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Technical Memorandum
DRDC Valcartier TM 2006-675
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Abstract

The objective of the design work for the HEMi booster was to determine the missile configuration that provides the lightest system while achieving performance and payload objectives. The equations governing the kinematic performance of the missile were integrated into an aerodynamic prediction tool, a structural design module and a solid rocket motor performance model in order to perform an integrated analysis of the performance of the HEMi booster in its baseline configuration. Following the results obtained from the baseline configuration, several parametric analyses were performed to investigate design sensitivity and to develop a method of improving the performance of the design. Results of variations in mass, radius, length, motor chamber pressure and nozzle exit radius are presented. Analysis of the design space using parametric analysis showed that simultaneously achieving the length, mass and speed (lethality) requirements is challenging. To drive the design to a concept that meets these requirements, an optimization algorithm was used to adjust independent variables while keeping dependent values like velocity constant. This restricted the concept investigation to concepts that met the requirements. The integrated analysis tool indicated that the current HEMi configuration cannot achieve the initial specification of total missile length 1.25 m, mass 23.0 kg and velocity Mach 7 in 0.4 s. However, it is possible to reach Mach 7.0 if both the length and the mass of the missile are increased.

Résumé

L'objectif du travail de conception pour le propulseur de HEMi était de déterminer la configuration de missile qui fournit le système le plus léger tout en atteignant des objectifs de performance et de charge utile. Les équations régissant la cinématique du missile ont été intégrées à un outil de prédiction aérodynamique, à un module de conception de structure et à un modèle de performance de moteur-fusée afin de faire l'analyse intégrée des performances de la configuration initiale de référence de HEMi. À partir de la configuration de référence, plusieurs analyses paramétriques sont effectuées pour étudier la sensibilité de la conception et pour trouver les façons d'améliorer la performance de concept. Les résultats des variations de la masse, du rayon, de la longueur, de la pression de chambre du moteur et du rayon de sortie de la tuyère ont été présentés. L'analyse de l'espace de conception employant l'analyse paramétrique a prouvé que réaliser simultanément les spécifications de longueur, de masse et de vitesse (létalité) est un défi. Pour diriger la conception vers un concept qui répond aux spécifications, un algorithme d'optimisation est employé pour modifier les variables indépendantes tout maintenant constantes les valeurs dépendantes telles que la vitesse. Ceci permet de limiter la recherche aux concepts qui répondent aux spécifications. L'outil intégré d'analyse indique que la configuration HEMi actuelle ne peut pas réaliser les spécifications initiales d'une longueur totale de missile de 1,25 m, une masse de 23,0 kg et une vitesse du Mach 7 en 0,4 s. Cependant il est possible d'atteindre Mach 7,0 en augmentant la longueur et la masse du missile.

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Executive summary

The objective of the design work for the HEMi booster was to determine the missile configuration that provides the lightest system while achieving performance and payload objectives.

To expedite the HEMi concept refinement process, an effort was made to integrate in a single architecture many of the engineering analysis codes used to analyze and define the HEMi model parameter sets. Aerodynamic estimates are made using Missile DATCOM software. Nozzle length is computed as a function of nozzle exit radius, nozzle cone angle and throat area. The propellant performance module computes the thrust, specific impulse and mass flow rate of the propellant as a function of chamber pressure and nozzle throat and exit areas. The case and retention tube thickness and weight are computed as a function of chamber pressure and construction material properties. Using a one-degree-of-freedom simulation, the kinematic performance of the system is evaluated.

With the baseline configuration, the missile length is close to the requirement of 1.25 m but missile weight slightly exceeds the required maximum of 23.0 kg. The main design problem is that maximum missile velocity at end of boost is only 2052.7 m/s while a velocity of 2370 m/s is required for the dart to provide the desired lethality.

Following the results obtained from the baseline configuration, several parametric analyses were performed to investigate design sensitivity and to develop a method of improving the performance of the design. Results of variations in mass, radius, length, motor chamber pressure and nozzle exit radius are presented. Analysis of the design space using parametric analysis showed that simultaneously achieving the length, mass and speed (lethality) requirements is challenging. To drive the design to a concept that meets these requirements, an optimization algorithm was used to adjust independent variables while keeping dependent values like velocity constant. This restricted the concept investigation to concepts that met the requirements.

The integrated analysis tool indicated that the current HEMi configuration cannot achieve the initial specification of total missile length 1.25 m, mass 23.0 kg and velocity Mach 7 in 0.4 s. However, it is possible to reach Mach 7.0 if both the length and the mass of the missile are increased.

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Sommaire

L'objectif du travail de conception pour le propulseur de HEMi était de déterminer la configuration de missile qui fournit le système le plus léger tout en atteignant des objectifs de performance et de charge utile.

Les équations régissant la cinématique du missile ont été intégrées à un outil de prédiction aérodynamique, à un module de conception de structure et à un modèle de performance de moteur-fusée afin de faire l'analyse intégrée des performances de la configuration initiale de référence de HEMi. L'estimation des paramètres aérodynamique est faite à l'aide du logiciel Missile DATCOM. La longueur de la tuyère est calculée en fonction de son rayon de sortie, de l'angle du cône et de la surface de gorge. Le module de calcul de performance du propergol calcule la poussée, l'impulsion spécifique et le débit massique en fonction de la pression dans la chambre de combustion et le surface de gorge et de sortie de la tuyère. Les épaisseurs et masses du boîtier et du tube de rétention sont calculés en fonction de la pression dans la chambre de combustion et des propriétés physiques des matériaux. La performance cinématique du système est évaluée en utilisant une simulation à un degré de liberté.

Avec la configuration de référence, la longueur du missile est proche de la spécification de longueur désirée de 1,25 m mais excède légèrement le poids maximum de 23,0 kg. Le principal problème de conception est que la vitesse maximale atteinte à la fin de la phase d'accélération n'est que de 2052,7 m/s alors qu'une vitesse de 2370 m/s est requise afin que le dard produise l'effet terminal désiré.

À partir de la configuration de référence, plusieurs analyses paramétriques sont effectuées pour étudier la sensibilité de la conception et pour déterminer la méthode pour améliorer la performance de concept. Les résultats des variations de la masse, du rayon, de la longueur, de la pression de chambre du moteur et du rayon de sortie de la tuyère ont été présentés. L'analyse de l'espace de conception employant l'analyse paramétrique a prouvé que réaliser simultanément les spécifications de longueur, de masse et de vitesse (létalité) est un défi. Pour diriger la conception vers un concept qui répond aux spécifications, un algorithme d'optimisation est employé pour modifier les variables indépendantes tout en maintenant constantes les valeurs dépendantes telles que la vitesse. Ceci permet de limiter la recherche aux concepts qui répondent aux spécifications.

L'outil intégré d'analyse indique que la technologie qu'il représente ne peut pas réaliser les spécifications initiales d'une longueur totale de missile de 1,25 m, une masse de 23,0 kg et une vitesse du Mach 7 en 0,4 s. Cependant il est possible d'atteindre Mach 7,0 en augmentant la longueur et la masse du missile.

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1. Introduction

The objective of the design optimization work for the HEMi booster was to determine the missile configuration that provides the lightest system while achieving performance and payload objectives. The design optimization involves the design of an aerodynamic configuration and propulsion system that are realized through a feasible airframe capable of withstanding all the loads involved.

HEMi TD [1] attempts to demonstrate the following concept: an advanced kinetic energy (KE) penetrator (e.g. long rod, segmented or telescopic penetrators, or other novel geometries), accelerated to the hypervelocity regime within a 400-m range by a high-performance solid rocket motor, and flying at this regime to at least a 5-km range. Guidance and control (G&C) will be essential to achieve the desired hit probability. A minimum weight solution will have to be adopted to ensure maximum energy transfer to the KE penetrator in order to defeat a T-72S MBT between 400 m and 5 km. The following technologies are critical to enable the successful accomplishment of the system performance goals and objectives: propulsion, lethality, aerodynamics, guidance, control and structure. Concept design and system trade-off studies of the different technologies will make extensive use of physics-based and engagement-level modelling and simulation. Consideration of system integration issues will ensure optimum missile performance within a size and weight envelope estimated to be 1.25 m for maximum length and 23 kg for maximum weight.

Preliminary trade-off studies on the HEMi concept [2] revealed that a KE warhead was needed in order to meet the short time-of-flight specification and to defeat advanced armour, and a small two-stage configuration was needed to meet the stringent mass and length specifications. The two-stage configuration allows a high-output rocket-powered booster to bring a dart carrying the penetrator to hypervelocity. The dart then separates from the booster. The low-drag dart can then coast for a long distance with minor drag-induced deceleration.

The system level specification document [3] presents specifications for subsystems of HEMi in order to meet the system level performance requirements. These specifications provide design indications that allow subsystem designers to optimize their individual subsystem while ensuring that the overall system performance is met. However, this does not guarantee that the overall system is optimized.

In this study, the equations governing the kinematic performance of the missile were coupled to an aerodynamic prediction tool, a structure design module and a solid rocket motor performance model in order to perform an integrated analysis of the performance of the HEMi booster in its baseline configuration.

1.1 HEMi booster

The system level specification document [3] indicates that the booster must accelerate the 6.4 kg dart to a speed of not less than 2370 m/s in less than 0.4 s.

The booster must have a cavity along its centre-line to accommodate the dart. This cavity must be 0.050 m in diameter and 1.040 m in length. Total length must be less than 1.25 m.

Figure 1 presents an illustration of the proposed concept for the booster.

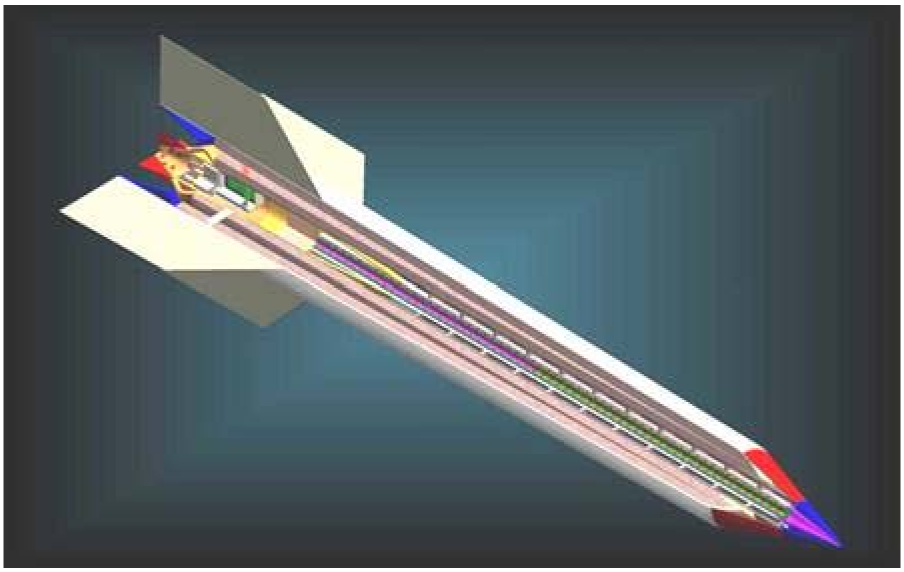

Figure 1. *Proposed concept for the booster*

The main components of the booster are the propellant, nozzle, outer case, dart retention tube and fins.

1.1.1 Propellant

In order to rapidly accelerate the dart to the hypervelocity regime within a 400-m range, the high-performance solid-fuel booster must employ a high-energy, high loading density and fast-burning propellant in order to maximize the delivered energy and minimize the burn time.

