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Conceptual Design of a Hypersonic Vehicle Demonstrator

School of Aerospace, Transport and Manufacturing
Aerospace Vehicle Design: Aircraft Design Option

MSc

Academic Year: 2024- 2025

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Conceptual Design of a Hypersonic Vehicle Demonstrator

Juan Camilo Buriticá Yate

I.ABSTRACT

The development of airbreathing hypersonic vehicles presents additional technical challenges as the result of various design compromises in disciplines such as aerodynamics, propulsion, systems, and materials, all of which need to be integrated to produce a configuration that is not optimized for a specific speed region but provides enough performance to complete its mission. Given this added complexity to the design process, this paper presents a methodology applicable for conceptual design of airbreathing hypersonic vehicles covering the areas of parametric sizing, propulsion, aerodynamics, weights and balance, and trajectory. This methodology was used to develop the conceptual design of an airbreathing hypersonic vehicle demonstrator capable of cruising at Mach 5 at 80,000 ft with a 3500 nm range and a 10,000 lb payload. For this concept, hydrogen and hydrocarbon fuels were considered as well as two different body types: Wing Body and Blended Body. The effects of these design characteristics were assessed in the development of the propulsion system and the parametric sizing process. The propulsion analysis for the ramjet showed significant advantages of hydrogen over hydrocarbon fuel, as hydrogen presented 62.62% lower fuel consumption as well as 6.69% increase in specific thrust. During parametric sizing, it was found that Blended Bodies were lighter than Wing Bodies using the same fuel, due to having lower wetted surfaces areas and hence lower dry weights. When comparing the effects of fuel type, it was found that even though a hydrogen ramjet would consume less fuel in principle, the effects of its low density (74.6 kg/m^3) produced Blended Body vehicle configurations with up to 47.1% heavier MTOW's, 96% heavier dry weights, and 22.7% more fuel required for the same mission, structural and system characteristics as the Blended Body hydrocarbon vehicle. Based on these results, the chosen configuration was the Blended Body hydrocarbon. Using the values obtained from parametric sizing, the vehicle's design was defined in the configuration layout phase and was evaluated for aerodynamics, propulsion and trajectory performance, from which it was determined that the vehicle could complete the mission.

II.NOMENCLATURE

AR	Aspect Ratio, Area Ratio
AoA	Angle of Attack
BB	Blended Body
CD	Conceptual Design
C_{D0}	Parasite Drag
C_{DW}	Wave Drag
CE	Configuration Evaluation
CL	Configuration Layout
C_L	lift coefficient
k_{ve}	Engine Volume Coefficient
K_w	Wetted-to-planform area ratio
LD	Lift-to-Drag Ratio
M	Mach Number
MTOW	Maximum Take-off Weight
OEW	Operating Empty Weight
OWE	Operating Weight Empty
PS	Parametric Sizing
TBCC	Turbine Based Combined Cycle
TW0	Thrust-to-Weight Ratio at Take-off
WB	Wing Body
WS	Wing Loading
h_{PR}	Fuel Heating Value

A. Greek Symbol

α	Engine Lapse Rate
β	Shock Angle, Weight Ratio
θ	Ramp Angle
γ	Ratio of Specific Heats

1.0 INTRODUCTION

The aerospace industry, since its inception, has been a field that has characterized itself for being a leader in innovation, whether by developing new technologies or by creating business models that have positively impacted the way people travel and interact with others around the world, the objective of pushing the boundaries, which lies at its roots, has progressively been narrowing down the gap between what engineers and scientists think is possible and impossible, reaching new milestones with each coming decade. One goal that has been present for hundreds of years in society has been the pursuit of traveling further and faster by the most efficient means possible. The transition from walking to riding horses, or from taking a train to flying on a commercial airliner, reminds the industry that time is a precious asset and the better it is spent, the more room for economic growth and human connection opportunities there is. Although the aerospace industry has reduced worldwide travelling times to historic minimums by the development and use of transonic and supersonic aircraft, the curious nature of engineers and scientists' postulate that hypersonic speeds (speeds greater than Mach 5) could be the next breakthrough for manned aerospace vehicles for commercial and military applications. This paper presents an overview of airbreathing hypersonic vehicle

development, a general review of technical considerations for engineering disciplines, and the synthesis of multidisciplinary methods applicable for conceptual design with the objective of developing a manned vehicle concept capable of achieving hypersonic cruise.

Keywords: Hypersonic, Conceptual Design, Parametric Sizing, Aerodynamics, Propulsion, Trajectory.

