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NUMERICAL MODEL FOR ANALYSIS AND SPECIFICATION OF A RAMJET PROPELLED ARTILLERY PROJECTILE

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Somchem of South Africa is investigating the viability of significantly increasing the range performance of a 155 mm artillery round by employing ramjet or ram rocket propulsion. A numerical baseline is defined for trajectory simulation of an artillery projectile featuring generic ramjet propulsion. Existing simulation codes [1] were modified to suit the artillery application and to present the input methodology in a way that allows flexibility regarding data types (theoretical, estimates or experimental). Analysis is performed on a spin-stabilized projectile featuring an axi-symmetrical inlet at the front, with the air duct passing through an annular warhead to the rear-mounted ram engine. The low drag figures associated with such a layout, combined with the inherently high efficiency of the ramjet, result in a ballistic range in excess of 75 km.

TRAJECTORY MODEL

The trajectory model is principally a time domain point mass model with a forces model subroutine to calculate the ram forces at each time step. The design of the code strives to simulate at the projectile system level. It employs sub-system performance data generated by separate simulation or testing. No sub-system simulation (inlet, drag, combustion) is integrated into the higher level model. To this end, the relevant sub-system performances are defined in terms of characteristics that can readily be assimilated numerically by way of tabular input or curve fitting methods. A variety of inlet and combustion options can be investigated by changing only estimated performance inputs and geometrical reference parameters. This generality is required because, at the start of the investigation, it is not known what propulsion type would eventually be best suited to the application (fuel type, gas generator, solid fuel ramjet or other).

Conversely, ideal sub-system parameters towards a specific goal (for instance long range) can be arrived at through iterative methods, thereby setting a specification for development work.

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Combustor model

The combustor is simplified to the level of expressing delivered thrust and resulting back pressure as a function of the expected fuel flow rate, air flow rate and air temperature:

$$P_{02} = A_1 + B_1.\, m_f + C_1.\, m_f^2 + D_1.\, m_a + E_1.\, m_a^2 + F_1.T_{02} \quad (1)$$

$$F_{exit} = A_2 + B_2.\, m_f + C_2.\, m_f^2 + D_2.\, m_a + E_2.\, m_a^2 + F_2.T_{02} \quad (2)$$

With P_{02} = total pressure at the inlet dump plane, m_f = fuel mass flow rate, m_a = air mass flow rate, T_{02} = total temperature at the inlet dump plane, F_{exit} = delivered thrust at the nozzle exit.

For initial concept investigations, the combustion functional above is generated by means of a separate combustion code. Once static combustion tests have been performed [2], the output of this combustion code is corrected. Data along a possible flight envelope, with m_a and T_{02} varying with altitude and velocity, is generated to serve as the base line data set. The constants A to F are found through curve fitting methods on this data set. Fig. 1 shows that P_{02} is strongly dependent on ma and less so on mf. The influence of T_{02} is slight: the upper mesh is for T_{02} = 700 K and the lower one represents T_{02} = 590 K.

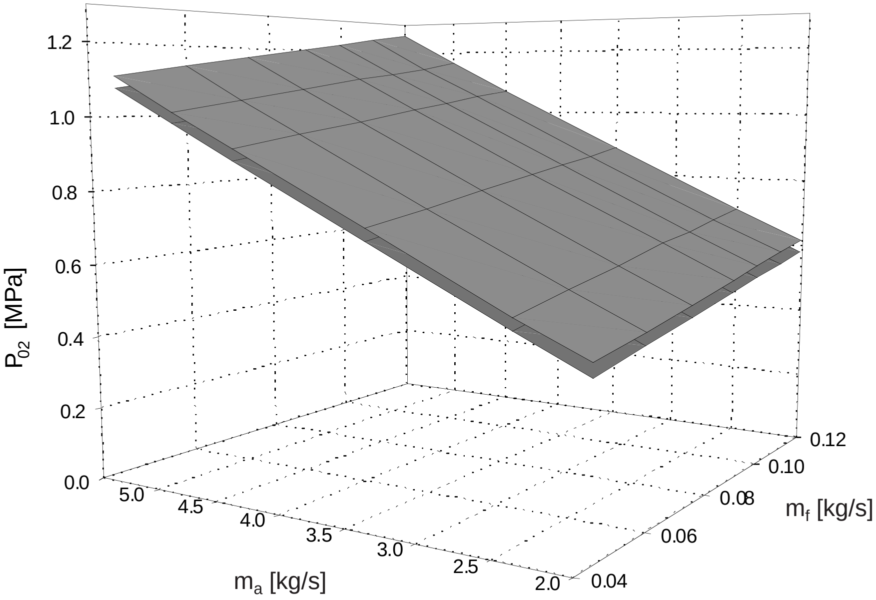

Figure 1: Curve fits to model combustor performance.