A rod and tube configuration for the propellant was chosen to ensure neutral burning area, which generates constant chamber pressure and minimizes the need for casing insulation.

The specific impulse of the propellant is a function of the operating pressure in the motor chamber and the exit area of the nozzle. Optimization of inert component mass involves primarily the motor casing and is achieved by operating the motor at a pressure level where the effect of increasing specific impulse is counteracted by the effect of increasing structure mass. The nozzle throat area is a design parameter used to adjust the chamber pressure.

1.1.2 Nozzle

The motor nozzle is the primary component for converting chemical energy into impulse. Ideal expansion ratio and nozzle profiling must be considered. With regard to expansion ratio, it is possible that a nozzle exit diameter which is greater than the actual motor diameter may be the optimal configuration.

1.1.3 Outer case

The outer case provides the structural integrity for the missile and must withstand the motor chamber pressure. Several materials were studied for the case construction [5]. A graphite-epoxy structure was selected.

1.1.4 Dart retention tube

The dart retention tube contains the cavity for the dart. The chamber pressure is applied on the dart. As with the outer case, a graphite-epoxy material was chosen for this component.

1.1.5 Aerodynamic configuration

Fins are needed to ensure that the booster is aerodynamically stable. Stability requirements were analyzed [4] and it was concluded that stability is not a crucial factor at velocities below Mach 1 because the thrust offset would be much greater than aerodynamic force. Thrust vector control is required to compensate for thrust misalignment. Additionally, the booster is expected to fly in the subsonic regime for less than 0.1 s. Wrap-around fins that provide stability at velocities greater than Mach 1 were chosen.

2. Problem formulation of the integrated analysis

In order to increase the pace of HEMi concept refinement process, an effort was made to integrate into a single architecture many of the engineering analysis codes used for the analysis and definition of the model parameter sets of HEMi. Aerodynamic estimation is calculated using the Missile DATCOM software [7]. Nozzle length is computed as a function of the nozzle exit radius, nozzle cone angle and throat area. The propellant performance module computes the thrust, specific impulse and mass flow rate of the propellant as a function of the chamber pressure and nozzle throat and exit areas. Case and retention tube thickness and weight are computed as a function of the chamber pressure and the structure's material properties. Using a one-degree of freedom simulation, the kinematic performance of the system is evaluated.

Figure 2 presents the interdependency of design and analysis codes in the integrated analysis architecture.

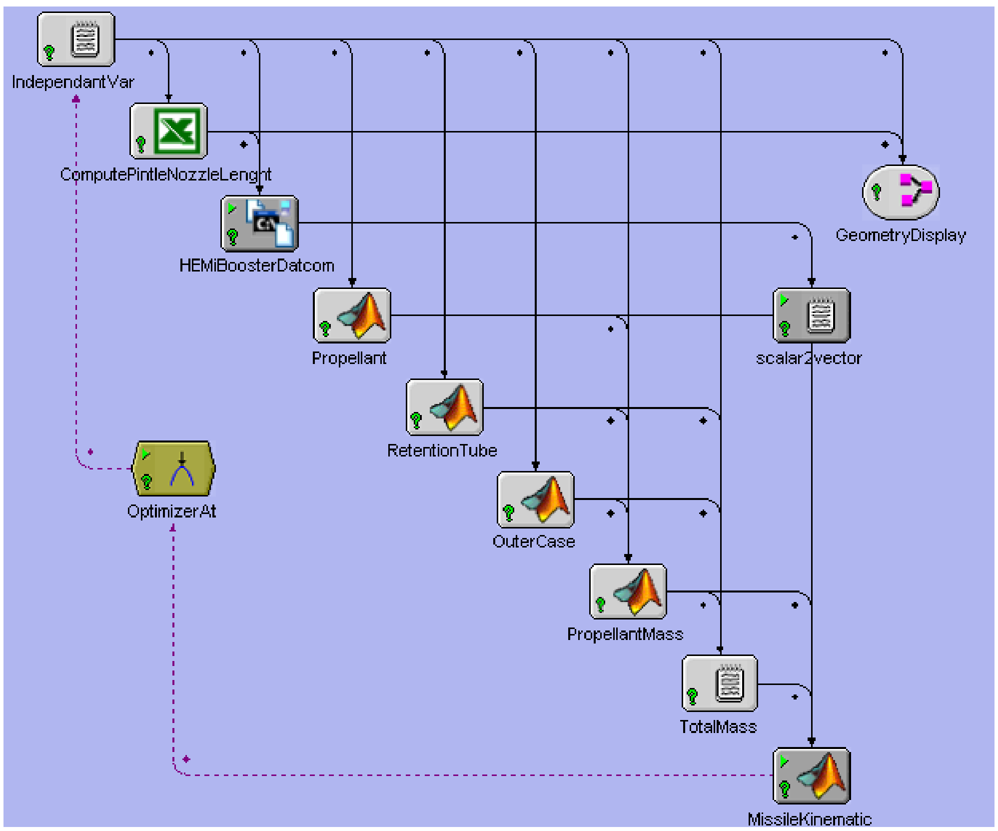

Figure 2. *Interdependency of design and analysis codes in the integrated analysis architecture*

The integration of the analysis codes permitted the linking of the interdependent variables allowing multidisciplinary analysis of HEMi. The codes were integrated in

Phoenix Integration Model Center™ which offers tools for parametric analysis and design of experiment.

However, the greatest benefit from this integrated analysis architecture is the possibility for optimisation algorithms to vary the concept independent variables in order to optimise the performance of the system thus providing multidisciplinary optimisation.

The integrated approach does not only simulate a specific realisation of a concept but also simulates a whole family of concepts defined by independent variables that can be varied infinitely. Emphasis is on the simulation of a feasible family of systems rather than a single point design simulation.

2.1 Nozzle length

This module computes the length of the nozzle as a function of the proposed missile geometry (Figure 3).

Nozzle length is computed as a function of the input parameters in Table 1. The corresponding output variables are given in Table 2.

Table 1. *Nozzle length module input parameters*

VARIABLE	INITIAL VALUE	DEFINITION
R_4[m]	0.075	Nozzle exit radius.
X_3[m]	1.15	Booster length excluding nozzle
A_t [m²]	0.0030	Nozzle throat area.
α [deg]	7.5deg	Nozzle cone angle

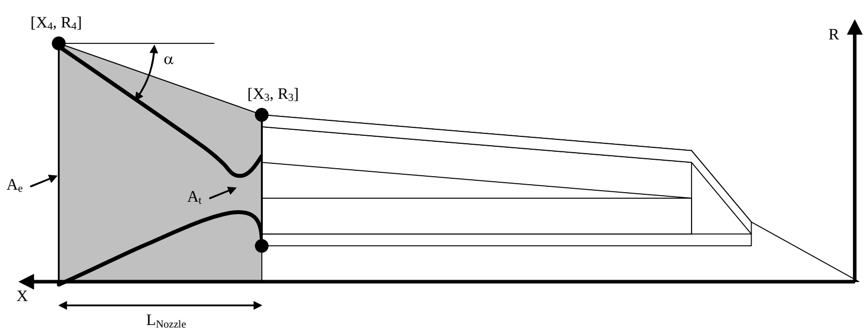

Figure 3. *Nozzle parameterization*

Table 2. Nozzle estimation output values

VARIABLE	DEFINITION
L_{Nozzle} [m]	Nozzle length
X_4[m]	Booster length

Length is computed so that expansion is similar to a baseline conical nozzle. Annex 2 presents the Excel spreadsheet used to compute the nozzle length.

2.2 Aerodynamic estimation

Aerodynamic estimation is made using the Missile DATCOM software [7].

The booster shape (Figure 4) is defined by two vectors, R and X, defining the radius of the body as a function of the position relative to nose tip.

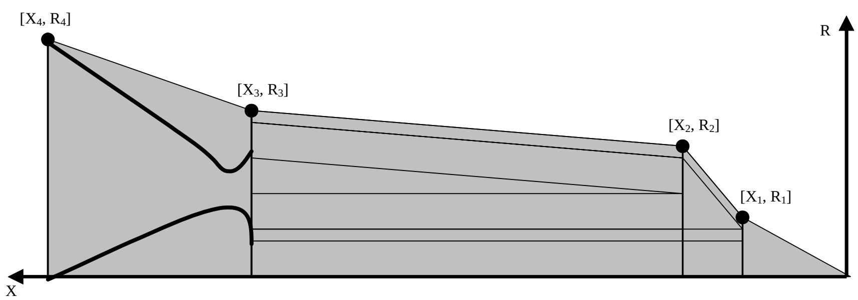

Figure 4. *Aerodynamic parameterization*

A file template for DATCOM is provided to indicate other values needed by DATCOM [7]. The template contains variables for R and X that will be substituted by their current values for the evaluation of the aerodynamic properties of the resultant airframe configuration.

Input parameters to the aerodynamic estimation module are given in Table 3. The corresponding output variables are given in Table 4.

__Table 3.__ Aerodynamic estimation input parameters

VARIABLE	INITIAL VALUE	DEFINITION
X_1[m]	0.1	Station position relative to nose tip.
X_2[m]	0.2	
X_3[m]	1.15	
X_4[m]	1.25	
R_1[m]	0.025	Body radius at corresponding station position.
R_2[m]	0.058	
R_3[m]	0.075	
R_4[m]	0.075	
Template filename	See Annex 1	Filename of the parameter file template used as input file by DATCOM.

__Table 4.__ Aerodynamic estimation output values

VARIABLE	DEFINITION
$\overline{M}$	Vector of Mach numbers for which drag coefficient $\overline{C}_{X0}$ is provided.
$\overline{C}_{X0}$	Vector of drag coefficient corresponding to the Mach numbers $\overline{M}$
L_{REF} [m]	Reference diameter used for computation of reference surface for drag force computation.

The reference diameter L_{REF} is hard coded in the template filename and has a constant value of 0.05m (dart diameter) and does not vary with the outside diameter of the booster. The Mach number vector (14 values) is also hard coded.

2.3 Propellant performance

The propellant performance module computes the thrust, the specific impulse and the mass flow rate of the propellant as a function of the chamber pressure and nozzle throat and exit areas. Input parameters are given in Table 5. Output values are given in Table 6.

__Table 5.__ Propellant performance module input parameters

VARIABLE	INITIAL VALUE	DEFINITION
R_4[m]	0.075	Nozzle exit radius.
A_t [m²]	0.0030	Nozzle throat area.
P_o [Pa]	$20x10^6$	Combustion Chamber pressure

__Table 6.__ Propellant performance module output values

VARIABLE	DEFINITION
F_d [N]	Motor thrust.
$\dot{m}_{prop}$ [kg/s]	Propellant mass flow rate.
I_{sp} [s]	Propellant specific impulse.
ρ_{prop} [kg/m³]	Propellant density.

Propellant performance model is further detailed in annex 2.

2.4 Outer case design

The case wall is sized to prevent failure either by exceeding the material's yield strength or the ultimate strain capacity at the aft-end of the booster because the largest diameter occurs at that location. Input parameters are given in Table 7 and illustrated in Figure 5. Output values are given in Table 8.

