Design Model of High-Performance Ramjet or Scramjet-Powered Vehicles

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A procedure is described for designing a long-range, high-speed vehicle integrating airframe and propulsion components. As envisioned here, the process involves the following steps: 1) establish performance criteria, 2) vary engine and airframe parameters to optimize performance, and 3) find a configuration that fits these desired parameters. In this paper, an approximate analysis of scramjet or ramjet engine performance is developed and coupled with appropriate airframe parameters. An example is provided to show how the procedure could be applied to the preliminary design of a scramjet-powered missile that cruises at Mach number 6.

Nomenclature

A_e = exit area
A_o = capture area
A_∞ = reference area
C = nondimensional heat addition
C_p = specific heat
C_T = thrust coefficient
D = drag
E = equivalence ratio = (fuel/air ratio)/(stoichiometric)
f_o = stoichiometric fuel/air ratio
F_8 = spillage drag
g = acceleration of gravity
H = heating value of fuel, ft-lb/slug of fuel
I = specific impulse
L = lift
M_a = Mach number at point (a) in combustor
M_b = Mach number after addition of heat
M_C = combustion Mach number
M_e = Mach number at exit
M_∞ = freestream Mach number
N = number of steps in numerical integration
P = propulsion weight fraction
P_2 = pressure at spillage
P_∞ = freestream pressure
q_∞ = freestream dynamic pressure
R = gas constant for air
T = thrust or temperature
T_∞ = freestream temperature
t = time of flight, s
W_E = weight of propulsion system
W_F = weight of fuel
W_P = weight of payload
W_S = weight of structure
W_T = total weight
γ = local ratio of specific heats
γ_e = ratio of specific heats at exit
γ_∞ = ratio of specific heats of air
δ_{av} = average flow deflection angle
δ_{inlet} = flow detection angle at cowl lip
δ_{wedge} = inlet wedge angle
η_c = combustion efficiency
η_N = nozzle efficiency

Received April 20, 1989; revision received May 3, 1990; accepted for publication May 3, 1990. This paper is declared a work of the U.S. Government and is not subject to copyright protection in the United States.

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I. Introduction

THE first step in the design of a complex system is to break it down into simpler independent components. For a rocket-powered missile, it is convenient to regard the propulsion system as a separate self-contained unit that fits into its allotted space in the airframe. However, when a long cruising range is required, an air-breathing engine becomes much more attractive than a rocket. Furthermore, the engine inlet and the airframe forebody share the same air and cannot be designed independently. Similarly, other parts of the engine, especially the exhaust nozzle, can strongly affect airframe performance.

For the airframe designer, there are many analytical and computational aids ranging from preliminary design codes to Navier-Stokes solvers. Engine design involves ignition and combustion as well as inlet and nozzle performance. Methods are available to address these problems.

However, in the early stages of the design process, where the configuration is being defined, the designer must juggle geometrical flight and performance parameters, and needs to relate thrust requirements to engine size and air-handling capacity. Thus, the purpose of this article is to provide the missile airframe designer with a means of assessing the effects of propulsion-related parameters on the performance of the vehicle. Then, with a few simple calculations, the designer can see how the engine and the airframe might be synergistically combined.

Application of the first and second laws of thermodynamics to the flight system, for example, can provide valuable design guidance. Investigations of this type, as summarized in Ref. 1, are particularly valuable in relating system performance to component efficiencies. Furthermore, by introducing an approximate analysis of Brayton cycle combustion, the performance can be related to some engine, vehicle, and flight parameters.

In the current paper, a different approach is employed to give the conceptual designer a feel for the influence of the engine. The propulsion system is represented by a simplified model that retains the fundamental physics of the process. Furthermore, the results are given in terms of easily identified parameters such as fuel/air ratio and combustion Mach number, so that the most sensitive variables and significant trends are made evident. Considerable skill may be required, however, to actually achieve the values assumed for these variables.

