The Performance of a Boron Loaded Gel Fuel Ramjet

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The ramjet engine is an attractive propulsion system due to its simplicity and high efficiency. This work focuses on the possibility of combining the advantages of ramjet propulsion with the high energy potential of boron. Boron is an extremely energetic metal additive to fuels. However the use of boron poses two major challenges. The first, common to all solid additives to liquid fuels is particle sedimentation and poor dispersion. This problem is solved through the use of a gel fuel. The second obstacle, specific to boron enriched fuels, is the difficulty in realizing the full energetic potential of boron. This could be overcome by means of an aft-combustion chamber, where fuel rich combustion products are mixed with cold bypass air. Cooling causes the gaseous boron oxide to condense and, as a consequence, release the latent heat of evaporation trapped in the gaseous oxide. The merits of such a combination are assessed through its ability to power an air-to-surface missile, of relatively small size, capable of delivering a large payload to over a distance of about 1000 km. This paper details the preliminary design of a missile powered by a ramjet using a gel fuel loaded with boron, it deals with the thermochemical aspects of the two stage combustion of such a fuel, and it makes a comparison with a missile powered by a solid rocket motor and launched under the same conditions as the ramjet powered missile.

Nomenclature

AoA	=	angle of attack
C_D	=	drag coefficient
C_L	=	lift coefficient
C_M	=	pitching moment coefficient
D	=	drag
d	=	reference diameter
g	=	acceleration of gravity
F_s	=	specific thrust
f	=	fuel-to-air mass ratio
I	=	moment of inertia about the body longitudinal axis
I_{sp}	=	specific impulse
L	=	lift
M	=	pitching moment, Mach number
M_0	=	free stream Mach number
M_D	=	design Mach number
m	=	mass
$\dot{m}$	=	mass flow rate
$\dot{m}_f$	=	fuel mass flow rate
$\dot{m}_a$	=	air mass flow rate
$\dot{m}_{Ia}$	=	air mass flow rate going into the first combustion stage

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q	=	pitch rate
r	=	bypass ratio
S	=	reference area
st	=	stoichiometric
SRM	=	solid rocket motor
T	=	thrust, temperature
T_{04}	=	total temperature at the end of combustion
ΔT	=	temperature rise
$TSFC$	=	thrust specific fuel consumption
t	=	time
tot	=	total
V	=	velocity
x	=	horizontal coordinate
z	=	vertical coordinate
α	=	angle of attack
θ	=	flight path angle
φ	=	pitch attitude, equivalence ratio

I. Introduction

THE ramjet engine belongs to the air breathing propulsion family. A ramjet engine functions without moving parts. The compression of intake air is achieved by ram effect instead of a compressor. Air breathing engines are more efficient than chemical rockets because they use the surrounding air as oxidant rather than carrying oxidant on board. Eliminating moving parts allows operation at higher speeds and higher temperatures than turbojet engines. Consequently, the ramjet engine is a very attractive propulsion system. However relying on the ram effect for compression causes ramjet engines to have a significant shortcoming. At low flight speeds, the ramjet is very inefficient and even unusable due to insufficient compression. The vehicle must therefore be accelerated to a supersonic speed before the ramjet engine kicks into action. A rocket booster is usually used for this purpose. Fry[1] presented an excellent review of ramjet development and applications. The ramjet engine is also contemplated to power low cost missile systems.[2,3] Operational systems powered by ramjets include the Talos, the Sea Dart, the SA-6 Gainful, the Kh-31 and the Brahmos. Most operational systems use liquid fueled ramjet engines; some use solid fuel, gas-generator ducted rockets; none of the presently operational missiles is powered by solid fuel ramjets. The currently in development Meteor is based on boron enriched solid-fuel ducted rocket.

Boron is considered as an attractive additive to fuel because of its high energy content. Metals are commonly added to fuel to enhance performance. Among metal additives, boron is the most energetic. However extracting the energy stored in boron is a rather complicated task.[4] A proposed solution by Natan and Gany[5] is completing the combustion and energy extraction of the boron particles in an aft-burner with bypass air.

Adding metal particles to solid fuel is common practice. Liquid fuel on the other hand presents a challenge when it comes to solid additives. The solid particles tend to sink in the liquid fuel when subjected to gravitational and centrifugal forces. This problem is solved if a gel fuel is used instead of a liquid fuel. Metal particles are evenly dispersed in a gel fuel and they do not have a tendency to sink. A gelled fuel has solid-like behavior until shear stress is applied. Under shear stress the gel looses its viscosity and ends up behaving like a liquid. A gel fuel combines the storability properties of a solid fuel with the throttleability of a liquid fuel. An extensive review on gel fuels was conducted by Natan and Rahimi.[6]

This work considers the possibility of combining the advantages of a liquid fuel ramjet and boron as fuel additive, through the use of gel fuel and a bypass system for better exploitation of boron energy. A feasibility study for such an arrangement is done through the design of a missile capable of delivering a 500 kg payload to a range of about 1000 km. The missile is an air to surface missile, fired at Mach number 0.85, at an altitude of approximately 10 km. The goal is to achieve the range and payload capability of a strategic missile, with a size and ease of launch comparable to those of a tactical missile. Two systems for the aforementioned mission are considered, one powered with by a solid rocket motor and one powered by a boron loaded gel fuel ramjet.

II. Calculating the Trajectory

The performance parameter controlling this study is range. The range attainable is evaluated through a trajectory simulation. This simulation is based on a simple three-degree of freedom, point-mass model. The variables involved are the planar location, the angle of attack and the flight path angle. Acting on the point mass are lift, drag, thrust and pitching moment. The model, defined by Eq. (1), is schematically presented in Fig. 1. The differential equations in question are solved with the 4th order Runge-Kutta method. The aerodynamic coefficients for various altitudes and Mach numbers were found using "missile datcom".[7] The solver uses a bilinear interpolation on the "missile datcom" output to find the aerodynamic coefficients at each time step. The solver takes into account the changes in atmospheric properties with altitude.

$$\begin{aligned}
&\frac{dV}{dt} = -\frac{D}{m} - g\sin\theta + \frac{T\cos\alpha}{m} \\
&\frac{d\theta}{dt} = \frac{-g}{V}\cos\theta + \frac{1}{m}\cdot\frac{L}{V} + \frac{1}{m}\cdot\frac{T\sin\alpha}{V} \\
&\frac{dx}{dt} = V\cos\theta \\
&\frac{dz}{dt} = V\sin\theta \\
&\frac{d\varphi}{dt} = q \\
&\frac{dq}{dt} = \frac{M}{I} \\
&\varphi = \theta + \alpha \quad (\alpha = \varphi - \theta) \\
&D = \frac{\rho S V^2 C_D}{2} \quad L = \frac{\rho S V^2 C_L}{2} \quad M = \frac{\rho S d V^2 C_M}{2}
\end{aligned} \tag{1}$$

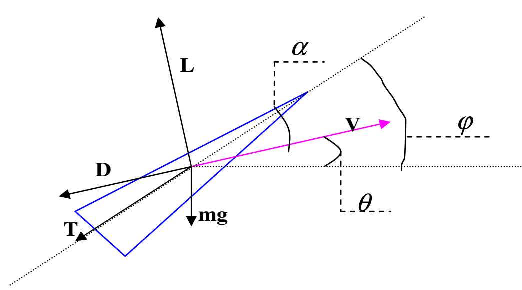

Figure 1. The model used for the trajectory simulation.