Table 7. *Outer case input parameters*

VARIABLE	INITIAL VALUE	DEFINITION
X_1[m]	0.1	Station position relative to nose tip.
X_2[m]	0.2	
X_3[m]	1.15	
R_1[m]	0.025	Body radius at corresponding station position.
R_2[m]	0.058	
R_3[m]	0.075	
P_o [Pa]	20×10^6	Combustion chamber pressure

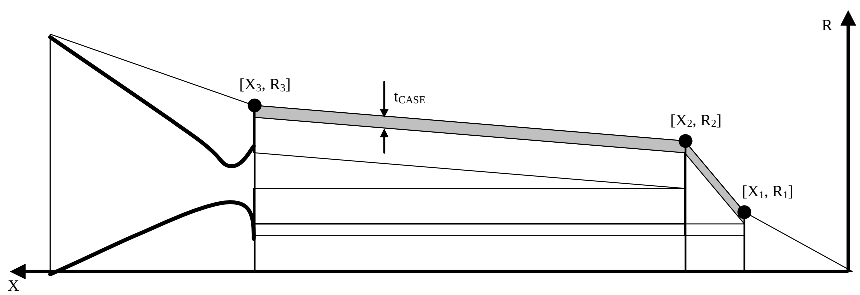

Figure 5. *Outer case parameterization*

Table 8. *Outer case output values*

VARIABLE	DEFINITION
t_{CASE}[m]	Outer case thickness
m_{CASE}[kg]	Outer case mass

The case thickness and mass are given by [5]:

$$t_{stress} = \frac{F_{safety} P_0 \cdot 2 \cdot \max(R_1, R_2, R_3)}{\frac{4}{\sqrt{3}} \sigma + F_{safety} P_0}$$

$$t_{strain} = \frac{\frac{F_{safety} P_0}{2E}\left(1 - \frac{\upsilon}{2}\right)}{\varepsilon + \frac{F_{safety} P_0}{2E}\left(1 - \frac{\upsilon}{2}\right)} \cdot 2 \cdot \max(R_1, R_2, R_3)$$

$$t_{CASE} = \max(t_{stress}, t_{strain})$$

$$m_{case} = \rho \cdot \pi \cdot t_{case}\left[(X_3 - X_2)(R_2 + R_3) + (X_2 - X_1)(R_1 + R_2)\right]$$

where material properties $(\sigma, \upsilon, E, \varepsilon)$ are defined in Table 9. A safety factor F_{safety} of 2.3 has been used.

Table 9. *Material properties*

Material	Yield stress $\sigma(\mathrm{MPa})$	Breaking strain $\varepsilon(-)$	Modulus $E(\mathrm{MPa})$	Poisson ratio $\upsilon(-)$	Density $\rho(kg/m^3)$
P-650/42 graphite-epoxy laminate	1720	0.01	170 000	0.30	1600

2.5 Retention tube design

HEMi was conceived as a rod and tube design where a penetrator is located on the central longitudinal axis of the booster. The purpose of the dart retention tube is to support the dart in the booster during the storage, launch and manoeuvre phases and to protect the dart from combustion gases. Figure 6 present the parameterisation of the retention tube. Input and output parameters are presented in Table 10 and Table 11 respectively.

Table 10. *Retention tube input parameters*

VARIABLE	INITIAL VALUE	DEFINITION
X_1[m]	0.1	Station position relative to nose tip.
X_3[m]	1.15	
R_{Tube}[m]	0.025	Retention tube inner diameter
P_o [Pa]	20×10^6	Combustion chamber pressure

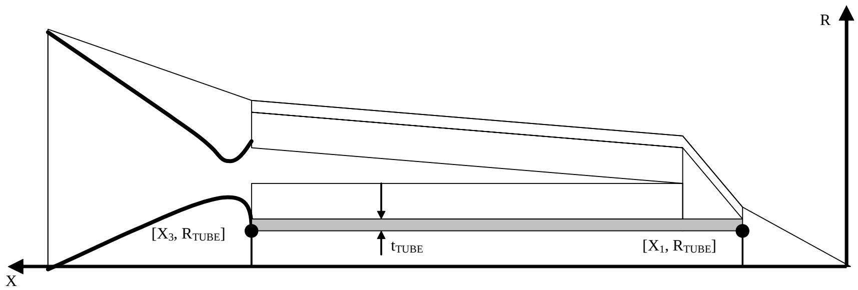

Figure 6. *Retention tube parameterization*

Table 11. *Retention tube output values*

VARIABLE	DEFINITION
t_{tube}[m]	Retention tube thickness
m_{tube}[kg]	Retention tube mass

The retention tube thickness is given by the minimum thickness of either the thickness required for sustaining the buckling, the yield or the longitudinal loads

$$t_{buckling} = 2 \cdot R_{tube} \frac{\left[\frac{1-\upsilon^2}{2E} F_{safety} P_0\right]^{1/3}}{1-\left[\frac{1-\upsilon^2}{2E} F_{safety} P_0\right]^{1/3}}$$

$$t_{yield} = \frac{\frac{F_{safety} P_0}{\sigma}}{1-\frac{F_{safety} P_0}{\sigma}} \cdot 2 \cdot R_{tube}$$

where t_{min} is the minimum thickness required to sustain longitudinal acceleration and has a value of 0.0027m.

For simplification, combined loads are not computed. Thickness is dictated by the greatest load:

$$t_{tube} = \max\left(t_{buckling}, t_{yield}, t_{min}\right)$$

The tube mass is computed according to the volume and the material density.

$$m_{tube} = \rho \cdot 2\pi \cdot t_{tube} \left(X_3 - X1\right)\left(R_{tube}\right)$$

where material properties $(\sigma, \upsilon, E, \varepsilon)$ are defined in Table 9. A safety factor F_{safety} of 2.3 has been used.

2.6 Propellant mass estimation

This function provides an estimate of the propellant mass based on the geometry volume (Figure 7) and the propellant density. Input and output parameters for propellant mass estimation are presented in Table 12 and Table 13 respectively.

Table 12. *Propellant Mass Estimation input parameters*

VARIABLE	INITIAL VALUE	DEFINITION
X_2[m]	0.2	Station position relative to nose tip.
X_3[m]	1.15	
R_2[m]	0.058	Body radius at corresponding station position.
R_3[m]	0.075	
t_{tube}[m]	From retention tube design	Retention tube thickness
t_{case}[m]	From outer case design	Outer case thickness
R_{Tube}[m]	0.025	Retention tube inner diameter
ρ_{prop} [kg/m³]	From propellant performance	Propellant density.

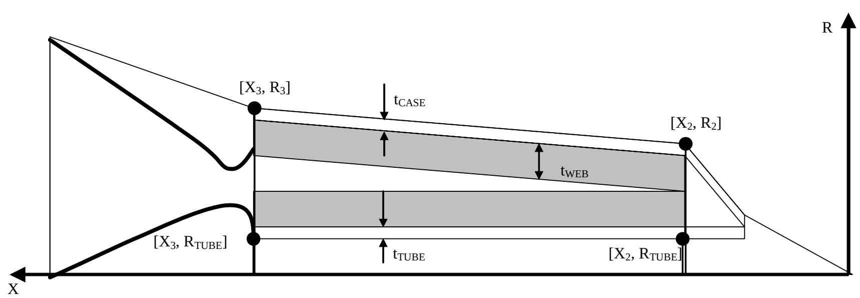

Figure 7. *Propellant mass estimation parameterization*

__Table 13.__ Propellant Mass Estimation output values

VARIABLE	DEFINITION
$m_{propellant}$[kg]	Propellant mass

$$m_{Propellant} = m_{InnerWeb} + m_{OuterWeb}$$

Where

$$m_{InnerWeb} = \rho_{prop}(X_3 - X_2)\pi\left((R_{TUBE} + t_{tube} + t_{WEB})^2 - (R_{TUBE} + t_{TUBE})^2\right)/2$$

$$m_{OuterWeb} = \rho_{prop}(X_3 - X_2)\pi\left((R_3 - t_{CASE})^2 - (R_3 - t_{CASE} - t_{WEB})^2 + (R_2 - t_{CASE})^2 - (R_2 - t_{CASE} - t_{WEB})^2\right)/2$$

$$t_{WEB} = \left(R_2 - R_{TUBE} - t_{CASE} - t_{TUBE}\right)/2$$

2.7 System mass estimation

System mass estimation provides total system mass by summing subsystem masses. Input and output parameters are presented in Table 14 and Table 15 respectively.

__Table 14.__System Mass Estimation input parameters

VARIABLE	INITIAL VALUE	DEFINITION
m_{tube}[kg]	From retention tube design	Retention tube mass
m_{case}[kg]	From outer case design	Outer case mass
$m_{propellant}$[kg]	From propellant performance	Propellant mass
m_{othert}[kg]	7.67	Other mass

__Table 15.__ System Mass Estimation output values

VARIABLE	DEFINITION
$m_{missile}$[kg]	Total Missile Mass

where m_{other} is the sum of the masses of the dart, nozzle, fins and any other items for which mass do not vary with the problem formulation.

2.8 Kinematics missile performance

A single degree of freedom simulation of the missile is used with the computed thrust, mass and drag coefficients. Input and output parameters are presented in Table 16 and Table 17, respectively. The block diagram of the Matlab/Simulink implementation is presented in Figure 8.

Table 16. *Kinematics missile performance input parameters*

VARIABLE	INITIAL VALUE	DEFINITION
$\overline{M}$	From aerodynamic estimation	Vector of Mach numbers for which drag coefficient $\overline{C}_{X0}$ is provided.
$\overline{C}_{X0}$	From aerodynamic estimation	Vector of drag coefficient corresponding to the Mach numbers $\overline{M}$
L_{REF} [m]	0.05	Reference diameter used for computation of reference surface for drag force computation.
F_d [N]	From propellant performance	Motor thrust.
$\dot{m}_{prop}$ [kg/s]	From propellant performance	Propellant mass flow rate.
$m_{missile}$[kg]	From system mass estimation	Total Missile Mass
$m_{propellant}$[kg]	From propellant mass estimation	Propellant mass

Table 17. *Kinematics missile performance output values*

VARIABLE	DEFINITION
V_{max}[m/s]	Missile maximum velocity at end of boost
T_{max}[s]	Missile flight time during boost phase
X_{max}[m]	Missile range at end of boost phase