The equations given in the report can be readily programmed for numerical computation. Such a program is available in Ref. 2. Furthermore, the computations are brief enough to be incorporated into other airframe design codes.

To illustrate the procedure, calculations are carried out for a scramjet-powered vehicle, although the method of analysis and some of the results are also applicable to ramjets. These calculations illustrate, for example, the significance of combustion temperature in determining engine performance. The vehicle in this example could be a long-range hypersonic missile or airplane. Here it will be taken to be a missile that achieves its long range by cruising at constant speed and altitude.

In order to make intelligent decisions, it is necessary to have a performance criterion. Furthermore, a tradeoff of advantages against disadvantages requires that they be measured in the same units. Therefore, the next section of the article establishes procedures for expressing effects of different parameters in terms of the same measures, and criteria for evaluating and comparing performance.

The next requirement is for a means of evaluating the effects of various design parameters on engine performance. Therefore, a generic scramjet-engine model is defined and its internal flow is analyzed approximately so that effects of available design parameters can be compared. Results of the analysis are then described leading to some suggestions for the design of a high-performance configuration.

II. Performance Criteria

A. Minimum Weight

Somewhat arbitrarily, minimum total weight will be the basis of comparison of different configurations that satisfy the range, velocity, payload, and other requirements and constraints. For a missile, the weight can be conveniently subdivided into the following components:

$$W_T = W_P + W_S + W_F + W_E$$

where W_P is fixed by the mission and is independent of missile size: guidance and warhead, for example; W_S is the part of the total weight that scales with missile size; W_F is the weight of expendables; and W_E is the part of the total weight that scales with engine thrust.

Normalized by the total weight, fractions are related by

$$1 = \frac{W_P}{W_T} + \frac{W_S}{W_T} + \frac{W_F}{W_T} + \frac{W_E}{W_T}$$

By definition, the payload weight and the structural weight fraction are fixed. Certainly the structural weight fraction will be influenced by engine and airframe performance parameters, but for simplicity, the structural weight fraction will be assumed to be a given constant. Thus, the configuration of minimum weight for given payload will be one which minimizes

$$P = \frac{W_F}{W_T} + \frac{W_E}{W_T}$$

B. Performance Parameters

From the formula for range with constant flight conditions (an approximation),

$$\frac{W_F}{W_T} = 1 - \exp\left[-\frac{t}{I(L/D)}\right]$$

where L/D is the lift/drag ratio of the vehicle.

The engine weight fraction can be expressed as

$$\frac{W_E}{W_T} = \frac{1}{(T/W_E)(L/D)}$$

The lift/drag ratio is assumed to be constant throughout the flight. Thus, with this approximation, optimum performance is defined by the minimum value of the propulsion weight parameter P:

$$P = 1 - \exp\left[-\frac{t}{I(L/D)}\right] + \frac{1}{(T/W_E)(L/D)}$$

The thrust, in turn, is expressible in terms of a thrust coefficient:

$$T = C_T \times q_\infty A_\infty$$

where q_∞ is assumed constant and A_∞ is the area normal to freestream enclosed by the cowl lip. Then

$$P = 1 - \exp\left[-\frac{t}{I(L/D)}\right] + \frac{1}{C_T(L/D)(q_\infty A_\infty/W_E)}$$

The design parameters therefore are listed in Table 1.

Three of these quantities ($I, C_T, L/D$) are influenced directly by the selection of engine and airframe design parameters. In general, the higher the values of these quantities, the better will be the vehicle performance as measured by the weight required to carry a given payload for a specified period of time at a prescribed velocity.

Since the flow over part of the body of a missile with airbreathing propulsion is swallowed by the engine, there is some ambiguity in how much drag is charged to the airframe and how much to the engine. In the final analysis, it makes no difference how the drag is apportioned as long as all components are included just once.

III. Analytical Model of Internal Flow

A. Component Breakdown

To analyze its internal flow, a missile configuration is first subdivided into the components shown schematically in Fig. 1.