III. The Solid Rocket Motor Powered Missile

This section deals with an air-launched missile, carrying a 500 kg payload. The missile size is limited by the carrying platform. The maximum possible range is found for a high performance missile fitting into the geometrical constraints.

A. Propulsion

The factors leading the propellant choice are high performance and data availability. The propellant is composed of 70% AP, 14% HTPB and 16% Al, with a density of 1760 kg/m^3, burning rate coefficient of 0.364 and burning rate exponent of 0.21.[8] The combustion temperature, calculated with the CEA[9] code, is 3200 K.

We chose a 10 convex-points star grain, with a web fraction of 0.46. The grain geometry was chosen to provide a satisfactory volume loading (80%) and an almost neutral thrust profile. Reference 10 provides the methodology for the star grain burning calculations.

B. Sizing

This study deals with a missile launched from a plane. As a consequence, the size and weight of the missile are limited by the capabilities of the carrying platform. We limited the weight to 2500 kg and assumed that the carrying aircraft could accommodate a missile of 600 mm diameter.

As previously stated, the missile carries a 500 kg payload. The structure is assumed to constitute 10% of the total weight. The nozzle weight is assessed using Ref. 11. The weights of other elements were evaluated based on missiles of similar size and weight, and are presented in Table 1. The different component weights add-up to 825 kg. The remaining 1675 kg are assumed to be all-propellant.

Following common design practice,[12] the overall length to diameter ratio was set to 11 and the nose fineness ratio was set to 2, as a compromise between low drag and good packing capabilities. The aft-body length to diameter ratio was set to 0.7. Assuming densities[12] for the various components, the diameter and length accommodating all the missile units were found. The calculation took into account the geometrical constraints on fineness ratios detailed above. The resulting design has a length of 6.54 m and 595 mm diameter.

The resulting geometry was input into "missile datcom" and a tail was provided to ensure static stability.

Table 1. SRM powered missile component mass break-up.

Propellant	1675 kg
Structure	250 kg
Payload	500 kg
GNC	70 kg
Nozzle	5 kg
Total mass	2500 kg

C. Results

The aerodynamic coefficients of the statically stable configuration found with "missile datcom" were used as input to the motion solver equations together with appropriate initial conditions. The best range attainable, with the air-launched 2500 kg, is 222 km, much less than the desired 1000 km. The results of the simulation for maximum range are shown in Figs. 2 and 3. It is worth remembering that operational ballistic missiles of ranges close to 1000 km have much larger dimensions and are almost 10 times heavier than the missile discussed in this section.

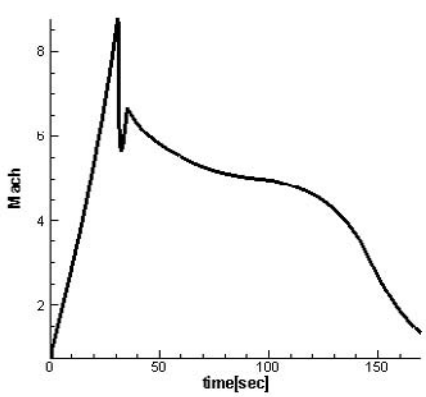

Figure 3. Mach profile of SRM powered missile.

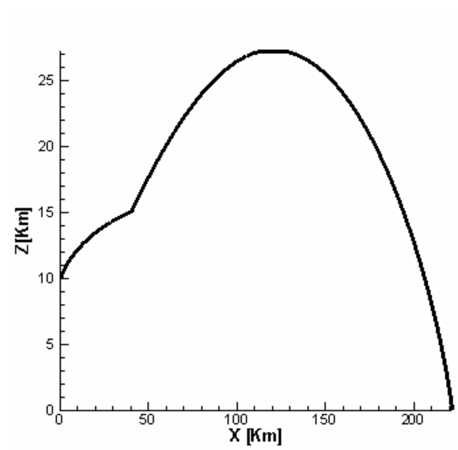

Figure 2. Trajectory of SRM powered missile.

IV. Boron as a Fuel Additive

Boron is a highly energetic element, with both gravimetric and volumetric heat of combustion significantly higher than those of commonly used fuels and fuel additives.[13] This can be seen in Fig. 4, which compares boron to other fuels.

A. Extracting the energy stored in Boron Oxide

Boron oxide, B_2O_3, has a boiling point of 2300 K[14] and a latent heat of vaporization of 366.5 kJ/mol. Therefore, realization of the full energetic potential of boron lies in the condensation of the boron oxide formed during burning. A two-stage combustor can allow better utilization of boron energy. The air flow coming through the inlets is split at the diffuser exit. The first part is burned with boron loaded gel fuel at a higher than stoichiometric fuel-to-air ratio. In the second stage the bypass air is mixed with the combustion products. With an adequately high bypass ratio, the addition of cold bypass air to the combustion products cools the mixture below the boron oxide boiling point, leading to the condensation of the boron oxide and consequently to the release of the latent heat of vaporization stored in its gaseous form. For this setup to be advantageous the overall fuel-to-air ratio should be less than the stoichiometric fuel-to-air ratio.

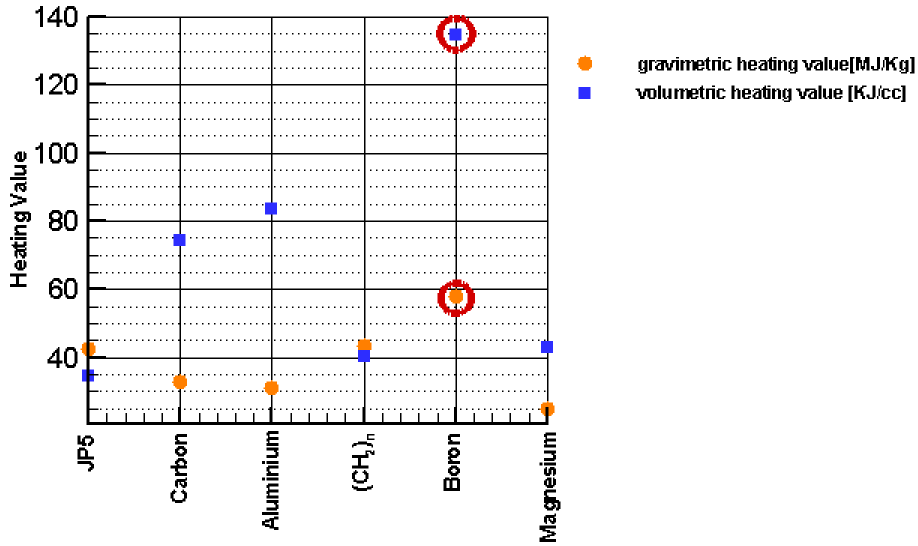

Figure 4. The heat of combustion of various fuels.

B. Thermo-chemical calculations

The prediction of the combustion temperature and products is performed using the NASA – Gordon and McBride code, CEA.[9] Two mixtures are examined. The first mixture serves as a reference and consists of a 100% of Jet-A fuel mixed with an organic gellant. The second mixture is composed by 60% wt. Jet-A fuel mixed with the gellant, and 40% wt. boron. Both fuel mixtures are assumed to have a temperature of 300 K. The oxidant is air at 670 K, and the reaction is supposed to take place at a pressure of 12 atm. The program was used to find the equilibrium temperature and the equilibrium compositions, specifically the molar fractions of the boron compounds in the combustion products. The calculations were performed for various equivalence ratios. The results are depicted in Figs. 5 and 6. These calculations correspond to the first step of the combustion where a part of the compressed air is mixed with the fuel and ignited

The choice of the equivalence ratio for the first stage of combustion is based on the quantity of boron oxide, relative to the other boron compounds. At an equivalence ratio φ=2, as could be seen in Fig. 6, the molar fraction of boron oxide B_2O_3 is larger than the molar fractions of most of the remaining boron compounds, and this equivalence ratio is chosen to be the working equivalence ratio for the first step of combustion. At this equivalent ratio, the mixture without boron reaches a temperature of 1830 K, whereas, the mixture loaded with boron reaches 2220 K.