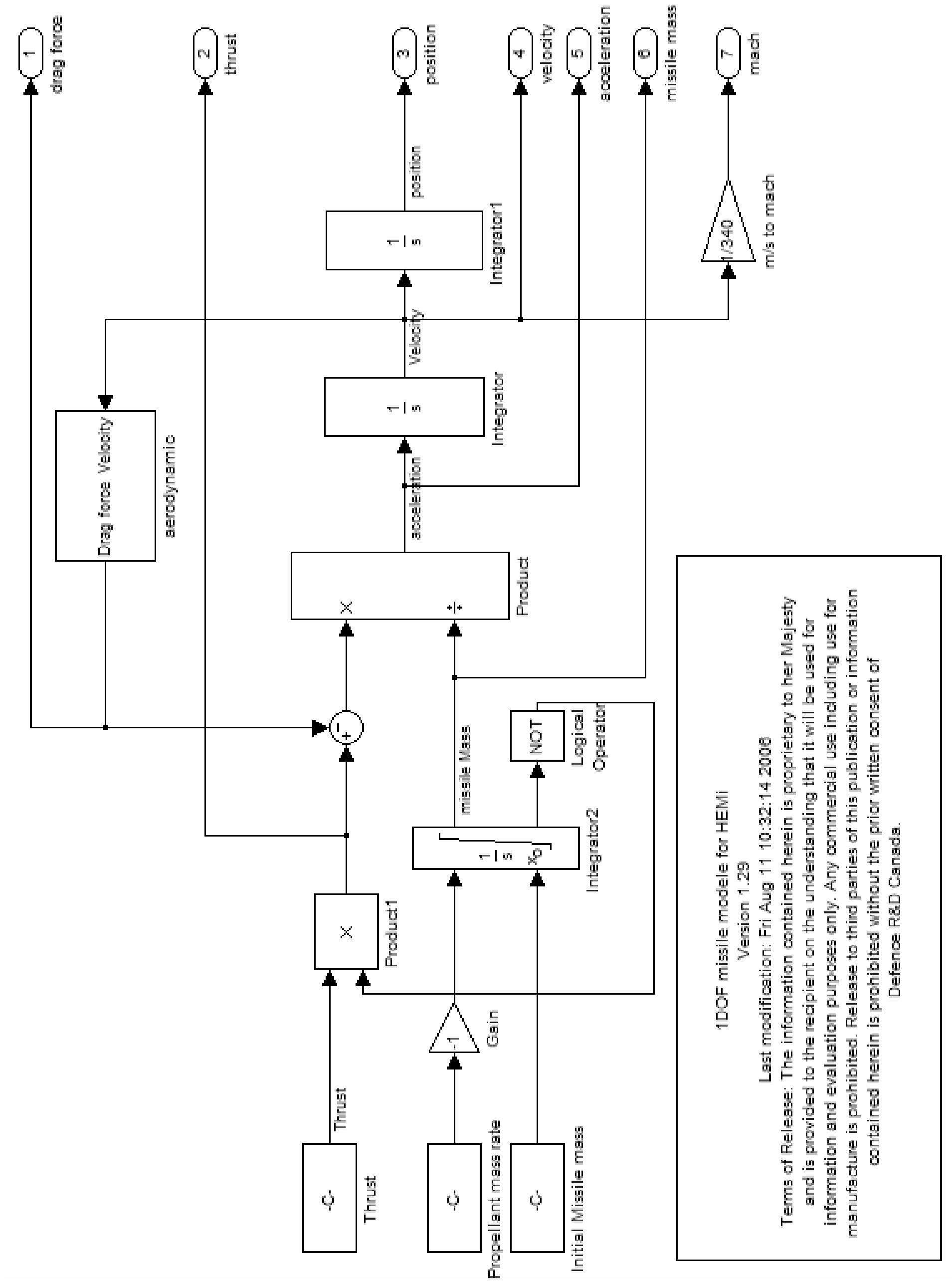

Figure 8. *Kinematics simulation model*

3. Integrated parametric analysis

The integrated analysis codes are used here to perform the analysis of the initial baseline configuration.

Following the results obtained from the baseline configuration, several parametric analyses are performed to investigate design sensitivity and to figure out methods for developing a method of improving the performance of the design.

3.1 Baseline results

A summary of the baseline performance values is presented in Table 18. Complete results are presented in Annex 4.

Table 18: Baseline performance

Variable	Units	Value
Booster length	m	1.253
Missile mass	kg	23.49
Propellant mass	kg	14.108
Propellant specific impulse.	s	248.06
Missile maximum velocity at end of boost	m/s	2052.7

From these results, it is obvious that the length of the missile is close to the length requirement of 1.25 m but slightly exceeds the required maximum weight of 23 kg.

The main design problem is that the maximum missile velocity at end of boost is only 2052.7 m/s while a velocity of 2370 m/s is required in order for the dart to provide enough lethality.

A sensitivity analysis on several parameters will be performed in order to investigate ways of improving the design.

3.2 Sensitivity to inert mass

The payload of the system is represented by a fixed mass m_{other} of 7.67 kg. Since the baseline missile slightly exceeds the weight specification, the inert mass m_{other} has been varied and the resulting performance parameters are presented in Figure 9.

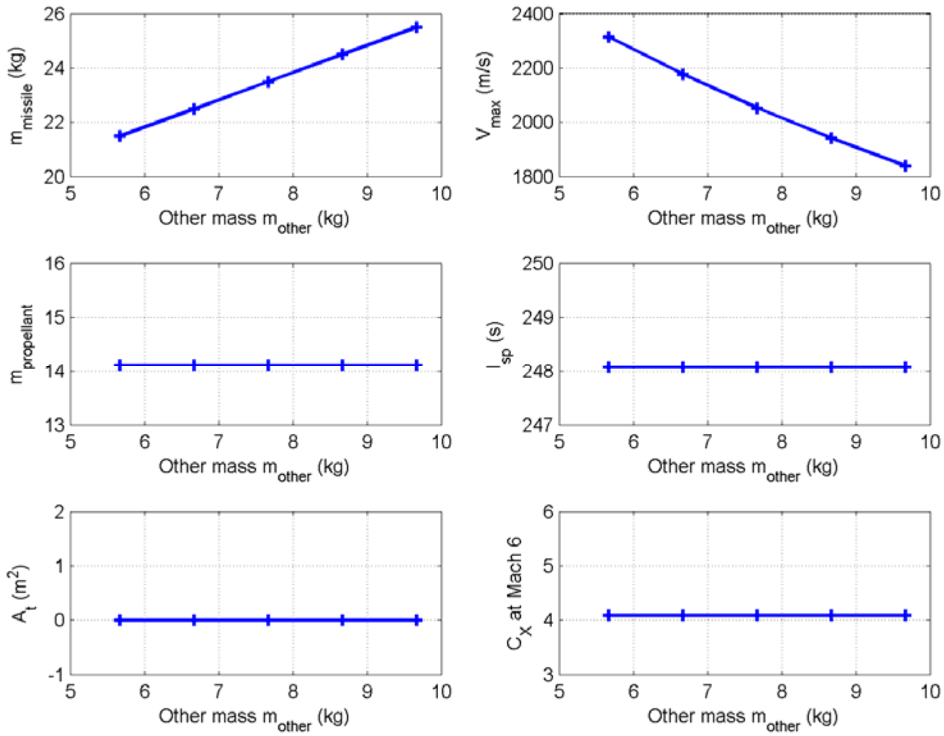

Figure 9: Sensitivity to inert mass

The results clearly show that decreasing the mass of the missile permits an increase in terminal velocity.

From a value of 7.67 kg and a constant mass of propellant, an inert mass reduction of at least 2 kg is required to obtain the desired velocity. However, the HEMi missile system mass budget is very stringent. It is very unlikely that weight components such as the dart, nozzle, fins, etc. could be reduced by this much.

3.3 Sensitivity to increase in the missile diameter

To increase the maximum velocity of the baseline configuration, the missile radius is increased by ΔR such that:

$$R_2 = R_{2BASELINE} + \Delta R$$
$$R_3 = R_{3BASELINE} + \Delta R$$
$$R_4 = R_{4BASELINE} + \Delta R$$

3.3.1 Constant nozzle throat area

The increase of the missile radius is first applied while maintaining the nozzle throat area constant and thus a constant propellant mass flow rate.

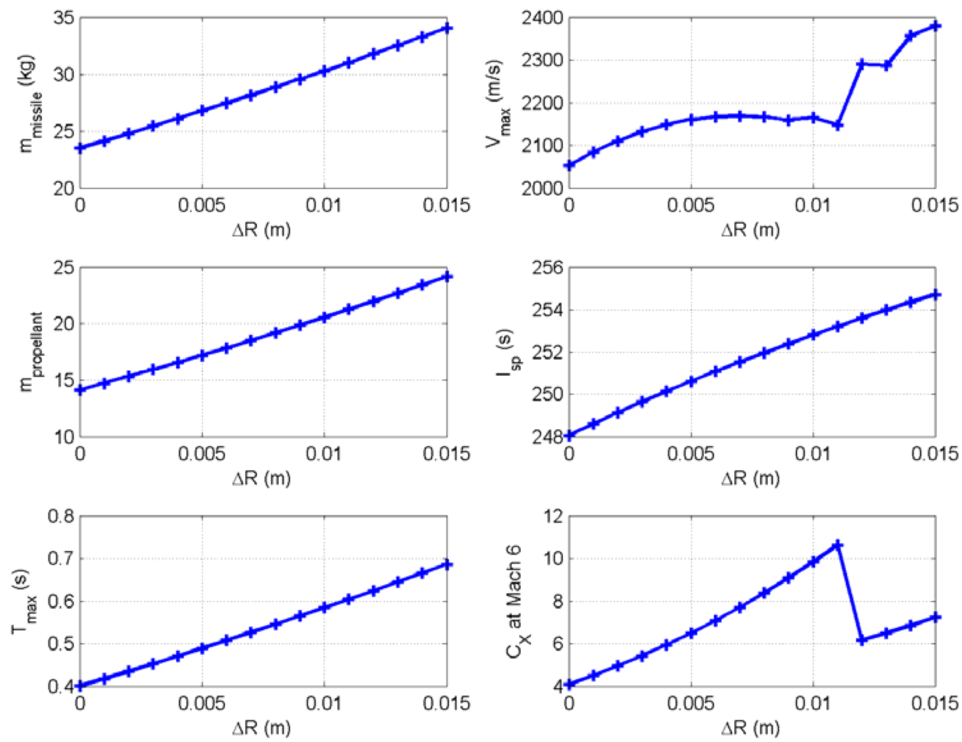

Figure 10: Sensitivity to increase in diameter at constant nozzle throat area

Increasing the missile radius increases the amount of propellant but also the missile mass and drag.

Figure 10 shows that the drag coefficient C_x at Mach 6 presents a discontinuity between ΔR =0.011m and 0.012m. This behaviour is probably due to a switch in the method used by the aerodynamic prediction code DATCOM because of the resulting low sharpness of the nose cone for large ΔR . This discontinuity does not allow comparing the results of larger ΔR and these values are thus considered outliers.

The velocity plot shows that the velocity reaches a peak before the aerodynamic coefficients become unreliable. The increase in missile diameter increases the mass of propellant. Since the mass flow rate is constant, it takes more time to reach maximum velocity as illustrated by the plot of T_{max} in Figure 10. The next sensitivity analysis will adjust the throat area to maintain the time required to reach maximum velocity to the target value of 0.4 s.

3.3.2 Adjusted nozzle throat area for a constant burn time

In this section, a parametric sensitivity analysis is performed to investigate the sensitivity of the missile velocity to an increase in diameter while maintaining the booster burn-time constant to 0.4 s by adjusting the nozzle throat area A_t.

For each value of ΔR , a numerical gradient based algorithm is used to vary A_t so that T_{max}= 0.4 s. Results are presented in Figure 11.