The geometry is assumed to be two-dimensional, and is analyzed by two-dimensional methods up to the lip of the inlet cowl. The remaining internal flow is calculated by one-dimensional flow. The analysis is documented in more detail in Ref. 2.

Inlet Wedge

The flow over the two-dimensional wedge angle is calculated by oblique-shock theory (Ref. 3, for example).

Isentropic Compression

The wedge flow is then compressed isentropically to a specified inlet Mach number. The compression is calculated by Prandtl-Meyer formulas, also given in Ref. 3.

Turn

The flow is then turned toward the combustor. An oblique shock appears if the inside of the cowl lip is not aligned with the flow approaching the inlet. As long as it is expanding, the internal flow from the turn to the combustor intake is nearly isentropic but a total pressure decrease can be specified to account for shock waves, friction, flow nonuniformity, or other losses up to the duct.

Table 1 Design parameters

Engine-related	
	Specific impulse I: increasing $I \rightarrow$ less fuel
	Thrust coefficient C_T: increasing $C_T \rightarrow$ smaller engine
	Weight parameter $\frac{W_E}{q_\infty A_\infty}$: decreasing $\frac{W_E}{q_\infty A_\infty} \rightarrow$ lighter engine
Mission-related	
	Time of flight t: increasing $t \rightarrow$ more fuel
Airframe-related	
	Lift/drag ratio L/D: increasing $L/D \rightarrow$ less fuel, lighter engine

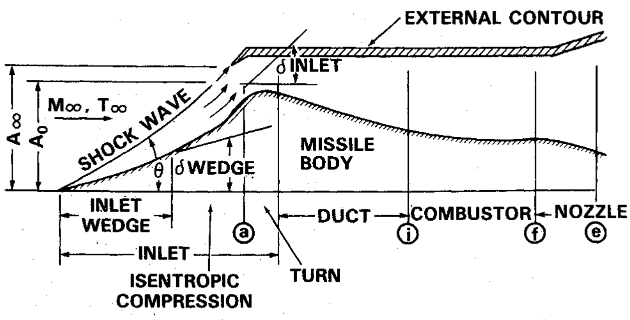

Fig. 1 Internal flow model.

Duct

A duct carries the flow from the inlet to the combustor intake where the Mach number is specified. The process is assumed to be isentropic; but as in the inlet, pressure losses can be taken into account.

Combustor

Combustion is assumed to be a continuous process with fuel being burned along the length of the combustor. The combustor cross-sectional area is adjusted to maintain a specified longitudinal distribution of Mach number. Each increment of the process is divided into two subincrements: a constant-area heat addition followed by an isentropic expansion to the specified local Mach number. The formula for heat addition (Ref. 4, for example) from point (a) to point (b) gives:

$$M_b^2 = \Big[1+\gamma M_a^4 - 2\gamma M_a^2 C \pm (1+\gamma M_a^2)\sqrt{(M_a^2-1)^2 - 2(\gamma+1)M_a^2 C}\Big] \div \Big[2\gamma M_a^2 + 2\gamma^2 M_a^2 C - \gamma + 1\Big]$$

The nondimensional local heat addition C is given by

$$C = H\frac{Ef_o}{NC_pT}\eta_C$$

η_C is a specified efficiency factor accounting for heat transfer to combustor walls, incomplete burning, or other losses. H is the local heating value of the fuel, given, for example, in Ref. 5. Similarly, γ is a local value for the ratio of specific heats. For computational purposes, these quantities were approximated for JP-5 fuel by the following formulas where T is in degrees Rankine:

$$H\times 10^{-9} = 0.517 + 0.04\left(\frac{T-460}{1500}\right) - \left[0.03167 + 0.025\left(\frac{T-460}{1500}\right)\right]E + \left[1.267 + 0.1\left(\frac{T-460}{1500}\right)\right]E^2 - (0.19913 - 0.0000155\,T)E$$

$$\gamma = 1.41817 - 0.0000395\,T$$

However, this model does not account for the products of combustion that are accumulating along the combustor. Furthermore, at temperatures above 4500°R, the dissociation of air can introduce large errors into this formula.