The colder bypass air is now mixed with the hot combustion products leading on one hand to the cooling of the hot combustion gases, and on the other, to the completion of the combustion. It is assumed that after the addition of colder air, the gaseous B_2O_3, which is present at the end of the first combustion stage, does not react anymore with other species and undergoes condensation only. Figure 7 shows the temperature after mixing the bypass air, and adding the heat of condensation of boron oxide to the products, as a function of the bypass ratio. The bypass ratio being defined as the ratio of the mass of bypass air to the mass of air involved in the first step of mixing.

The working point bypass ratio was chosen such that the temperature of the gases exiting the combustor would be lower than the boiling point of boron oxide, but still be high enough to allow for satisfactory acceleration of the exhaust gases. This bypass ratio has the value r=3, and leads to a final temperature of about 2060 K.

The Jet-A, gellant and boron mixture used has a stoichiometric fuel-to-air ratio of f_{st}=0.08. The equivalence ratio of the first stage of the combustion is φ=2.0, thus the fuel-to-air ratio has the value f=0.16. The overall fuel ratio, with an r=3.0 bypass ratio, is found with Eq. (2) to be f_{tot}=0.04, which gives the overall equivalence ratio a value of 0.5. Burning Jet-A fuel only at this equivalence ratio would lead to a combustion product temperature of 1785 K, about 200 K lower than the temperature reached with the two step combustor and boron and gel fuel mixture. Moreover burning the boron containing mixture at the above equivalence ratio in a single step gives a temperature of 1950 K , more than 100 K lower than the two step combustion setup.

$$f_{tot} = \frac{\dot{m}_f}{\dot{m}_{atot}} = \frac{\dot{m}_f}{\dot{m}_{aI} + r\dot{m}_{aI}} = \frac{f}{1+r} \tag{2}$$

C. The influence of flight Mach number and altitude on combustion performance

The temperature and pressure of the air supplied to the combustion chamber change with flight altitude and Mach number. Hence, the influence of Mach number and altitude is studied through the influence of pressure and temperature of the reacting air. Calculations at 12 atm and 670 K led to the determination of a working point for the equivalence ratio, set to a value of 2, and the bypass ratio, set to a value of 3.

CEA calculations for the above values of equivalence ratios and bypass ratio and for air at 600 K and pressure varying between 4 and 16 atm led to results with a standard deviation of 0.21%, as seen in Table 2, thus the changes in the temperature at the exit of the combustion chamber can be considered negligible and the effect of pressure change can be ignored. The influence of temperature change was checked at a pressure of 8 atm only, since it has been established that pressure has little influence on the result. The results in Table 3 showed that the influence of temperature changes although small, is not negligible.

The relation between the supplied air temperature and the temperature rise attained can be linearly fitted with sufficient accuracy as is shown in Eq. (3).

$$\Delta T = 1642 - 0.355T \tag{3}$$

D. Ideal Performance

The above calculations refer to an ideal combustor: no losses in static pressure, 100% combustion efficiency, and total recovery of the latent heat of evaporation stored in the gaseous born oxide formed in the first combustion stage. This ideal combustor is incorporated in an ideal ramjet, a ramjet with ideal components, i.e. having 100% efficiency, and with exhaust gases pressure equal to ambient pressure. Operated at Mach number 3.5 at an altitude of 12 km, with an overall fuel-to-air ratio of 0.04, as previously discussed, the engine will provide 750 N of thrust, for each kg/s of air flow. Figure 8 shows the specific impulse and the thrust specific fuel consumption with an overall fuel-to-air ratio f_{tot}=0.04, for a gel fuel with and without boron. This figure shows the superiority of the two stage burner setup with a boron loaded fuel over more conventional burners and fuels.

Finally, the influence of the bypass ratio on the behavior of the specific impulse and the specific thrust, when the ideal ramjet is operated at Mach 3.5, at 12 km, is shown in Fig. 9. Ideally, the specific impulse could almost reach 2500 s; however, this comes at the expense of the specific thrust. The choice of bypass ratio is a compromise between an as high as possible specific impulse, and a satisfactory specific thrust.

Table 2. Temperature at the end of combustion of air at 600K.

p [atm]	T_{04} [K]
4	2024
6	2026
8	2028
10	2031
12	2032
14	2034
16	2036

Table 3. Temperature and temperature rise at the end of the combustion with supplied air at 8 atm.

T [K]	T_{04} [K]	ΔT [K]
500	1966	1466
550	1996	1446
600	2028	1428
650	2060	1410
700	2093	1393
750	2127	1377

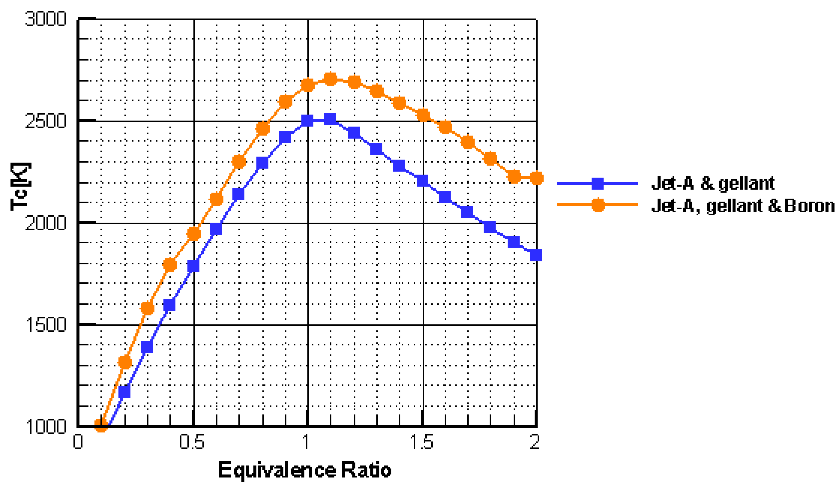

Figure 5 Combustion temperature for two gel fuel mixtures as a function of the equivalence ratio.

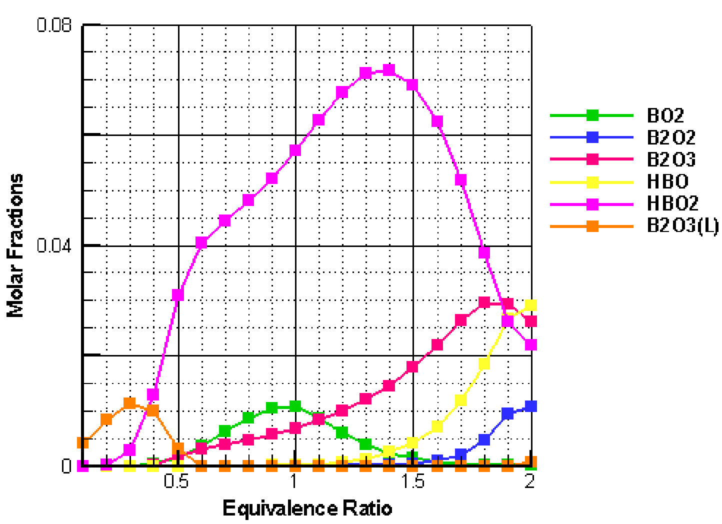

Figure 6. Molar fractions of boron compounds as a function of equivalence ratio.