Air Inlet and drag

The inlet is characterized by its pressure recovery versus flow ratio performance at various Mach numbers. Input data reflects the maximum pressure recovery at full flow conditions, the critical point, as well as the buzz limit. These characteristics are determined by means of wind tunnel tests, where a condition of increasing back pressure is simulated by the closing of a valve at the end of the inlet duct.

The ram forces subroutine firstly calculates the inlet air flow, m_a, from the known flight conditions and the inlet supercritical flow ratio, which is a function of the Mach number. For choked gas generator applications the value of m_f is usually given as a function of time. For other applications, an estimate of the requirement can serve as a starting point. With m_a, m_f and T_{02} known, P_{02} is calculated from equation 1. If P_{02} is less than the critical pressure recovery of the inlet at that Mach number, a matched operating condition exists. F_{exit} is calculated from equation 2 and corrected according to the difference between the reference pressure of the data base line and the ambient pressure in flight. The inlet momentum thrust, F_{inl}, is obtained from:

$$F_{inl} = (P_{inl} - P_{amb})\, A_{inl} + \cos\alpha\, m_a\, V \qquad (3)$$

Equation 3 is simplified by assuming $P_{inl} = P_{amb}$, the local static air pressure, and angle of attack, α, small enough for $\cos\alpha = 1$. The net thrust is given by $F_{net} = F_{exit} – F_{inl}$. V is the flight velocity.

The external drag during ram functioning comprises of wave, friction, base and inlet additive drag components, excluding the inlet and exit momentum flux, which is accounted for in the net thrust calculation. After the ram motor has burnt out, the coast phase drag includes this momentum difference and is therefore considerably higher.

With the thrust and drag values known, a force balance is established (as shown in Fig. 2) and the trajectory calculation can be executed.

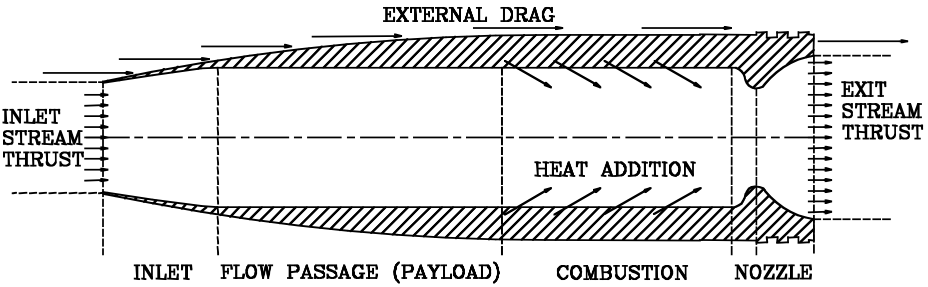

Figure 2: Axial force balance definition.

CASE STUDY

Sub-system performances are specified for a case study configuration. This projectile is spin-stabilized, has an axi-symmetrical, fixed geometry inlet and a Solid Fuel Ramjet combustion chamber. The payload is configured to allow throughflow of inlet air: either via a central or annular duct. The nominal launch velocity is 900 m/s. Inlet diameter is 84.2 mm. The basics of the layout is shown in Fig. 3.

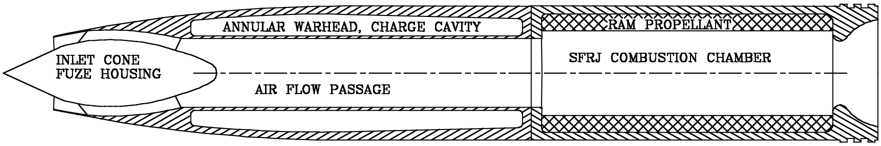

Figure 3: Case study flight configuration.

Sub-System Performances

Inlet performance at M = 2.65 is estimated as follows: The local flow conditions at the end of the external compression cone can be calculated with the Taylor-Maccoll method. For a 25° half angle cone, the pressure recovery is 0.907. With some internal compression, the flow is decelerated isentropically to M = 1.715, which is associated with a 0.85 normal shock recovery. According to guidelines by Mahoney [3], a practical subsonic diffuser can yield recoveries in the order of 0.9 in this environment. The combined recovery is 0.693. Improvement on this is possible through the use of an isentropic cone, more internal compression and a more efficient diffuser. However, performance increase is limited by viscous effects, start restrictions and length.