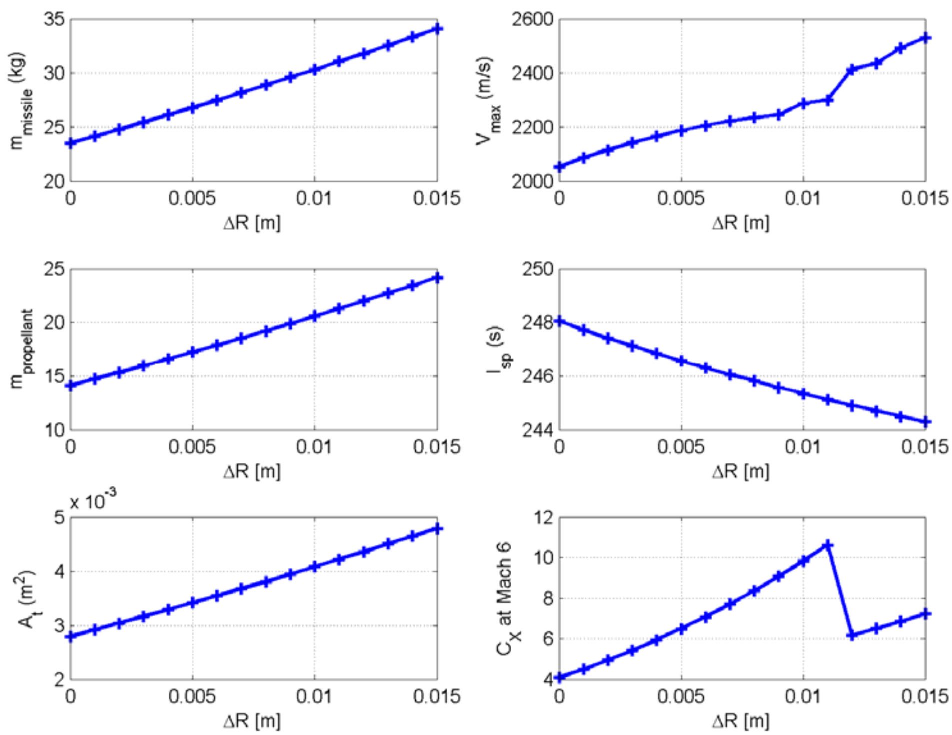

Figure 11: Sensitivity to increase in diameter at constant burn time

The results show the improved velocity performance compared to Figure 10. This illustrates that the design variables for the booster are coupled to each other and that it is important to evaluate the impact of a design change on all system variables. From now on, the nozzle throat area will be adjusted to maintain T_{max} of 0.4 s in all the next sensitivity analyses.

3.4 Sensitivity to increase in length

An option to increase the maximum velocity of the booster is to increase the length of the system instead of increasing its diameter.

To evaluate the sensitivity of the booster performance to a change in length, the length of the motor chamber is increased by increasing X_3 while maintaining the nose length X_2 and the nozzle length. All diameters were maintained.

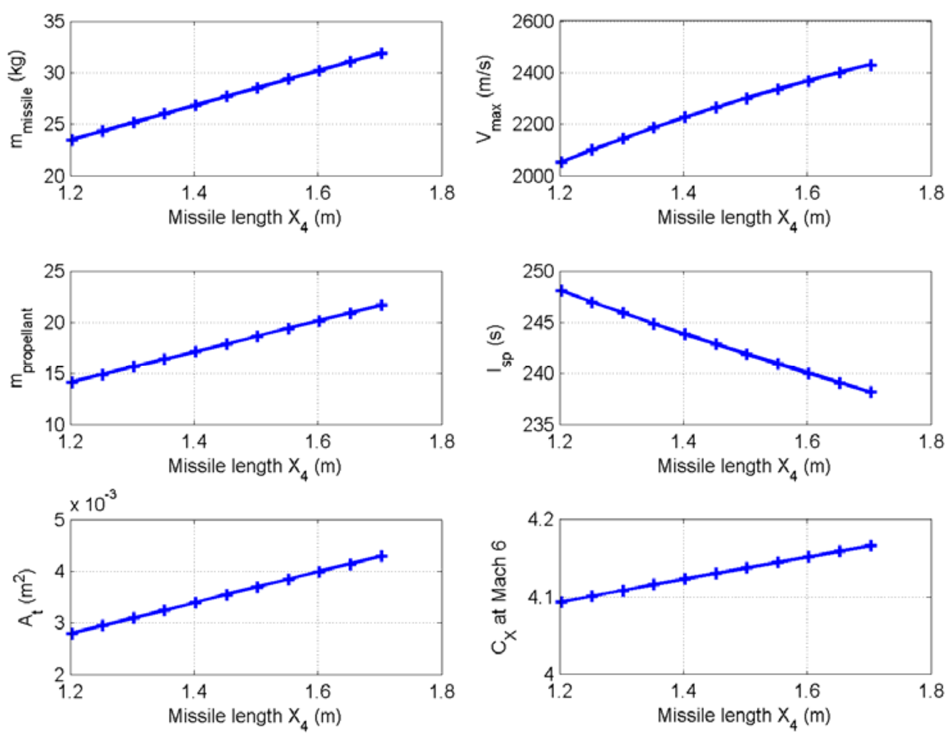

Figure 12: Sensitivity to increase in length

The results in Figure 12 show the missile characteristics as a function of total missile length X_4. Increasing the length increases the mass of propellant $m_{propellant}$ while the increase in drag is minimal. Increasing the length shows to be more efficient than increasing the radius because although both increase the amount of propellant, the increase in length is less detrimental to the drag coefficient. Increasing the length allows to reach the required speed of 2370 m/s although the missile mass climbs to 30.2 kg.

3.5 Sensitivity to motor chamber pressure

Increasing the amount of propellant to increase the velocity also increases the missile mass. To increase the velocity without increasing the mass, an option is to increase the propellant energetic efficiency by increasing its specific impulse I_{sp}. A method to increase the I_{sp} is to increase the motor chamber pressure.

To investigate the benefits of increasing the chamber pressure, a parametric analysis is performed on that variable. Results are presented in Figure 13.

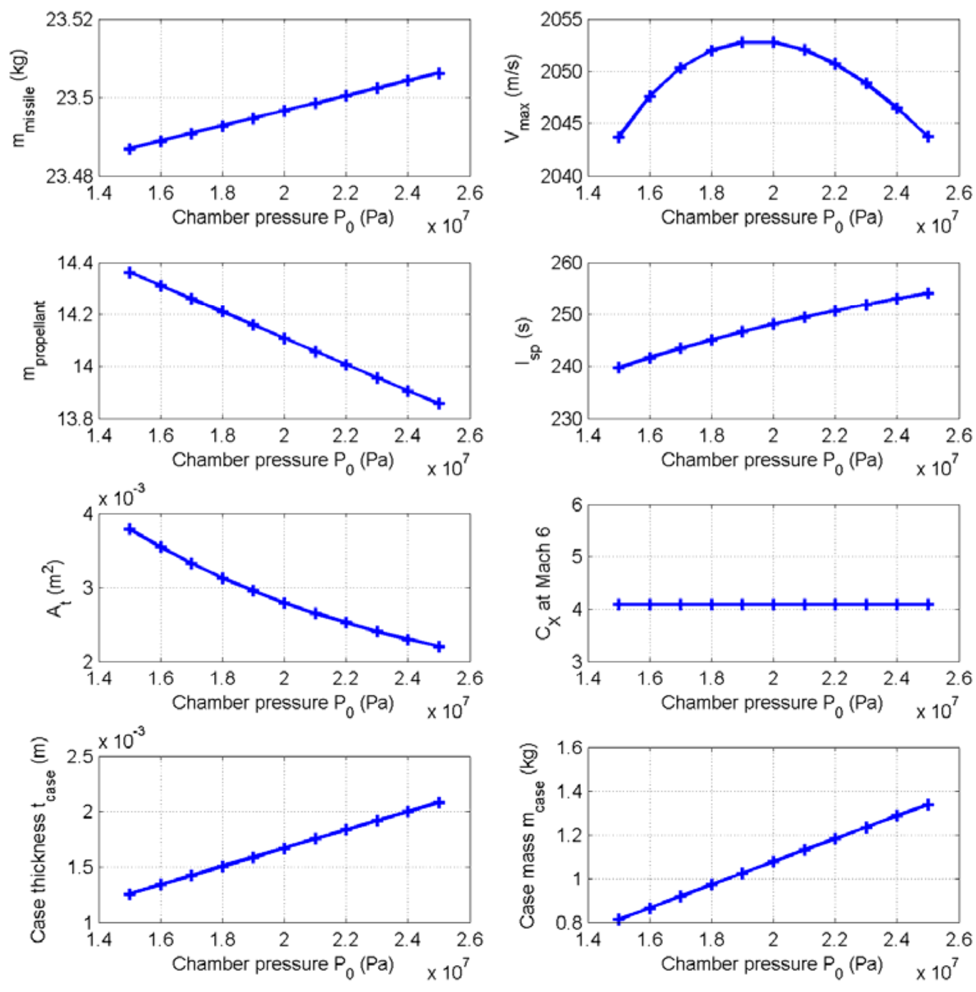

Figure 13: Sensitivity to motor chamber pressure

The results clearly show that increasing the chamber pressure p_0 increases the propellant specific impulse. However, in order to survive to the increase in pressure, the motor casing wall thickness t_{case} must also be increased. Increasing the wall thickness while maintaining the same missile external diameter leads to a smaller combustion chamber containing less propellant. This is shown by the reduction of propellant mass m_{prop} in Figure 13. The missile mass is almost constant as the case and propellant density are similar.

The total effect on the system is that any gain in final velocity benefit from propellant specific impulse provided by increasing the pressure is offset by the effect of the increased wall thickness required to resist to the pressure. The maximum velocity is obtained at a chamber pressure close to the baseline value of p_0= 20MPa.

3.6 Sensitivity to nozzle exit radius

This sensitivity study looks at increasing the nozzle exit area by changing it radius R_4. Results are presented in Figure 14.

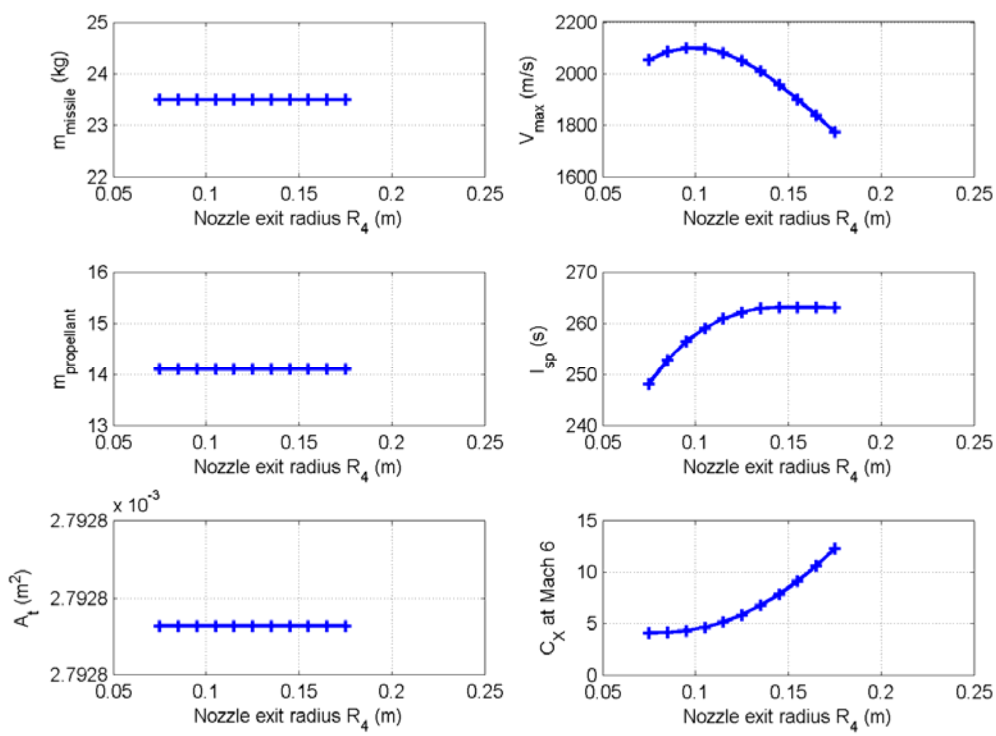

Figure 14: Sensitivity to nozzle exit radius

Increasing the nozzle exit radius increases the propellant specific impulse until complete gas expansion is achieved at radius R_4 of about 0.13 m. However, increasing the nozzle radius also increases the drag coefficient.