In general, a numerical integration is required to calculate the combustion process.

Exhaust Nozzle

The flow in the exhaust nozzle is calculated as a one-dimensional, frozen flow, isentropic expansion to freestream pressure using the formulas given in Ref. 3. In practice, it is expedient to use an underexpanded nozzle, thereby saving nozzle weight and space at only small cost in engine thrust. Losses are accounted for by applying a nozzle efficiency factor to the calculated exit momentum.

B. Thrust and Specific Impulse

The thrust is the difference between the momentum leaving the engine and the initial momentum in the oncoming freestream. A thrust coefficient, defined by

$$C_T = \frac{T}{A_\infty q_\infty}$$

is given by

$$C_T = \frac{2}{\gamma_\infty M_\infty^2}\left[\frac{A_e}{A_\infty}\eta_N(1+\gamma_e M_e^2) - 1 - F_8 - \frac{A_e}{A_\infty} + \frac{A_o}{A_\infty}\right] - 2\frac{A_o}{A_\infty}$$

where F_8 accounts for the momentum lost by fluid not captured by the engine inlet, and is given by

$$F_8 = \left(\frac{P_2}{P_\infty} - 1\right)\left(1 - \frac{A_o}{A_\infty}\right)$$

where P_2 is the pressure behind an oblique shock associated with the average turning angle

$$\delta_{av} = \frac{\delta_{wedge} + \delta_{inlet}}{2}$$

The specific impulse (thrust divided by fuel rate), then, is given by

$$I = \frac{C_T M_\infty}{2f_o Eg}\sqrt{\gamma_\infty RT_\infty}$$

C. Three-Dimensional Effects

Starting aft of the cowl lip entrance, the analysis is approximately valid for three-dimensional configurations. For an axisymmetric annular configuration, for example, the main differences would be that the leading wedge would be replaced by a cone, and the reference area would be circular instead of rectangular. Therefore, reasonable comparisons between different three-dimensional configurations can be made on the basis of this two-dimensional and one-dimensional analytical model. However, quantitative comparisons between two-dimensional and three-dimensional configurations would require modifications of the method.

IV. Results of Calculations

The analytical process can now be applied to sample cases in order to indicate potentially favorable configurations.

In the following examples, the freestream Mach number will be fixed at 6 and the temperature at 400°R. The fuel is assumed to be JP-5, for which the stoichiometric fuel/air ratio is 0.06829. The combustor efficiency (ratio of heat added to theoretical value) is assumed to be 0.8. All other efficiencies are assumed to be 100% unless otherwise stated; i.e., the flow in the inlet (except for the oblique shock on the leading wedge), the duct, and the nozzle is taken to be isentropic. Up to the combustor, the ratio of specific heats is constant at 1.4. In the combustor, it varies in accordance with the values given in Ref. 5. Flow in the nozzle is assumed to be frozen, fully expanded, at the ratio of specific heats found at the end of the combustor.

One of the first questions that can be addressed is: What value should be chosen for the equivalence ratio E?

In Fig. 2a, the thrust coefficient, specific impulse, and maximum combustion temperature are plotted against equivalence ratio for combustion at Mach number 2. As the amount of fuel increases, the thrust coefficient also goes up (naturally), but the specific impulse decreases. In order to select an optimum value of equivalence ratio, it is necessary to investigate its effect on weight. The weight-fraction parameter defined previously is plotted against equivalence ratio in Fig. 2b for several values of L/D and time of flight. The value of the engine weight parameter $W_E/q_\infty A_\infty$ is arbitrarily fixed at 0.25. The weight is minimized at an equivalence ratio near 1, but the variation about the optimum is very small, particularly for long flights at high L/D. Therefore the equivalence ratio is arbitrarily fixed at 0.85 for the remaining calculations in order to keep the combustion temperature below 4000°R.