Drag measurements and estimations are covered under a separate paper [4]. Practical experience has taught conservatism regarding drag, therefore higher drag coefficients are used for this analysis. At M = 2.65, a ram phase value of 0.165 is used, compared with a coast phase value of 0.261. The very low ram phase figure is a result of the specific configuration, where the placement of the inlet causes 30% less frontal area for drag forces to act on. Typically, a configuration with side-mounted inlets would feature a drag coefficient of around 0.36 at this Mach number.

Combustion performance inputs were estimated from a requirement point of view. Assuming that a pure SFRJ propellant will burn too slowly for the thrust levels required, it is taken into account that the addition of oxidizer (to increase burn rate) will result in a lower specific energy. The resulting combustor is characterized by its ability to effect a stagnation temperature of 374 K to the air stream at launch conditions with fuel mass flow rate = 0.11 kg/s.

Trajectory calculations are performed, showing possible sea level range of 75 km. This scenario requires 1.65 kg of ram propellant being burnt within 20 seconds, at a specific impulse peaking at around 12 300 Ns/kg.

Considering practical combustor grain sizes (e.g. length = 0.3 m and inner diameter = 100 mm), SFRJ propellant density less than 1000 kg/m^3 and burn rates typically less than 1 mm/s, it is concluded that a redesign is warranted, opting for compositions with lower energy content.

The trajectory level results of this exercise is very similar to the first, the only difference being that more propellant is required to do the same job: a propellant mass of 1.89 kg is consumed. Specific impulse varies between 9800 and 10900 Ns/kg, as shown by Fig. 4. After some initial deceleration, the velocity is sustained until burnout at an altitude of 13.7 km. Fig. 5 indicates a fuel flow rate maintaining higher levels over the first 7 seconds, compared to the pure regressive nature of the air flow rate history. This (or even more pronounced) high flow rate is required to prevent excessive deceleration initially and is one of the major combustor design challenges for this application. The natural SFRJ trend is for m_f to follow the m_a curve.

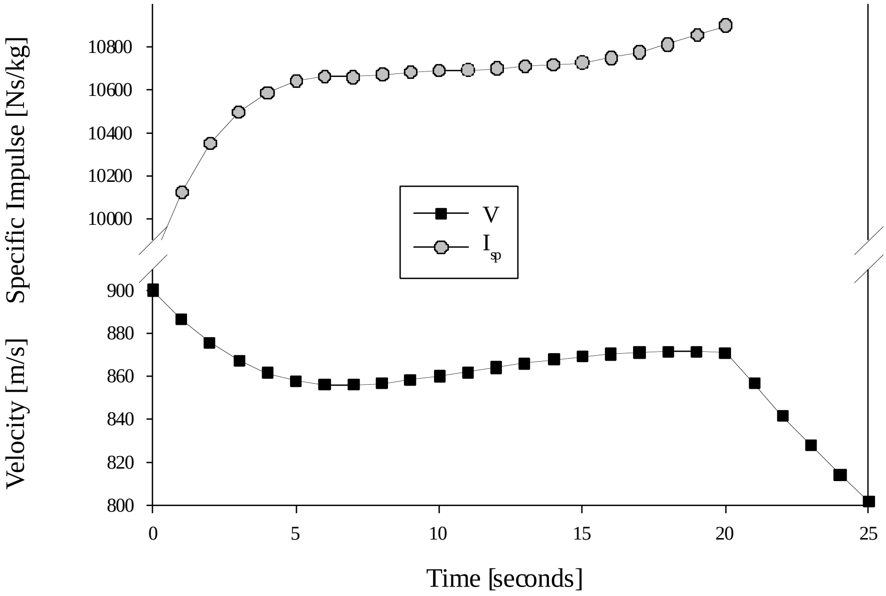

Figure 4: Variation of I_{sp} and velocity.