From a system perspective, evaluated by the maximum velocity, the optimum nozzle exit radius is R_4=0.109 m.

Increasing the nozzle radius from 0.075m to 0.109m increases the maximum velocity by 50m/s without increasing the system mass. This result is, however, somewhat questionable as the current problem formulation does not vary the nozzle mass as a function of its dimensions.

4. HEMi concept optimisation

As demonstrated in the parametric studies, achieving simultaneously the requirements of length, mass, and speed (lethality) is challenging.

Building a missile that respects the 1.25 m length constraint is straightforward. However, respecting the maximum mass is more challenging because it requires the determination and respect of a mass distribution budget for all the components. Also, given the challenging requirements, it is likely that even if the mass distribution is optimised, the speed and thus lethality may not be met.

For the optimisation of the HEMi concept, the team decided that the most important requirement to respect is the speed because it directly impacts the intended effect (lethality and response time) of the missile system.

Assuming now that the missile length is an independent variable and that velocity is the most important requirement, the objective of the design optimisation work for the HEMi booster was to determine the missile configuration that provides the lightest system while achieving speed performance and payload objectives.

The resulting optimisation problem definition is present in Table 19.

Table 19: Optimisation problem definition

Optimisation Objective:
Minimize total motor mass
Propellant + Case + Nozzle mass
Design variables:
Missile length
Effect on drag, propellant volume and weight
Missile diameter
Effect on drag, propellant volume and weight
Chamber pressure
Effect on propellant specific impulse, motor case thickness and weight
Nozzle throat area
Effect on the motor burn-rate
Constraints:
Achieve a missile velocity of 2370m/s (Mach 7)
in a range of 400m
for a given overall missile length

Figure 15 presents the missile mass of the optimised design as a function of the overall missile length selected.

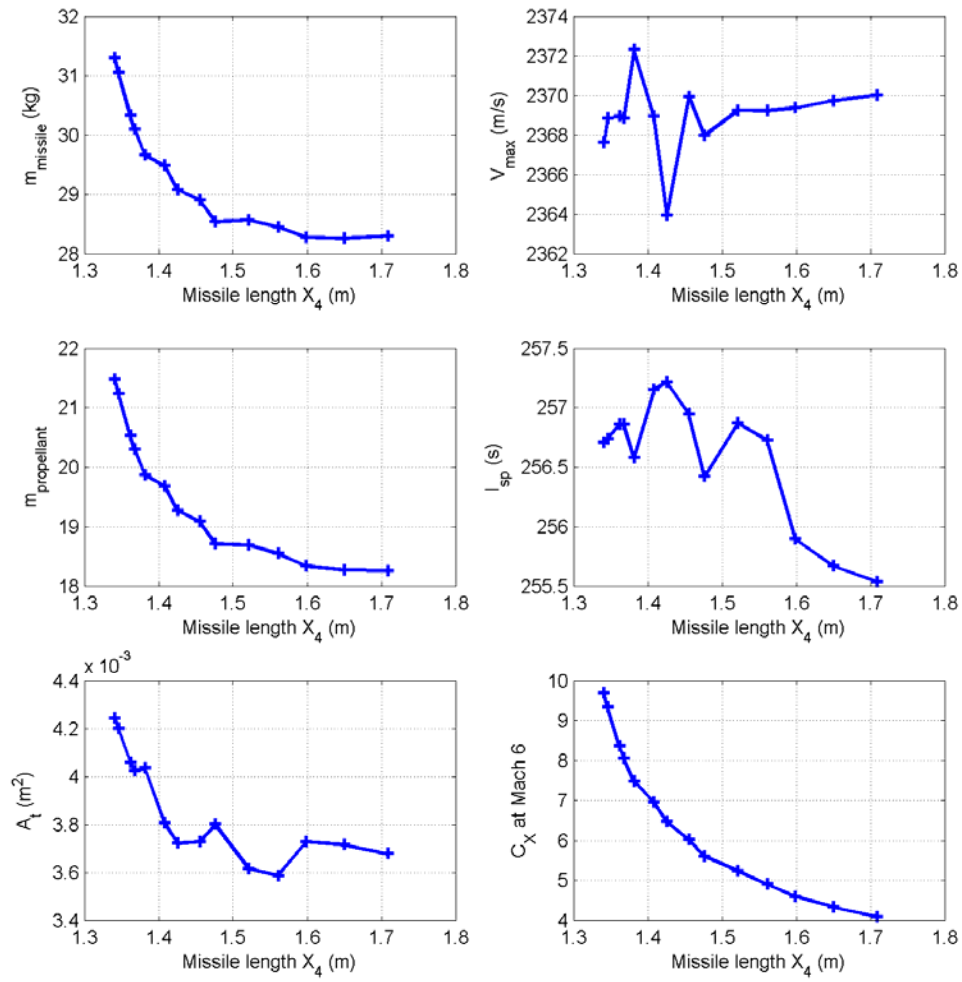

Figure 15: Design feasibility domain for HEMi

The important conclusion that can be draw from the figure is that the integrated design codes indicate the technology they represent cannot achieve the initial specification of a total missile length of 1.25, a mass of 23.0kg and a velocity of Mach 7 in 0.4s.

The current design reaches only Mach 6.03. However it is possible to reach Mach 7.0 by increasing both the length and the mass of the missile. Table 20 presents a tabular comparison of the results of Figure 15.

Table 20: Comparison of designs

Design	Length(m)	Mass(kg)	Velocity (Mach)
Targeted design (infeasible with the state of the technology captured by the integrated design codes)	1.25	23.0	7.0
Baseline	1.254	23.50	6.03
A feasible example at Mach 7.0	1.40	29.5	7.0

At the difference of parametric studies that varies the independent variables and uses discrete values, the optimisation allows to hold constant, using optimisation constraints, the values of dependant values such as velocity. This allows restricting the concept investigation to only the concepts that meet the requirements.

5. Limitations and benefits of design methods based on integrated analysis and optimisation

Missile system design requires high levels of integration among disciplines such as aerodynamics, structures, propulsion, and controls. These disciplines are mutually interactive and coupled and these effects must be fully considered in the analysis of missiles and in the prediction of system responses.

This is particularly true for HEMi since the length, weight and velocity requirements demands state-of-the-art technologies. Figure 16 presents a technology interaction matrix for HEMi. This matrix presents the factors by which the technologies interact. The matrix could present several other factors but presently contains enough information to state that all technologies are interrelated.

				Propulsion	
			Aerodynamic		- Booster size and length - Drag
		Control		- Canard design - Aero heating - Control forces and moments	- Thrust misalignment - TVC design
	Lethality		- Accuracy - Angle of penetration	- Angle of penetration - Velocity	- Velocity
Guidance		- Terminal accuracy	- Bandwidth - Update rate	- Control authority	- Smoke/plume
Structure	- Packaging -	- Bending stiffness	- Bending frequency - Volume	- Wing attachment - Separation	- Weight - Pressure - Integrity - TVC

Figure 16: Technology interaction matrix

Modeling and simulation is required to assess the integrated effect of the proposed technologies on the factors presented in the figure. The system performance model and the analysis should fully consider the interaction between disciplines and technologies.

The use of optimisation to conduct the design and analyze the system provides an automated way to tailor a conceptual weapon design to user defined requirements. The entire spectrum of the capabilities provided by the weapon can be investigated using parametric variation of the weapon performance requirements. If several weapon configurations can achieve a given performance requirement, the use of optimisation based design will allows the definition of the best configuration according to a performance criteria such as minimum take-off weight, minimum flight time or minimum cost.

However, when model complexity is augmented, the direct optimisation of a missile system including disciplines such as aerodynamics, structures, propulsion, and controls become unpractical due to excessive computational cost and complexity. Complex model optimisation methodologies based on multidisciplinary hierarchical optimisation can be investigated to improve the performance of the optimisation algorithm by taking advantage of the problem decomposition into subsystems. The most promising method, Bi-Level Integrated System Synthesis (BLISS), divides the optimisation problem into two levels: system (weapon) and subsystems (e.g. aerodynamics, structures, propulsion, and controls). The main system optimisation coordinates the subsystem optimisations to ensure that subsystems converge to global system optimum.

The capability to perform optimisation of complex models will deliver not only more accurate modelling and simulation results but also define the system that makes the best use of enabling technologies such as hypersonic weapons.

Since technologies are tightly integrated and coupled, an individual technology improvement can be found detrimental to the system performance if coupled subsystems are not adapted to the improved technologies. Additionally, a potential subsystem improvement may be detrimental to the weapon system performance once its impact on coupled subsystems is assessed.

Integrated analysis methodology can provide a fair comparison of the impact of technologies on the system performance by ensuring that they have been properly integrated with other subsystems. Sensitivity analysis of system performance to subsystem parameters can also be performed.

This allows adequate technology evaluation to guide its development in the most promising directions.

6. Conclusion

The equations governing the kinematic performance of the missile were coupled with an aerodynamic prediction tool, a structural design module and a solid rocket motor performance model in order to analyze the performance of the baseline configuration of HEMi.

Following the results obtained from the baseline configuration, several parametric analyses were performed to investigate the design sensitivity and to develop a method of improving the performance of the design.

Results of variations in mass, radius, length, motor chamber pressure and nozzle exit radius are presented. Analysis of the design space using parametric analysis showed that simultaneously achieving the length, mass, and speed (lethality) requirements is challenging. To drive the design to a concept that meets requirements, an optimization algorithm was used to adjust independent variables while keeping dependent values like velocity constant. This restricts the concept investigation to concepts that meet the requirements.

The integrated analysis tool indicates that the current HEMi configuration cannot achieve the initial specification of total missile length 1.25 m, mass 23.0 kg and velocity Mach 7 in 0.4 s. However, it is possible to reach Mach 7.0 if both the length and the mass of the missile are increased.

7. References

1. "High Energy Missile Technology Demonstration – Phase I (HEMi TD)", Synopsis Sheet Effective Technology Demonstration Project Approval, Version 1.5, Project Sponsor: DRDC 30 May 2001.

2. Lestage, R., "Résultats préliminaires de l'étude des configurations HEMi", Note technique du RDDC Valcartier TN 2003-199, Juillet 2004.

3. Lestage, R., Dubois, J., Lafrance, P., Wong, F., Lesage, F., "HEMi missile system level specifications, V.2.01", DRDC Valcartier Technical Note TN 2003-200, June 2004.

4. Lesage, F., Hamel, N., "Sizing of the initial HEMi missile concept using semi-empirical aerodynamics analysis", DRDC Valcartier Technical Memorandum TM 2003-140, December 2003.

5. Wong, F.C. "Preliminary Analysis of an Overwrapped Booster Case for the High Energy Missile (HEMi)". DRDC Valcartier Technical Memorandum TM 2003-150, October 2003.

6. Lesage, F., "Wind Tunnel test results for validation of HEMi initial configuration". DRDC Valcartier Technical Memorandum TM 2003-212, December 2003.

7. Blake, W.B., "Missile DATCOM User's Manual – 1996 Fortran 90 Revision", AFRL-VA-WP-TR-1998-3009, Air Force Research Laboratory, Wright-Patterson Air Force Base Ohio, USA, February 1998.