Another design parameter of interest is the inlet wedge angle. Figure 3 shows thrust coefficient and specific impulse plotted as functions of wedge angle. A wedge angle of 10 deg appears to suffer little performance loss while providing a reasonably convenient shape for the nose of the missile. Again, this value will be maintained for the remaining calculations.

In Fig. 4, the Mach number at the end of the combustor is varied while the Mach number at the entrance remains at 2, varying linearly with the fuel addition along the length of the combustor. Both thrust coefficient and specific impulse decrease with increasing Mach number, but the temperature also decreases. Changing Mach number at the beginning of the combustor has a similar effect. Thus, it appears to be desirable to burn at as low a Mach number as possible within feasible temperature levels. In fact, the most significant parameter is the heat added. Engine performance generally improves when more heat is added, so that the performance depends primarily on the temperature limit imposed by material constraints. However, this conclusion rests, in part, on the assumed combustion process in which the Mach number distribution is specified. To achieve this result, the distribution along the combustor axis of cross-sectional area and fuel burning rate are prescribed and the pressure varies accordingly. The Mach number distribution could be chosen to approximate a more conventional constant pressure combustion process. The advantage of lower combustion Mach number would then be less pronounced.

If constant Mach number combustion is assumed, then the equivalence ratio must be reduced at low Mach number to keep the temperature within material limits. Figure 5 shows the propulsion weight fraction as a function of combustion Mach number for combustion temperature limited to 4000°R with various values of lift/drag ratio and time of flight. Low combustion Mach number (and low equivalence ratio) may be best but optimum parameters cannot be realistically determined within the accuracy of this analysis.

Another important consequence of the controlling influence of heat addition is the ability to compensate for combustion inefficiency. Figure 6 shows the variation of performance and temperature with combustion efficiency. If the efficiency decreases due to incomplete burning of the fuel, or heat loss to the walls, for example, then the temperature level also decreases. Hence, it is theoretically possible to recover some of

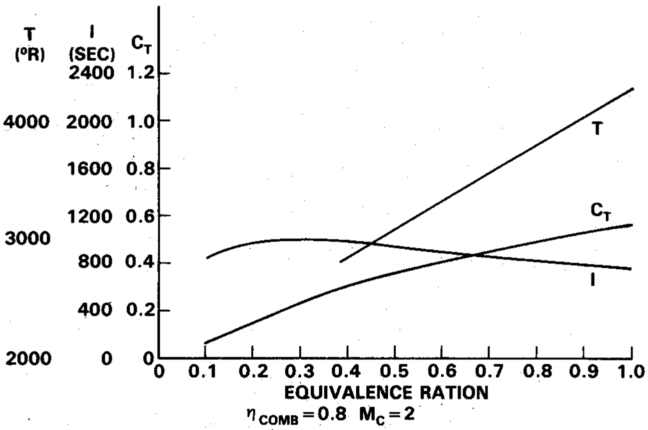

Fig. 2a Effect of equivalence ratio on performance parameters $M_C=2$.

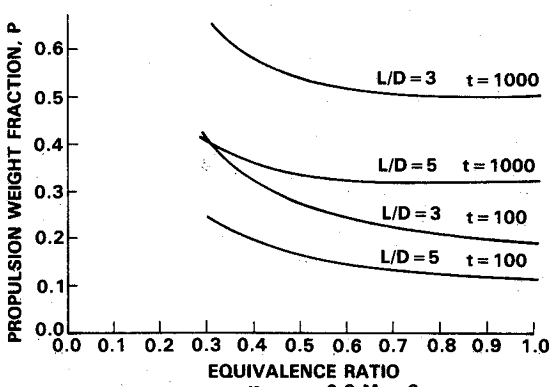

Fig. 2b Effect of equivalence ratio on weight $M_C=2$.

