obtained with more ramps in the combustion chamber. The geometry parameters considered are shown in Table 8.

Table 8. Combustion chamber geometry for Case 3.

Name	Value	Unity
h_0	0.01096	mm
$L_{isolator}, L_1, L_2, L_3, L_4$	50,60,60,60,60	mm
n	4	-

After 10 Runs, Black Hole, Particle Swarm, Evolutionary Algorithm, and Vortex Search achieved the same optimum objective function (Fig. 7). The results of all four metaheuristics with the best objective function values are presented in Table 9.

Table 9. Results of the metaheuristics for Case 3.

	Black Hole	Particle Swarm	Evolutionary	Vortex Search
θ_1 [deg]	1.77115	1.77114	1.77124	1.77114
θ_2 [deg]	1.78365	1.78363	1.78356	1.78363
θ_3 [deg]	2.10862	2.1086	2.10861	2.1086
θ_4 [deg]	2.3925	2.39259	2.39226	2.3926

σ **[Pa]**	1048.16	1048.16	1048.16	1048.16

Notice that even with the same objective function, the solutions present different angles (Table 9). Also, since the entrance pressure is 130.5942 kPa, the error RMS was just 0.8%, i.e., 1.04816 kPa (Table 9). Therefore, the pressure inside the combustion chamber varies in the range of [129.54604 kPa, 131.64236 kPa] on average.

6. **Conclusion**

This paper presents results for the independent optimization of a scramjet's inlet and combustion chamber. The adopted approach uses different metaheuristics, each with a limited number of runs, exploring distinct search algorithms instead of tuning the parameters of a given metaheuristic. The total number of runs for each case is more than one hundred. These results illustrate the initial steps of an ongoing project aimed at optimizing the entire scramjet in the conceptual and preliminary design phases using reliable low-order computations. The optimized and robust preliminary design should reduce the computational effort needed to execute a more robust evaluation by Computational Fluid Dynamics.

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