Annex 1: Aerodynamic Modelling using DATCOM

An input file template for DATCOM program is present here. In the template, variables within the following tag |> <| are replaced by actual values for evaluation of the aerodynamic properties of the resulting configuration by DATCOM.

```
$FLTCON NALPHA  =         4.00000,
        ALPHA   =         0.00000,0.25000,0.50000,1.00000,
        NMACH   =         14.00000,
        MACH    =
0.10000,0.50000,1.40000,1.50000,1.75000, 2.00000,2.50000,
        MACH(8) =
3.00000,3.50000,4.00000,5.00000,6.00000,7.00000,8.00000,
        ALT     =
0.00000,0.00000,0.00000,0.00000,0.00000,0.00000,0.00000,
        ALT(8)  =
0.00000,0.00000,0.00000,0.00000,0.00000,0.00000,0.00000,
        $END
$REFQ   XCG     =         0.00000,
        ZCG     =         0.00000,
        LREF    =         0.05000,
        SREF    =         0.00196,
        BLAYER  =         1.00000,
        SCALE   =         1.00000,
        $END
$AXIBOD XO      =         0.00000,
        NX      =         5.00000,
        X       =         |>X<|
        R       =         |>R<|
        DISCON  =         2.00000,3.00000,4.00000,
        $END
SOSE
BUILD
PART
PLOT
DIM M
DAMP
DERIV RAD
CASEID Booster without fin; HEMI 2.1 Iteration # XX
SAVE
NEXT CASE
```

Annex 2: Nozzle length computation

The following Excel spreadsheet presents the computation of the nozzle length. It adapts the pintle nozzle areas based on a baseline conical nozzle.

	A	B	C	D	E
1		Pintle Nozzle Geometry			
2		Ae/At	At	Dc	
3			(m2)	(mm)	
4					
5	Reference case	5.65501662302217	0.002969	100.4	
6	New case	=PI()*B34^2/C6	0.0029498	100.4	
7					
8		Pintle Nozzle Geometry			
9		Throat	Di	Do	De
10		Opening			
11		(mm)	(mm)	(mm)	(mm)
12					
13		=C5/(3.1415927*D5/1000)*1000	=D5-B13	=D5+B13	=SQRT(4*C5*B5/3.1415926)*1000
14		=C6/(3.1415927*D6/1000)*1000	=D6-B14	=D6+B14	=SQRT(4*C6*B6/3.1415926)*1000
15					
16		Conical Nozzle			
17		Equivalent	Nozzle	60% Bell	
18		Throat	Length	Length of	
19		Diameter		Nozzle	
20		(mm)	(mm)	(mm)	
21					
22	Reference case	=SQRT(C5*4/3.14159)*1000	=(E13-B22)/(2*TAN(15*3.141593/180))	=C22*0.6	
23	New case	=SQRT(C6*4/3.14159)*1000	=(B23/2)*(SQRT(B6)-1)/TAN(15*3.141593/180	=C23*0.6	
24					
25		Pintle	Total		
26		Nozzle	Nozzle		
27		Length	Length		
28		(mm)	(m)		
29					
30	Reference case	78	=(B30+22)/1000		
31	New case	=D23*$B$30/$D$22	=(B31+22)/1000		
32					
33					
34	Outputs	R4	X3	X4	
35		0.075293	1.15	=C35+C31	

Annex 3: Propellant Modelling

Nomenclature

F_d	=	Delivered thrust (N)
F_{cea}	=	Thrust calculated with the NASA Lewis Research Center thermochemical equilibrium code (CEA)[1] (N)
F_{an}	=	Thrust calculated with standard analytical one-dimensional relationships using the thermodynamic properties of the combustion chamber[2] (N)
Isp_d	=	Delivered specific impulse (s)
Isp_{cea}	=	Specific impulse calculated with the NASA Lewis Research Center thermochemical equilibrium code (CEA)[1] (s)
Isp_{an}	=	Specific impulse calculated with standard analytical one-dimensional relationships using the thermodynamic properties of the combustion chamber[2] (s)
$\dot{m}_d$	=	Delivered mass flow rate of combustion products (kg/s)
η_{cea}	=	Thrust based efficiency relating the analytically calculated thrust with that calculated with CEA
η_d	=	Thrust based efficiency relating the thrust calculated with CEA to that which would be delivered in a real system
ζ_{cea}	=	Specific impulse based efficiency relating the analytically calculated specific impulse with that calculated with CEA
ζ_d	=	Specific impulse based efficiency relating the specific impulse calculated with CEA to that which would be delivered in a real system
γ	=	Ratio of specific heats of the combustion products in the rocket combustion chamber
R	=	Particular gas constant of the combustion products in the rocket combustion chamber (J/Kg-K)
p_0	=	Rocket combustion chamber pressure (Pa)
T_0	=	Rocket combustion chamber temperature (K)
p_a	=	Ambient pressure (Pa)
p_e	=	Nozzle exit pressure (Pa)
A_t	=	Nozzle throat area (m^2)
A_e	=	Nozzle exit area (m^2)
α	=	Nozzle exit cone half angle (°)
g_0	=	Gravitational acceleration (m/s^2)

In this Annex, the equations used in the HEMi missile system performance model to determine the delivered mass flow rate, thrust and specific impulse for the rocket propulsion subsystem are discussed.

For any system performance model, the aim in subsystem modeling is to approximate as closely as possible, the real subsystem performance while maintaining simplicity of expression. The simplest formulation of the mass flow rate, thrust and specific impulse for the rocket propulsion subsystem is based on the quasi one-dimensional analytical theory found in basic propulsion texts, such as that by Sutton[1]. Since these expressions are analytical they can be implemented very easily into a system performance code. However, the expressions assume single values for the thermochemical properties of the combustion gases as they expand through the nozzle and as such, can produce results that deviate significantly from reality. In addition, the values of the thermochemical properties of the gases in the combustion chamber vary with motor operating pressure, diminishing the fidelity of a parametric analysis of the missile performance over a large range of operating conditions.

A more precise approach to the calculation of the mass flow rate, thrust and specific impulse is to run a thermochemical equilibrium code such as the NASA Lewis code (CEA)[2], for every missile system performance simulation. This remedies the problem of invariant thermochemical properties inherent in the analytical approach. However, it implies the linking of the thermochemical equilibrium code to the system performance code.

The approach taken in the present work was to use the basic analytical formulations adjusted with expressions for efficiencies and thermochemical parameters based on CEA simulations. The values of the actual delivered mass flow rate, thrust and specific impulse were then calculated by applying additional standard efficiencies[1].

The chemical formulation used in the CEA calculations was for a standard aluminized Hydroxl Terminated Polybutadiene / Ammonia Perchlorate (HTPB/AP) propellant. Given that the actual HEMi propellant was not yet developed, the aluminized HTPB/AP formulation was chosen as being representative of the level of chemical energy that would be delivered by the actual HEMi propellant.

The efficiencies relating the thrust and specific impulse from the CEA and analytical formulations are given in equations 1 and 2. The values used in the missile system simulations were 0.9966 and 0.954 respectively. These represent the average relationship of the CEA and analytical results calculated over a large range of motor operating conditions.

$$\eta_{cea} = \frac{F_{cea}}{F_{an}} \tag{1}$$

$$\zeta_{cea} = \frac{Isp_{cea}}{Isp_{an}} \tag{2}$$

The efficiencies relating the thrust and specific impulse from the CEA formulation to the actual delivered values are given in equations 3 and 4. A value of 0.920 was used in the missile system simulations for both efficiencies.

$$\eta_{d} = \frac{F_{d}}{F_{cea}} \tag{3}$$

$$\zeta_{d} = \frac{Isp_{d}}{Isp_{cea}} \tag{4}$$

The analytical expressions for the temperature, ratio of specific heat and particular gas constant in the motor chamber are given in equations 5, 6, and 7 respectively as a function of the motor operating pressure. The constants a_T, b_T, a_γ, b_γ, a_R and b_R were derived from curve fitting the results from CEA calculations. The values of these constants used in the missile simulations were 3319.81, 84.55, 1.123, 0.0036, 319.30, and -3.096 respectively.

$$T_0 = a_T + b_T \ln\left(\frac{p_0}{1 x 10^6}\right) \tag{5}$$

$$\gamma = a_\gamma + b_\gamma \ln\left(\frac{p_0}{1 x 10^6}\right) \tag{6}$$

$$R = a_R + b_R \ln\left(\frac{p_0}{1 x 10^6}\right) \tag{7}$$

The expressions for delivered mass flow rate, thrust and specific impulse used in the missile simulations are given in equations 8, 9 and 13 respectively. The value for the nozzle exit cone half angle, α, used in the missile simulations was 7.5º.

$$\dot{m}_d = \frac{\eta_d}{\zeta_d}\frac{\eta_{cea}}{\zeta_{cea}} p_0 A_t \sqrt{\frac{\gamma}{R\,T_0}}\left(\frac{2}{\gamma+1}\right)^{\frac{\gamma+1}{2(\gamma-1)}} \tag{8}$$

$$F_d = \eta_d \eta_{cea}\left\{\lambda\; p_0 A_t \sqrt{\left(\frac{2\gamma^2}{\gamma-1}\right)\left(\frac{2}{\gamma+1}\right)^{\left(\frac{\gamma+1}{\gamma-1}\right)}\left(1-\left(\frac{p_e}{p_0}\right)^{\left(\frac{\gamma-1}{\gamma}\right)}\right)} + \left(p_e - p_a\right)A_e\right\} \tag{9}$$

where

$$\frac{A_e}{A_t} = \frac{\Gamma}{\left(\frac{p_e}{p_0}\right)^{\frac{1}{\gamma}}\sqrt{1-\left(\frac{p_e}{p_0}\right)^{\left(\frac{\gamma-1}{\gamma}\right)}}} \tag{10}$$

$$\Gamma = \left(\frac{\gamma-1}{2}\right)^{\frac{1}{2}}\left(\frac{2}{\gamma+1}\right)^{\left(\frac{\gamma+1}{2(\gamma-1)}\right)} \tag{11}$$

$$\lambda = \frac{1}{2}(1+\cos\alpha) \tag{12}$$

$$Isp_d = \frac{F_d}{\dot{m}_d g_0} \tag{13}$$

References

1. Sutton, George, P. and Biblarz, Oscar, “Rocket Propulsion Elements”, John Wiley and Sons, 2001, Seventh Edition
2. Gordon, S., and McBride, B.J., “Computer Program for Calculation of Complex Chemical Equilibrium Compositions and Applications”, NASA RP 1311, Oct. 1994