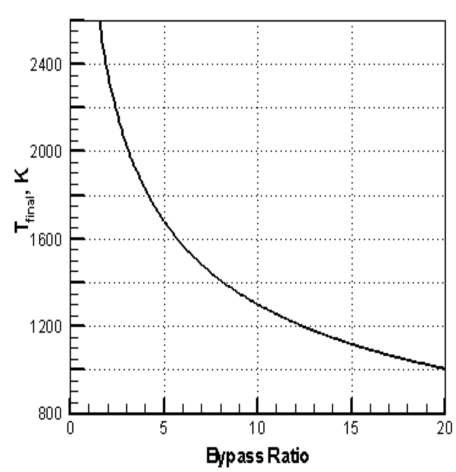

Figure 7 . Temperature reached after adding bypass air to the boron-fuel mixture combustion products as a function of the bypass ratio.

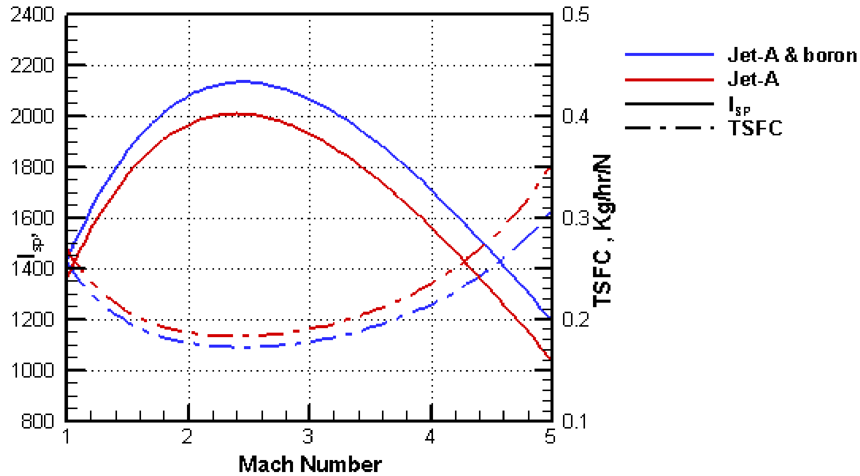

Figure 8. Ideal Ramjet performance for two fuel mixtures with the overall Fuel-to-Air ratio set to 0.04.

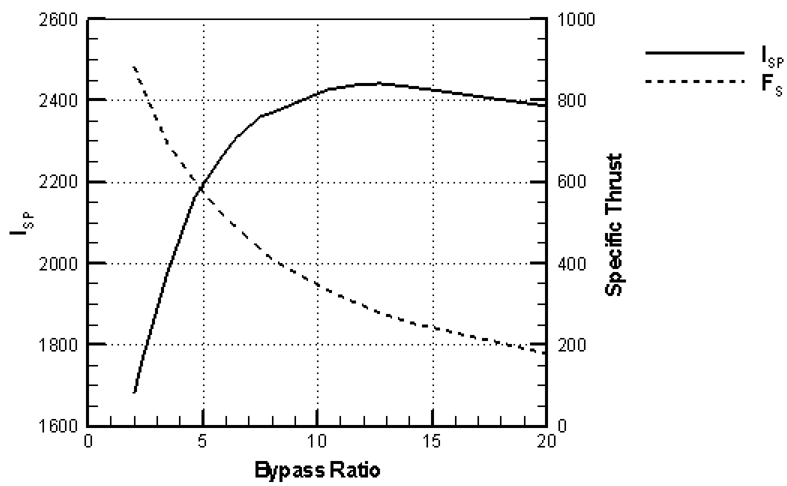

Figure 9. Ideal ramjet performance parameters at 12 km for various bypass ratios.

E. Variable Equivalence Ratio

As stated above, the working equivalence ratio chosen for the first combustion step is $\varphi = 2$. The effect of changing this equivalence ratio is studied through the variation of the ideal temperature found with the thermochemical computation program, CEA. Fig. 10 shows the results of these calculations. Based on the graph shown in Fig.9 the variation of the final combustion temperature for the 2.9 bypass ratio chosen for the ramjet operation with the equivalence ratio of the first step and consequently with the fuel to air ratio or the fuel mass flow rate . The result is shown in Fig 9.0;

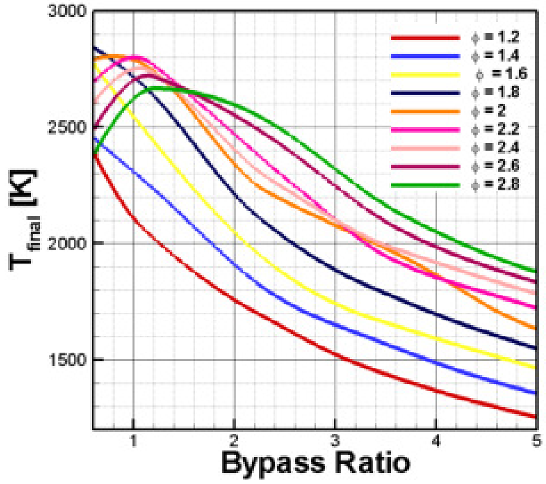

Figure 10. Temperature at the end of combustion vs. the bypass ratio for various first step equivalence ratios.

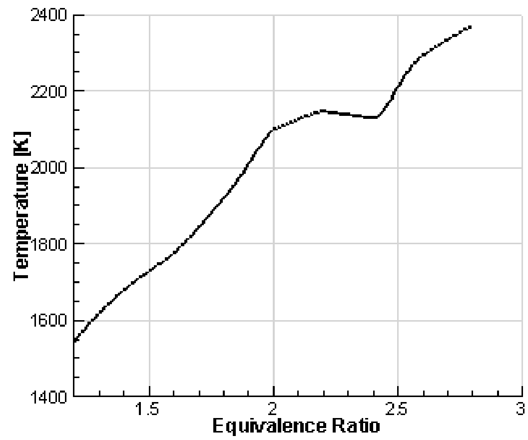

Figure 11. Final temperature variation as a function of various equivalence ratios with a bypass ratio = 2.9.

The Ramjet Powered Missile

This section describes the intake design and the sizing of a basic missile, powered by the ramjet engine discussed in this work, suitable for the mission described in the introduction.

A. The Air Intakes

1. Configuration

We opted for twin two-dimensional aft mounted inlets as shown in Fig. 10. Mounting inlets outside the main body increases drag, however a nose inlet reduces the volume available for the missile subsystems and the payload. Aft mounted inlets usually offer less pressure recovery than forward inlets, but they weigh less than forward inlets and have little impact on the payload delivery.[12] The compression is chosen to be external. The external compression mechanism is attractive mostly due to its simplicity. Moreover, external compression is less sensitive to internal fluctuations in the flow, as opposed to internal or mixed compression and is suitable for the Mach number regime in question.