1.1 Mission Requirements

The mission requirements were determined in the context of pre-existing studies done on hypersonic demonstrators by Pegg et al [1] as well as the SR-71 requirements provided by Mixon et al [2]. Additionally, the F100-PW-229 was chosen as the requirement for the gas generator as this turbofan is being used by Hermeus for its Chimera II TBCC, currently under development. The mission requirements are summarized as follows:

Table 1	
Mission Requirements	
M_{cruise}	5
$Altitude_{cruise}[ft]$	80,000
$R_{cruise}[nm][1]$	3500
$W_{payload}[kg][1]$	4536
Crew	1
Propulsion System	TBCC (F100-PW-229 + Ramjet)
$S_{TOG}[ft][2]$	4600

2.0 Literature Review

The literature review made for this study consisted of two main areas. The first is a technical literature review in which design characteristics and other technical considerations were gathered to better orient the configuration layout and the conceptual design characteristics needed to be analysed. The second area consisted of gathering the adequate methods to be used for analysis with the appropriate level of complexity for conceptual design. Both literature review areas are presented below.

2.1 Technical Considerations of Engineering Disciplines

Hypersonic vehicles are ones capable of operating within the hypersonic regime which is considered to begin at Mach 5. However, as Anderson states in [3], hypersonic effects do not occur abruptly once a vehicle goes past Mach 5 but rather after this point these effects that have been occurring at lower speeds become more pronounced and start to have greater impact on the vehicle. These effects produce complications in design, particularly in the areas of aerodynamics, propulsion, and materials selection. This section provides an overview of the technical considerations captured from an extensive literature review for aerodynamics and propulsion. The information presented in this section was obtained from published sources from dates ranging from the 1960s to 2025.

2.1.1 Aerodynamics

Hypersonic vehicles and their aerodynamic characteristics are heavily dictated by the type of mission they must perform. The factor that influences the overall design is how much time the vehicle spends in the atmosphere as this defines how much heat it will endure and the availability of oxygen throughout the mission. These two parameters, heat generation and oxygen availability, are crucial design characteristics as they influence the vehicle geometry (shape and volumetric characteristic) as well as the powerplant type that must be used (see section 2.1.2).

For the interests of this paper, focus will be on atmospheric cruisers, particularly horizontal take-off / horizontal landing configurations (HTHL), as these vehicles must operate through all the main aerodynamic regions. Aircraft designers may develop a vehicle that operates optimally within a given region, however, the optimal design requirements for each region are significantly different, meaning that for a vehicle that operates through all of them several design compromises must be made resulting in a configuration that is not ideal for any specific flight speed but can complete the mission. One particular example is the Concorde, whose aerodynamic design point lied in the supersonic region but had to operate through the subsonic and transonic regions, resulting in a configuration that had severe volumetric constraints (smaller cross sectional area to minimize wave drag meant less space for passengers resulting in more expensive business models which had an important role in its retirement) as well as the characteristic ogive planform shape required to minimize drag (specially through the transonic region) at the expense of poor subsonic performance which resulted in high noise during take-off (more thrust required to take-off within airport runways) and the characteristic drooped nose mechanism to improve pilot visibility during take-off and landing due to the required high angles of attack influenced by the reduced lift-curve slope of delta wings. HTHL hypersonic vehicles are subject of similar design compromises as they must follow specific design philosophies to increase aerodynamic efficiencies and reduce heating at hypersonic speeds at the expense of reduced performance in the other speed regimes.

Hamm and Best [4], define general airframe characteristics for hypersonic vehicles which revolve around 3 main design factors namely fineness ratio, low wing loading, and a focus on high lift to drag ratios. These characteristics result in configurations which can be grouped into 3 main body types: Wing-Body (WB), Blended Body (BB), and all-body (AB). As can be seen in Figure 1, these body types exhibit different behaviours dominated by mission and the type of fuel used in their propulsion system. WB and BB are predominantly useful for interceptors due to better aerodynamic characteristics at lower speeds (critical for HTHL missions) while AB are mainly used for long range transports, defence purposes (strike vehicles) and booster stages at higher speeds. Part of the reason for this transition from a more defined WB at lower speeds (Mach 4 – 6) to an AB configuration at higher speeds (Mach 12-25) is the type and amount of fuel that is required to achieve those speeds. Hydrocarbon and Hydrogen are the predominant types of fuel where Hydrogen provides more energy per unit mass than hydrocarbon, as it is energetically denser (kJ/kg), however, the density of hydrogen can be 11 times lower resulting in a fuel tank capacity with a volume requirement 4 times greater for the same energy. More volume means a stouter vehicle, which decreases its aerodynamic performance (L/D). Furthermore, the trade-off between usable volume and aerodynamic performance seems to be in better balance for the blended body configurations, as detailed in the MCAIR hypersonic research facilities study [5].