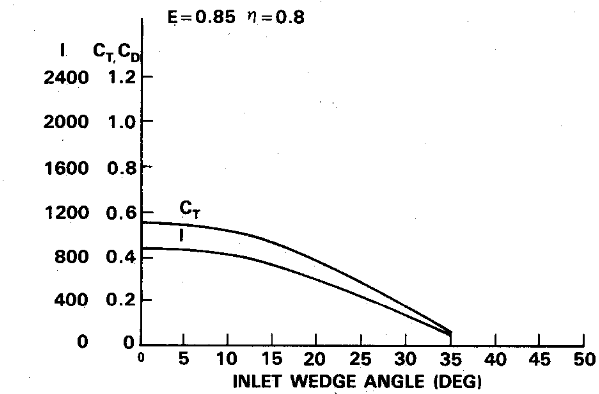

Fig. 3 Effect of inlet wedge angle.

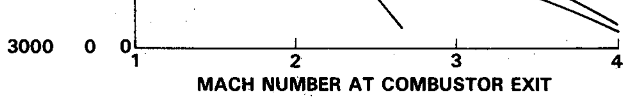

Fig. 4 Effect of Mach number at combustor exit.

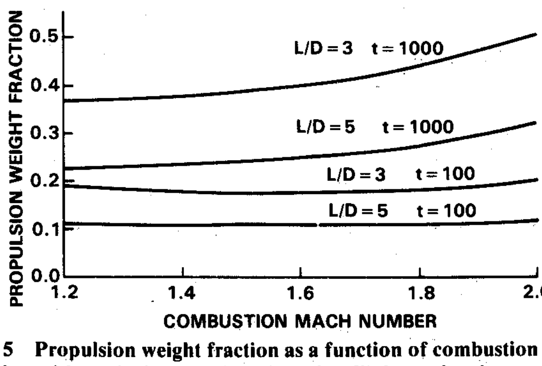

Fig. 5 Propulsion weight fraction as a function of combustion Mach number with equivalence ratio adjusted to limit combustion temperatures to 4000°R.

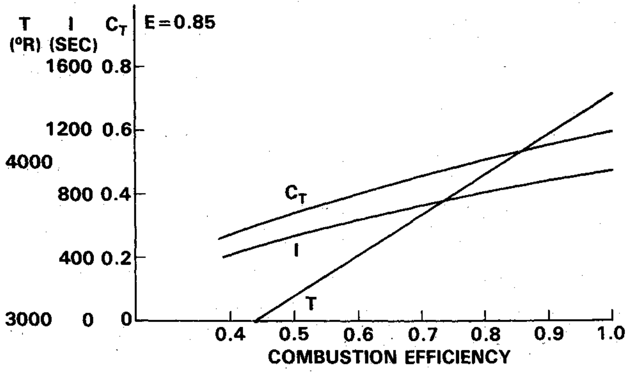

Fig. 6 Effect of combustion efficiency.

this loss by burning at a lower Mach number to bring the temperature back up to its limiting value.

Other useful information can be gleaned by further analysis. For example, tilting up the front of the cowl lip by a few degrees causes a small loss in performance due to the oblique shock wave generated inside the inlet. But this loss is likely to be accompanied by a decrease in external drag on the cowl lip achieved by aligning it more toward the oncoming flow. Also, subcritical operation is undesirable because the throughput of air is reduced from the critical value (bow shock intersecting leading edge of cowl lip) and also a so-called "spillage drag" appears to account for the force on that part of the entering stream tube between the capture area (A_o in Fig. 1) and the area bounded by the entrance lip (A_∞ in Fig. 1).

A further observation is that the nozzle exit area is around 50–100 times the area of the inlet perpendicular to the flow at the cowl lip. Hence, the configuration geometry must accommodate a large nozzle exit area compared to the inlet dimensions. Therefore, it is often expedient not to expand all the way to ambient pressure or to use a plug nozzle.

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