Annex 4: Baseline results

Variable name	Value	Units	Description
Model.ComputePintleNozzleLenght.R4	0.075	m	Nozzle exit radius.
Model.ComputePintleNozzleLenght.X3	1.15	m	3rd station relative to nose tip
Model.ComputePintleNozzleLenght.X4	1.253766	m	Booster length
Model.ComputePintleNozzleLenght.At	0.00294	m^2	Nozzle throat area
Model.ComputePintleNozzleLenght.Lnozzle	0.103766	m	Nozzle length
Model.HEMiBoosterDatcom.X1	0.1	m	First station relative to nose tip
Model.HEMiBoosterDatcom.X2	0.16	m	2nd station relative to nose tip
Model.HEMiBoosterDatcom.X3	1.15	m	3rd station relative to nose tip
Model.HEMiBoosterDatcom.X4	1.253766	m	4th station relative to nose tip
Model.HEMiBoosterDatcom.R1	0.025	m	Body radius at corresponding station position.
Model.HEMiBoosterDatcom.R2	0.06	m	Body radius at corresponding station position.
Model.HEMiBoosterDatcom.R3	0.073	m	Body radius at corresponding station position.
Model.HEMiBoosterDatcom.R4	0.075	m	Body radius at corresponding station position.
Model.HEMiBoosterDatcom.M14	8		Mach numbers for which drag coefficient is provided.
Model.HEMiBoosterDatcom.M13	7		Mach numbers for which drag coefficient is provided.
Model.HEMiBoosterDatcom.M12	6		Mach numbers for which drag coefficient is provided.
Model.HEMiBoosterDatcom.M11	5		Mach numbers for which drag coefficient is provided.
Model.HEMiBoosterDatcom.M10	4		Mach numbers for which drag coefficient is provided.
Model.HEMiBoosterDatcom.M9	3.5		Mach numbers for which drag coefficient is provided.
Model.HEMiBoosterDatcom.M8	3		Mach numbers for which drag coefficient is provided.
Model.HEMiBoosterDatcom.M7	2.5		Mach numbers for which drag coefficient is provided.
Model.HEMiBoosterDatcom.M6	2		Mach numbers for which drag coefficient is provided.
Model.HEMiBoosterDatcom.M5	1.75		Mach numbers for which drag coefficient is provided.
Model.HEMiBoosterDatcom.M4	1.5		Mach numbers for which drag coefficient is provided.
Model.HEMiBoosterDatcom.M3	1.4		Mach numbers for which drag coefficient is provided.
Model.HEMiBoosterDatcom.M2	0.5		Mach numbers for which drag coefficient is provided.
Model.HEMiBoosterDatcom.M1	0.1		Mach numbers for which drag coefficient is provided.
Model.HEMiBoosterDatcom.LREF	0.05		Reference diameter
Model.HEMiBoosterDatcom.CA14	4.7316		Drag coefficient corresponding to the Mach numbers
Model.HEMiBoosterDatcom.CA13	4.4094		Drag coefficient corresponding to the Mach numbers
Model.HEMiBoosterDatcom.CA12	4.1012		Drag coefficient corresponding to the Mach numbers
Model.HEMiBoosterDatcom.CA11	3.8405		Drag coefficient corresponding to the Mach numbers
Model.HEMiBoosterDatcom.CA10	3.6859		Drag coefficient corresponding to the Mach numbers
Model.HEMiBoosterDatcom.CA9	3.5546		Drag coefficient corresponding to the Mach numbers
Model.HEMiBoosterDatcom.CA8	3.4626		Drag coefficient corresponding to the Mach numbers
Model.HEMiBoosterDatcom.CA7	3.5288		Drag coefficient corresponding to the Mach numbers
Model.HEMiBoosterDatcom.CA6	3.7125		Drag coefficient corresponding to the Mach numbers
Model.HEMiBoosterDatcom.CA5	3.7736		Drag coefficient corresponding to the Mach numbers
Model.HEMiBoosterDatcom.CA4	4.2679		Drag coefficient corresponding to the Mach numbers
Model.HEMiBoosterDatcom.CA3	4.4715		Drag coefficient corresponding to the Mach numbers
Model.HEMiBoosterDatcom.CA2	0.7506		Drag coefficient corresponding to the Mach numbers
Model.HEMiBoosterDatcom.CA1	0.8241		Drag coefficient corresponding to the Mach numbers
Model.Propellant.R4	0.075	m	Body radius at corresponding station position.
Model.Propellant.At	2.79E-03	m^2	Nozzle throat area
Model.Propellant.p0	20000000	Pa	Combustion Chamber pressure
Model.Propellant.Fd	85699.96	N	Motor thrust.
Model.Propellant.Mpropdot	35.21656	kg/s	Propellant mass flow rate.
Model.Propellant.Isp	248.0646	s	Propellant specific impulse.
Model.Propellant.rhoprop	1688	kg/m3	Propellant density.
Model.RetentionTube.X1	0.1	m	First station relative to nose tip
Model.RetentionTube.X3	1.09871	m	3rd station relative to nose tip
Model.RetentionTube.Rtube	0.025	m	Retention tube inner radius
Model.RetentionTube.p0	2.00E+07	Pa	Combustion Chamber pressure
Model.RetentionTube.ttube	0.0027	m	Retention tube thickness
Model.RetentionTube.mtube	0.64111	kg	Retention tube mass
Model.OuterCase.R1	0.025	m	Body radius at corresponding station position.
Model.OuterCase.R2	0.06	m	Body radius at corresponding station position.
Model.OuterCase.R3	0.073	m	Body radius at corresponding station position.
Model.OuterCase.X1	0.1	m	First station relative to nose tip
Model.OuterCase.X2	0.16	m	2nd station relative to nose tip

Model.OuterCase.X3	1.09871	m	3rd station relative to nose tip
Model.OuterCase.p0	2.00E+07	Pa	Combustion Chamber pressure
Model.OuterCase.tcase	0.00167	m	Outer case thickness
Model.OuterCase.mcase	1.07773	kg	Outer case mass
Model.PropellantMass.R2	0.06	m	Body radius at corresponding station position.
Model.PropellantMass.R3	0.073	m	Body radius at corresponding station position.
Model.PropellantMass.X2	0.16	m	2nd station relative to nose tip
Model.PropellantMass.Rtube	0.025	m	Retention tube inner radius
Model.PropellantMass.rho	1688	kg/m3	Propellant density.
Model.PropellantMass.Tcase	0.00167	m	Outer case thickness
Model.PropellantMass.Ttube	0.0027	m	Retention tube thickness
Model.PropellantMass.mpropellant	14.1077	kg	Propellant mass
Model.PropellantMass.X3	1.09871	m	3rd station relative to nose tip
Model.TotalMass.mtube	0.64111	kg	Retention tube mass
Model.TotalMass.mcase	1.07773	kg	Outer case mass
Model.TotalMass.mpropellant	14.1077	kg	Propellant mass
Model.TotalMass.Mother	7.67	kg	Other mass
Model.TotalMass.mmissile	23.4966	kg	Missile mass
Model.MissileKinematic.Fd	85700	N	Motor thrust.
Model.MissileKinematic.mdot	35.2166	kg/s	Propellant mass flow rate.
Model.MissileKinematic.mpropellant	248.065	s	Propellant specific impulse.
Model.MissileKinematic.mmissile	23.4966	kg	Missile mass
Model.MissileKinematic.Machpoints[0]	0.1		Mach numbers for which drag coefficient is provided.
Model.MissileKinematic.Machpoints[1]	0.5		Mach numbers for which drag coefficient is provided.
Model.MissileKinematic.Machpoints[2]	1.4		Mach numbers for which drag coefficient is provided.
Model.MissileKinematic.Machpoints[3]	1.5		Mach numbers for which drag coefficient is provided.
Model.MissileKinematic.Machpoints[4]	1.75		Mach numbers for which drag coefficient is provided.
Model.MissileKinematic.Machpoints[5]	2		Mach numbers for which drag coefficient is provided.
Model.MissileKinematic.Machpoints[6]	2.5		Mach numbers for which drag coefficient is provided.
Model.MissileKinematic.Machpoints[7]	3		Mach numbers for which drag coefficient is provided.
Model.MissileKinematic.Machpoints[8]	3.5		Mach numbers for which drag coefficient is provided.
Model.MissileKinematic.Machpoints[9]	4		Mach numbers for which drag coefficient is provided.
Model.MissileKinematic.Machpoints[10]	5		Mach numbers for which drag coefficient is provided.
Model.MissileKinematic.Machpoints[11]	6		Mach numbers for which drag coefficient is provided.
Model.MissileKinematic.Machpoints[12]	7		Mach numbers for which drag coefficient is provided.
Model.MissileKinematic.Machpoints[13]	8		Mach numbers for which drag coefficient is provided.
Model.MissileKinematic.CA[0]	0.7951		Drag coefficient corresponding to the Mach numbers
Model.MissileKinematic.CA[1]	0.7303		Drag coefficient corresponding to the Mach numbers
Model.MissileKinematic.CA[2]	4.4514		Drag coefficient corresponding to the Mach numbers
Model.MissileKinematic.CA[3]	4.2484		Drag coefficient corresponding to the Mach numbers
Model.MissileKinematic.CA[4]	3.7482		Drag coefficient corresponding to the Mach numbers
Model.MissileKinematic.CA[5]	3.6906		Drag coefficient corresponding to the Mach numbers
Model.MissileKinematic.CA[6]	3.5107		Drag coefficient corresponding to the Mach numbers
Model.MissileKinematic.CA[7]	3.447		Drag coefficient corresponding to the Mach numbers
Model.MissileKinematic.CA[8]	3.5414		Drag coefficient corresponding to the Mach numbers
Model.MissileKinematic.CA[9]	3.6736		Drag coefficient corresponding to the Mach numbers
Model.MissileKinematic.CA[10]	3.831		Drag coefficient corresponding to the Mach numbers
Model.MissileKinematic.CA[11]	4.0936		Drag coefficient corresponding to the Mach numbers
Model.MissileKinematic.CA[12]	4.4032		Drag coefficient corresponding to the Mach numbers
Model.MissileKinematic.CA[13]	4.7263		Drag coefficient corresponding to the Mach numbers
Model.MissileKinematic.LREF	0.05	m	Reference diameter
Model.MissileKinematic.Vmax	2052.74	m/s	Missile maximum velocity at end of boost
Model.MissileKinematic.Tmax	0.4006	s	Missile flight time during boost phase
Model.MissileKinematic.Xvmax	366.406	m	Missile range at end of boost phase

List of symbols/abbreviations/acronyms/initialisms

α	Nozzle cone angle
σ	Yield stress
ε	Breaking strain
υ	Poisson ratio
ρ	Density
ΔR	Missile radius increment
1DOF	One degree of freedom
A_t	Nozzle throat area
BLISS	Bi-Level Integrated System Synthesis
$\overline{C}_{X0}$	Vector of drag coefficient corresponding to the Mach numbers $\overline{M}$
DND	Department of National Defence
E	Modulus
F_d	Motor thrust.
G&C	Guidance and control
HEMi TD	High Energy Missile Technology Demontrator
I_{sp}	Propellant specific impulse.
KE	Kinetic energy
L_{Nozzle}	Nozzle length
L_{REF}	Reference diameter used for computation of reference surface for drag force computation.

$\overline{M}$	Vector of Mach numbers for which drag coefficient $\overline{C}_{X0}$ is provided.
MBT	Main battle tank
m	Mass
$\dot{m}$	Mass flow rate.
P_o	Combustion Chamber pressure
R	Radius
t	Thickness
T_{max}	Missile flight time during boost phase
TVC	Thrust vector control
V_{max}	Missile maximum velocity at end of boost
X	Station position from nose tip along X-axis
X_{max}	Missile range at end of boost phase

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The objective of the design work for the HEMi booster was to determine the missile configuration that provides the lightest system while achieving performance and payload objectives. The equations governing the kinematic performance of the missile were integrated into an aerodynamic prediction tool, a structural design module and a solid rocket motor performance model in order to perform an integrated analysis of the performance of the HEMi booster in its baseline configuration. Following the results obtained from the baseline configuration, several parametric analyses were performed to investigate design sensitivity and to develop a method of improving the performance of the design. Results of variations in mass, radius, length, motor chamber pressure and nozzle exit radius are presented. Analysis of the design space using parametric analysis showed that simultaneously achieving the length, mass and speed (lethality) requirements is challenging. To drive the design to a concept that meets these requirements, an optimization algorithm was used to adjust independent variables while keeping dependent values like velocity constant. This restricted the concept investigation to concepts that met the requirements. The integrated analysis tool indicated that the current HEMi configuration cannot achieve the initial specification of total missile length 1.25 m, mass 23.0 kg and velocity Mach 7 in 0.4 s. However, it is possible to reach Mach 7.0 if both the length and the mass of the missile are increased.

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