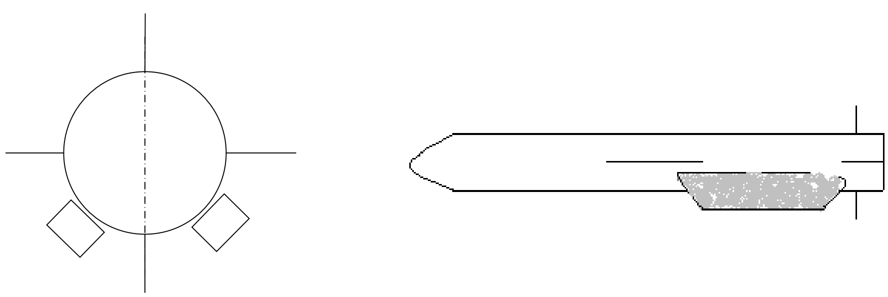

Figure 12. Schematic representation of the missile cross section with two-dimensional twin inlets.

2. The supersonic diffuser

The supersonic external compression diffuser slows the incoming flow through a series of oblique shocks followed by a terminal normal shock transforming the flow into subsonic. As the flow passes each shock wave it loses stagnation pressure. As the shocks get stronger, the losses become higher. An important parameter of a supersonic diffuser is the total pressure recovery, thus a series of small turns, that is, weaker oblique shocks, are preferred to a more sudden large turn.

The number of shocks giving an optimal pressure recovery depends on the free-stream Mach number. As the Mach number grows, so do the number of shocks needed, the limiting case being an isentropic compression ramp. However, increasing the number of oblique shocks leads to complicating the geometry and to increasing the weight and length of the intakes. This in turn leads to an increased drag. If the Mach number is not high enough the improved in pressure recovery achieved when using an isentropic compression ramp does not justify the losses and complication due to a longer ramp. Once the number of shocks and free-stream Mach number are determined, the total flow turning angle can be chosen to obtain optimal pressure recovery. Seddon and Goldsmith[15] supply data for determining the optimal shock number and the optimal total turning angle. This data was used in order to determine the number of turns and the ramp angles for various design Mach numbers.

When the intake is operating at the design Mach number, all the oblique shocks are focused on the lip and the terminal normal shock is in the throat section. Along with the turn angles and the oblique wave angles, this allows the determination of the supersonic diffuser geometry. With the geometry and the wave structure at the design Mach number known, it is now possible to evaluate the off-design performance of the various supersonic diffusers designed earlier. Sample results are presented in Figs. 11 and 12. For each compression ramp design, there exists a Mach number below which oblique shocks are replaced with a detached normal shock followed by subsonic flow. This shows as instability in Figs. 11 and 12. Also noticeable is the drop in capture area which leads to lower thrust due to lower air mass flow and to additional drag due to spillage. This behavior could cause a problem if the takeover Mach number is much lower than the design Mach number and not enough thrust is produced to overcome drag and accelerate to cruise speed.

The performance of the engine is evaluated with various inlets and at various flight speeds and altitudes. Efficiencies for the various components were assumed and drag coefficient, including additive drag when spillage occurs, is evaluated for a preliminary geometry using “missile datcom”. The goal is to have a wide enough range of Mach numbers where lift is higher than drag, and have the possibility to choose a ramjet-takeover Mach number where thrust is about 20% higher than drag in order to have sufficient acceleration. The reference area, i.e. the largest cross sectional area, and the air intake areas were determined iteratively. Both these quantities were varied until the performance of at least some of the inlets was satisfactory. In the final round, the cross sectional diameter was set to 420 mm and each intake inlet area was set to 0.025 m^2. Sample results are presented in Figs 13 and 14. The above calculations were performed in order to choose a suitable inlet and decide at which Mach number the ramjet engine will start operating. Based on the results, we chose the inlet designed for optimal performance at a design Mach number of 2.5, because this inlet offers a good performance over the widest range of Mach numbers and altitude among the studied intakes. It was decided to accelerate the missile with a booster to a takeover Mach number of about 2.2. This Mach number allows the generation of at least 20% more thrust than drag.

The supersonic diffuser is followed by a throat section, then a subsonic diffuser which slows the flow even more. The flow is finally slowed one more time when dumped into the combustion chamber. The Mach number at that point is about 0.2-0.3.

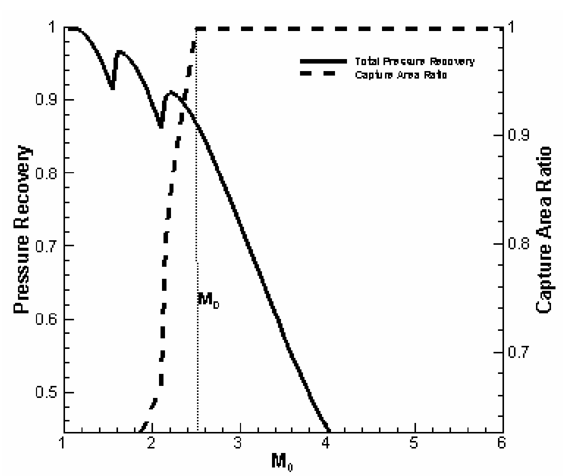

Figure 13. Off-design performance of an inlet designed for Mach = 2.5.

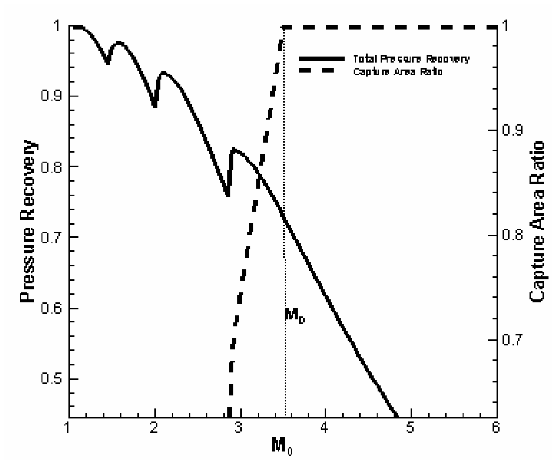

Figure 14. Off-design performance of an inlet designed for Mach = 3.5.

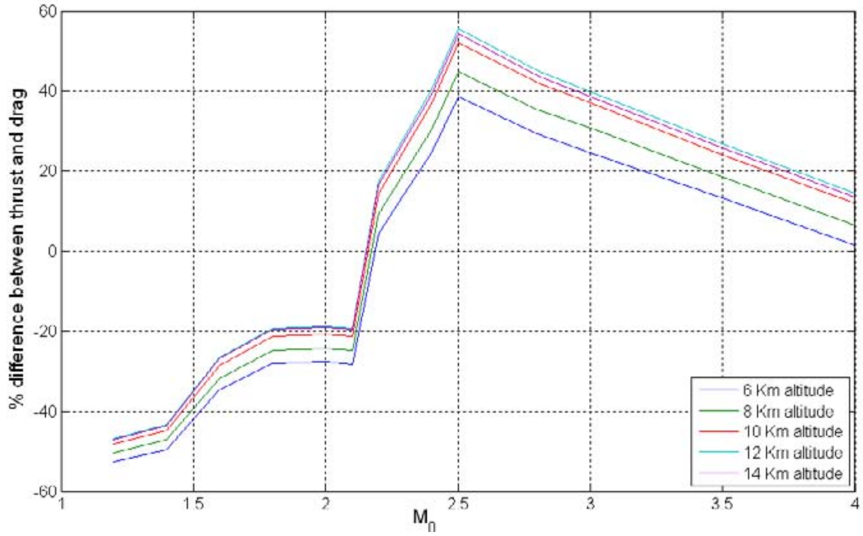

Figure 15. Performance of inlet designed for Mach 2.5.