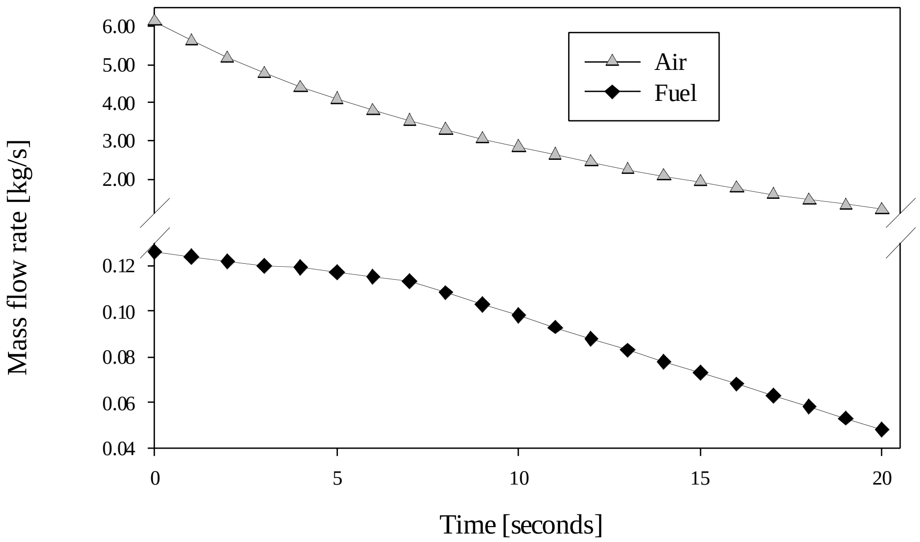

Figure 5: Variation of mass flow rates.

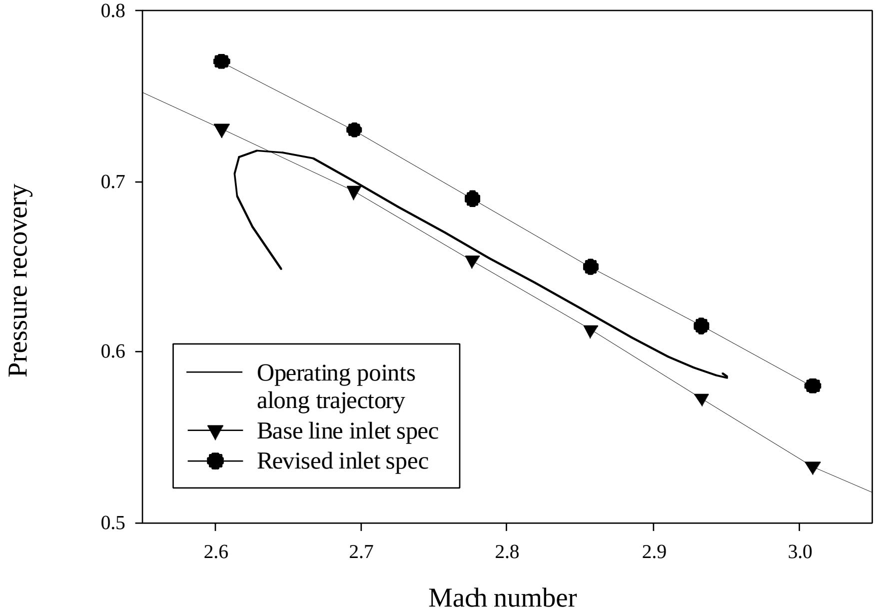

Figure 6: Inlet/combustor match.

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Plotting the combustor back pressure against Mach number for a given trajectory allows a direct comparison with inlet capability, providing insight into the matching of the components. As shown by figure 6, the base line inlet performance is transgressed. A new inlet specification is derived from this result.

DISCUSSION OF RESULTS

The numerical model has been defined in a manner that allows compatibility and uniformity between simulation, specification and experimental data. The model is accurate in as far as the input data is reliable.

Analysis results show that ramjet propulsion holds high potential for long range artillery applications, but that some serious challenges will need to be addressed.

Foremost is finding a propellant composition/geometry that will yield the the high fuel flow rates and strongly regressive nature required. Although the layout favours an SFRJ propulsion unit, the results of the above analysis lean towards the performance of a typical solid fuel gas generator motor. Other aspects of concern is the structural survivability of the propellant and ignition. Again, the SFRJ principle is not favoured by the launch velocity, yielding low stagnation temperature.

Very low drag figures can be achieved for configurations featuring an integrated payload and inlet. There is a very fine balance between thrust and drag over the duration of ram burning, with failure if the projectile decelerates to below ram optimal speeds. Variations of actual C_A (+10 or 20%) can be negated by adding more fuel. The extent to which this correction can be applied is limited. For configurations with much higher drag figures, the results of this study is invalid.

Assuming that a suitable combustor and propellant composition can be developed and that the rest of the data base is sound, extremely long ranges could be achieved with this conceptual layout. Once that goal has been achieved, accuracy and operational practicalities are more likely to determine what range requirement will be set. Against area targets, the use of a range correction fuze should improve effectiveness significantly.

The launch velocity is a critical factor. At higher velocities, the ram engine performance is better and conditions are more favourable for acceleration or sustaining good speeds up to burnout. The requirements of high acceleration and good volumetric efficiency are contradicting, resulting in very high demands on the structural design.

The ramjet artillery concept is both challenging and promising. If the various technological challenges could be solved, this concept might turn out to be the next generation 155 mm projectile.

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