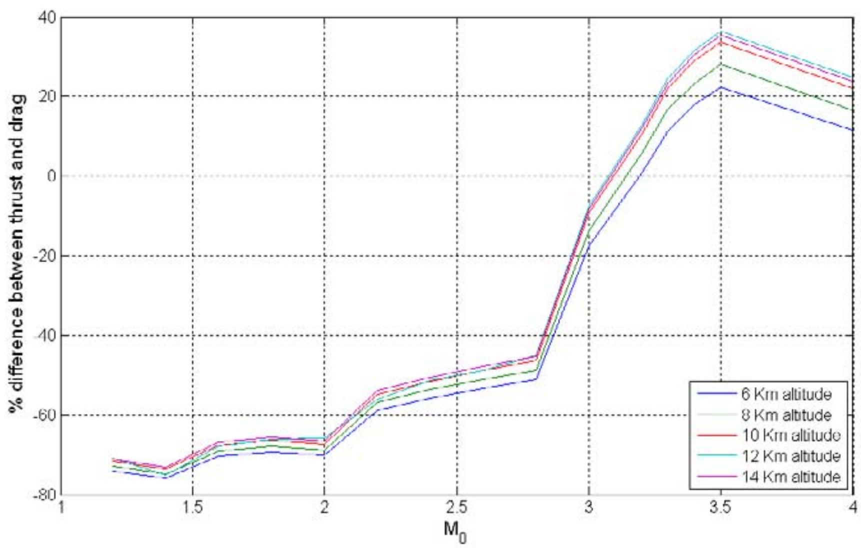

Figure 16. Performance on inlet designed for Mach=3.5.

B. Sizing

A basic configuration is sized in order to inspect the engine performance. The cross-sectional diameter has already been set to 420 mm, and the intake inlet area has also been previously set to 0.025 m^2. The intakes are supposed to be similar and rectangular with constant width of 0.25 m and a height starting at 0.1 m at the inlets and varying as the intake cross sectional area varies. The width is chosen much larger than the height in order to reduce drag and increase the lifting behavior of the inlets. The length of the supersonic diffuser is 0.14 m and the subsonic diffuser, including the throat section is 0.2 m long. The bypass duct leads the bypass air from the subsonic diffuser to be mixed with the combustion products. The main combustion chamber should be long enough to ensure good combustion. The length chosen was 0.5 m, which is actually the bypass ducts length. The intakes are thus about 0.85m long. Based on the above, the length of the aft-mixing chamber for the completion of the combustion is 0.8m.

The nozzle is designed as to expand the hot gases to ambient pressure at an altitude of 8 km. The exit Mach number is about 2.27 and the exit area is 0.138 m^2, almost 420 mm diameter. Choosing the nozzle half angle to be small enough to avoid separation, the length of the ramjet nozzle is about 0.7 m.

The nozzle and the intakes remain fixed during the subsequent iterations. The 500 kg payload and the 75 kg GNC system remain constant too. The fuel feeding and atomization system is assumed to weigh 5 kg. The fixed geometry ramjet nozzle is assumed to weigh 15 kg and the jettisonable solid rocket booster nozzle is supposed to weigh 5 kg. The structure's weight should change with the geometry. However, it is assumed to be fixed at 240 kg, including the inlets, as a conservative step.

The sizing procedure is similar to that of the SRM powered missile. The length occupied by each component is evaluated. The iterative process starts by limiting the missile length to 8 meters, because of carrying capability and bending stability. The combustion chamber is filled with booster propellant. The booster grain has the same shape and composition as the grain used for the solid rocket motor powered missile. The iterations showed that it is possible to have a missile designed with an integrated booster. The remaining space is filled with boron loaded gel fuel, and again iterations were made to find a quantity sufficient to reach the required range. The mass and length component break-up of the final configuration is detailed in Table 4. Finally, it should be noted that extra fuel has been provided for terminal maneuvers, and that, the missile datcom results indicate that the body and inlets can produce enough lift to carry the missile weight during cruising without having to install wings.

Table 4. The final configuration for the ramjet powered missile.

Component	Mass [kg]	Length [m]
GNC	70	0.85
Payload	500	2.1
Propellant	247	1.3
Gel Fuel	405	2.7
Other	10	0.4
Nozzles	15 + 5	0.7
Structure	240	
Total	1487	7.85
End of Boost	1240	
Empty	840	

C. Results

The missile above was considered in three different modes of ramjet operation: constant fuel-to-air ratio, constant fuel mass flow and variable fuel mass flow and fuel-to-air ratio. The bypass ratio was assumed to be constant and equal to 2.9. A variable bypass ratio implies variable inlets and thus keeping the bypass ratio constant will improve simplicity. The aerodynamic coefficients calculated for the missile show that the body and the inlets produce enough lift to carry the body during the sustain phase. During the ramjet operation, the flight is assumed to be level and the angle of attack is changed to ensure that lift produced can counter the missile weight . During the boost and unpowered phases, the angle of attack is set to zero in order to reduce drag. The ramjet operation starts at a mach number of 2.4 after the end of the booster's operation. . The calculations were made assuming a combustion efficiency of 0.98, a subsonic diffuser pressure recovery of 0.9, a combustor pressure recovery of 0.97 and a nozzle pressure recovery of 0.98. Earth's curvature was not taken into account while finding the range, hence the actual range will be longer than the calculated range.

1. Constant fuel-to-air ratio

In this case, the fuel-to-air ratio during ramjet operation is kept constant at $f = 0.04$. The air mass flow varies with the flight conditions and the fuel mass ratio varies accordingly. Using this setup, the missile reached a range of 1030 km within 17 min from launch. The ramjet operated for 15 s. Fig 17 shows the weight and lift variation during the ramjet operation. It can be seen that their values are close. The angle of attack needed to maintain this condition is also shown in Fig.17. During ramjet operation, the average thrust specific fuel consumption is 0.18 kg/hr/N and the average specific impulse is 2190 s. Their variation with time is shown in Fig. 18. The trajectory and variation of Mach number and thrust are shown in Figs. 19 and 20.

2. Constant fuel mass ratio

This is the simplest operation mode: the fuel mass ratio is kept constant during the whole ramjet operation. The variation of air mass flow with flight conditions will lead to changes in the fuel-to-air ratio. This will affect the temperature of the combustion products. Four different fuel mass flows were studied and the results are presented below in Figs. 21, 22 and 23. The mass flows used vary between 0.3 and 0.5. These mass flows ensure enough thrust to accelerate the vehicle to sufficiently high Mach numbers. When the fluid mass is not too high, the range requirement is met . The fuel mass ratio $\dot{m}_f = 0.5\,kg/s$ is highly inefficient and does not bring the missile to 1000 km. Lowering the fuel mass flow will lead to a more efficient trajectory and fuel can be saved , on the expense , however of the impact speed, as can be seen by comparing the results of $\dot{m}_f = 0.4\,kg/s$ and $\dot{m}_f = 0.3\,kg/s$.

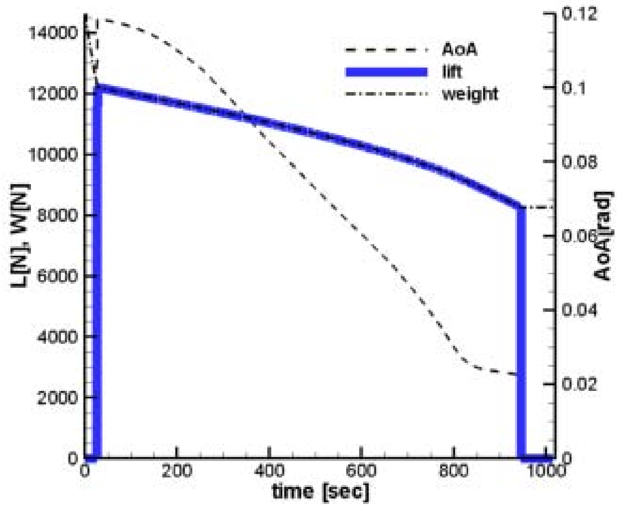

Figure 17 . Angle of attack, lift and weight variation with time for the constant air-to-fuel ratio case.

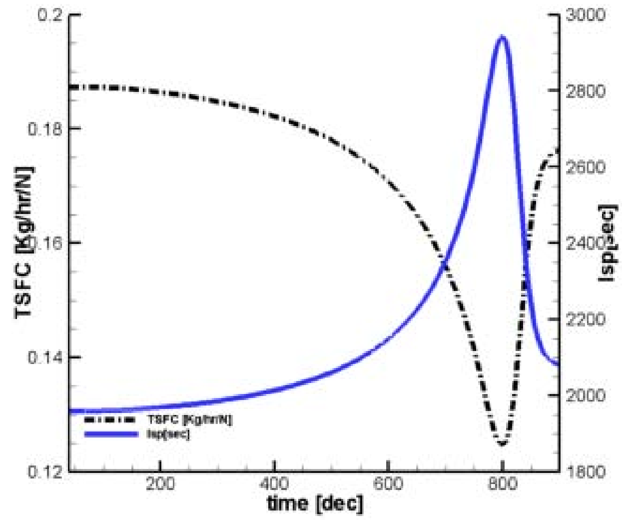

Figure 18. Specific impulse and thrust specific fuel consumption during sustain. for the constant air-to-fuel ratio case.

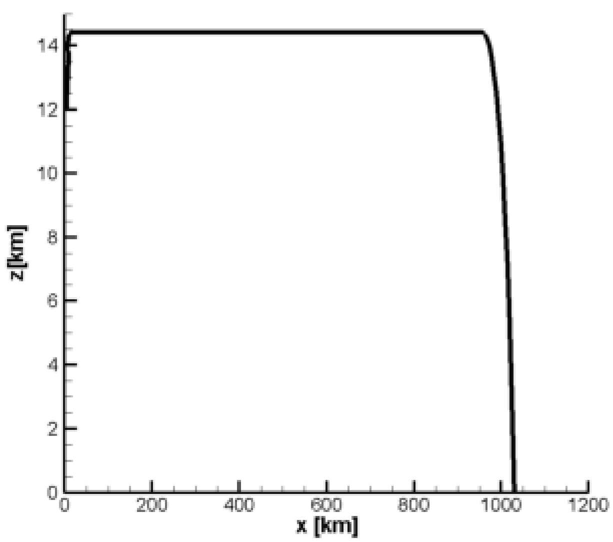

Figure 19. The boron loaded gel fuel ramjet powered missile trajectory when fuel-to-air ratio is held constant.

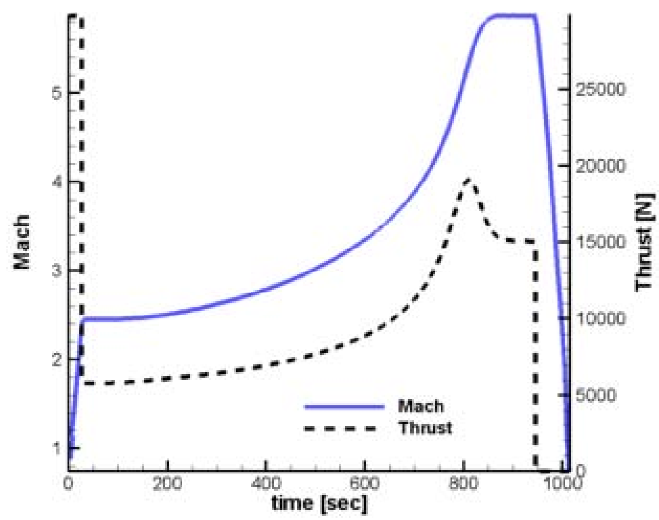

Figure 20. Mach number and thrust profiles for the ramjet powered missile when fuel-to-air is constant.

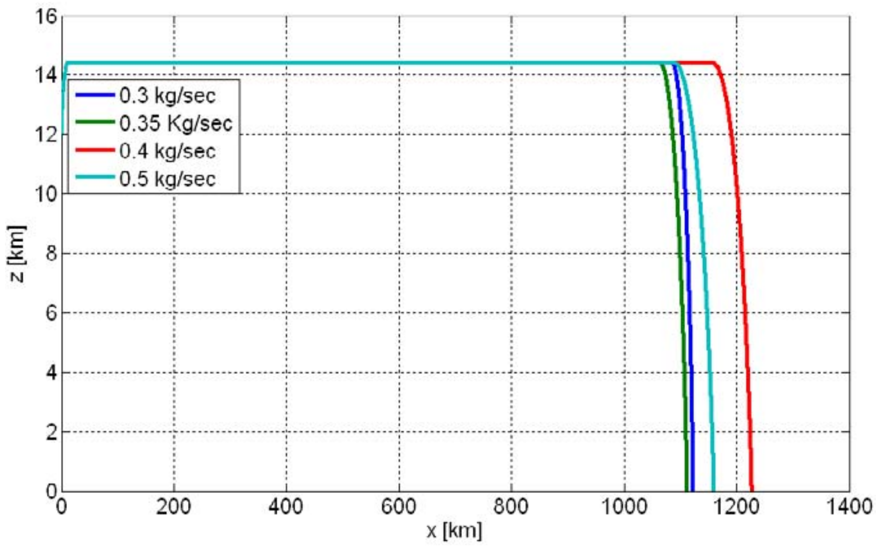

Figure 21. Trajectory for various fuel mass flows, when the fuel mass flow is held constant.

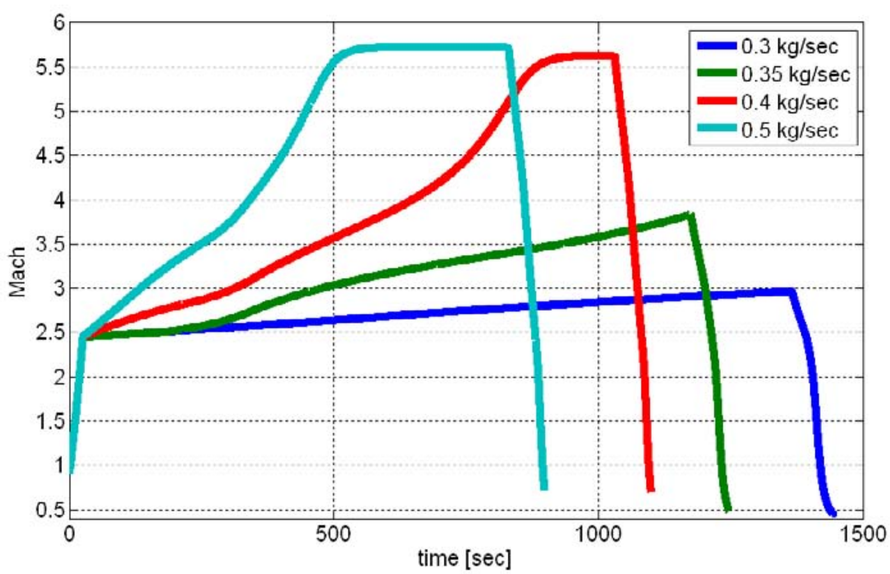

Figure 22. Mach number trajectory for various fuel mass flows, when the fuel mass flow is held constant.

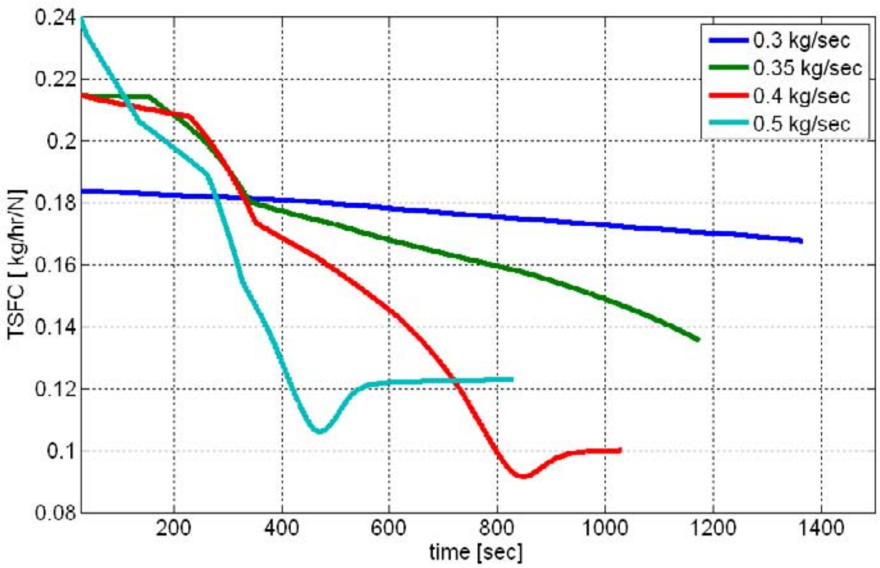

Figure 23. TSFC variation for various fuel mass flows, when the fuel mass flow is held constant.

3. Variable fuel mass flow and fuel-to-air ratio

As in the previous cases, during the ramjet operation, the flight is level and the angle of attack is changed so that the lift overcomes the weight. Additionally the fuel mass flow is changed in order to accelerate to a certain cruise mach number, and maintain it. The behavior of the mach number can be seen in Fig. 24 . The trajectories are shown in Fig. 25. This figure shows clearly the effect of the sustain Mach number on the attainable range. The average thrust specific fuel consumption and the average specific impulse in the sustain phase are detailed in Table 5. The trajectories in Fig. 25 and the values in Table 5 show that is a Mach number of 4 in the sustain phase offers a very good performance :in less than 18 min more than 1300 km can be reached, with an impact mach number of about 0.5.

Table 5 . Average TSFC and I_{SP} in the sustain phase for the variable fuel mass flow and variable fuel-to-air ratio

Sustain Mach Number	I_{SP} [s]	TSFC [kg/hr/N]
3.2	2300	0.15
3.5	2500	0.15
4.0	3000	0.12

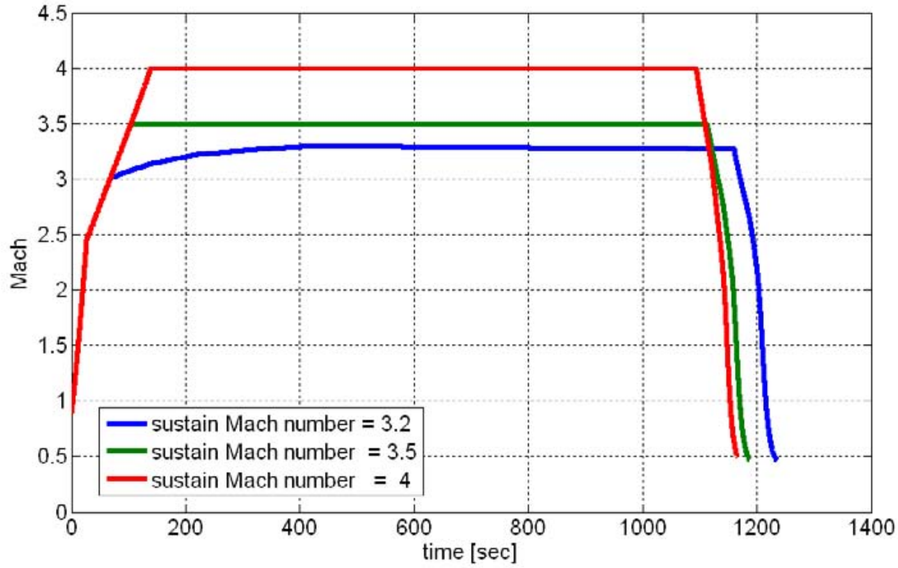

Figure 24. Mach number variation with time , when both the fuel mass flow and the fuel –to-air ratio vary to maintain cruising conditions.

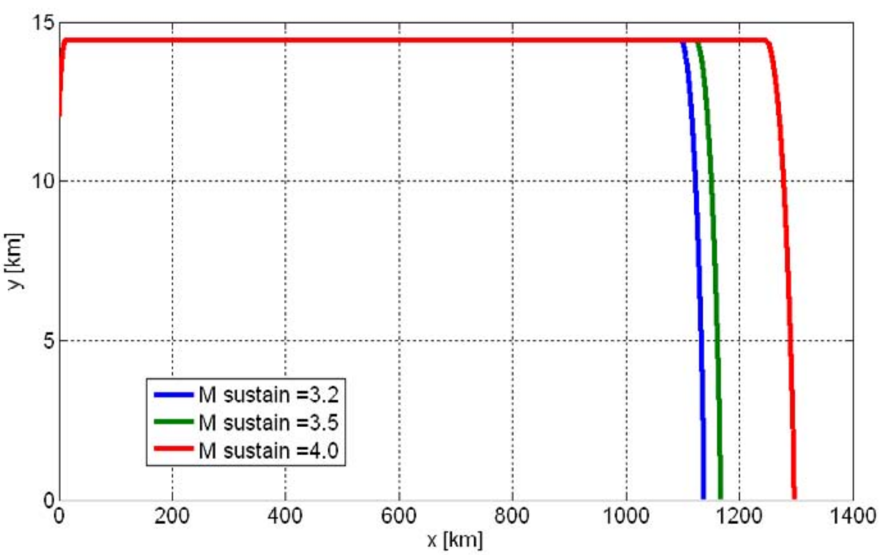

Figure 25. Trajectories for various sustain Mach numbers ,when both the fuel mass flow and the fuel-to-air ratio vary.

Conclusions

The present work proves the feasibility of a concept that combines the merits of ramjets, gel fuels and boron through the design of an air launched missile, of a relatively small size. The designed missile has a total weight under 1,500 kg, and delivers the 500 kg payload to a distance of more than 1000 km in about 17 min. Such a missile is a great improvement over rocket powered missiles of similar size, which have much shorter ranges. The solid rocket motor-powered missile studied in this paper, had a range of less than 250 km, for a total weight of 2,500 kg. Missiles for ranges close to 1000 km are of a much greater size and are difficult and slow to deploy. It was shown that the fuel quantity used can be reduced through appropriate fuel flow control, or alternatively, further targets can be reached with the same amount of fuel. This paper shows the advantage of adding bypass air to the boron enriched fuel combustion products and the simulation led to an impressive average specific impulse of 2190 s. This system can further be improved by optimizing the booster propellant quantity, by reducing the fuel consumption through better control, by slightly reducing the flight velocity, by designing more efficient intakes , and burning better fuel mixtures.

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