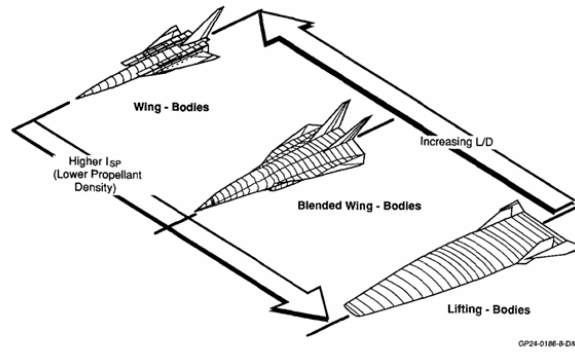


Figure 8. Aerodynamic Shape Represents a Tradeoff Between Thrust and Drag

Figure 1. Hypersonic Vehicle Body Types and General Trade-Offs [4].

Hamm and Best [4] offer additional considerations for drag. They emphasize on keeping the wing surface entirely within the region encompassed by the Mach cone generated at the design point to avoid strong shockwaves that cause wave drag. They also recommend the use of double wedge or biconvex aerofoils with 4% thickness and to design for lower wing loadings to improve aerodynamic conditions for take-off and landing.

2.1.2 Propulsion

HTHL vehicles present additional levels of complexity compared to launch and aerial release vehicles as their propulsion unit must operate through different speed regions. For HTHL airbreathing cruise vehicles, the propulsion system of choice is a turbine based combined cycle (TBCC) which consists of a gas generator (Mach 0 – 2.5) unit with the addition of a ramjet (Mach ~2.3 and Mach 6) or a scramjet engine (Mach 6+) as shown in Figure 2. This implementation is evidenced in the SR-71 (turboramjet), Saenger II (turboramjet) [6,7], and in hypersonic vehicle studies by Ingenito et al [8,9] (turboramjet), Kobayashi et al [10] (turboramjet), Zumwalt et al [11] (turboramjet), Ou-Yang et al [12], and MCAIR [5] (turbofan + ramjet/scramjet). Based on this precedence, the propulsion unit for this study will consist of a turboramjet TBCC as it comprises speeds ranging from Mach 0 to Mach 5 (cruise Mach). Regarding technical considerations, the focus will be shifted to gas generators and ramjets, with details regarding inlets, nozzles, and fuel types.

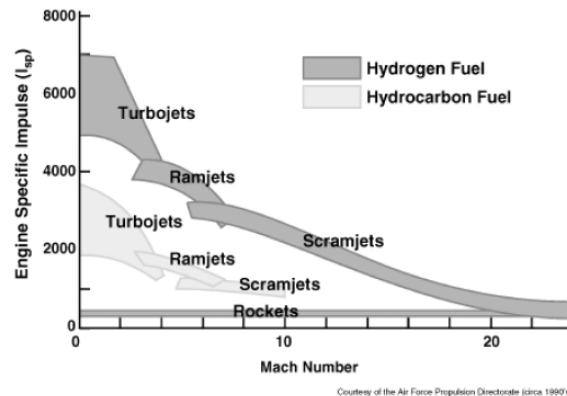


Figure 2. Engine Specific Impulse vs Mach Number [13]

For gas generators, the engine type of choice lies between turbojets or low bypass turbofans. The reason behind this is not only the smaller size for these engines which allows for better integration (minimizing drag) but also their higher efficiency at high speeds compared to high bypass turbofans.

Regarding technical considerations for the ramjet engine, this can be integrated with the gas generator in a coaxial/tandem or over-under configurations, as presented by Zumwalt et al [11] and Hamm et al [4] (shown in Figure 3). There are benefits and compromises for each configuration but based on the results presented by Zumwalt et al [11], the over-under configuration was selected for this study due to its simpler cycle modelling (no mixed flow analysis, especially at the combustor/after-burner exits) and also to their shorter length and lighter weights which allows for better integration into the airframe.

A critical component of the engine system is its inlet as it conditions the flow entering the engine by assuring it reaches optimal levels of pre-compression (due to ram effects at higher speeds) and the correct flow direction to minimize losses produced by spillage. When it comes to their design, at subsonic speeds the pressure recovery could be assumed to be constant and can be achieved internally (before the compressor), however, at supersonic speeds it decreases with increase in Mach due to compressibility effects. This forces high speed engines to require additional pre-compression mechanisms to maximize recovery. At speeds lower than Mach 2.5, the inlet can be designed to provide external compression via ramp deflections (as described by Prandtl-Meyer relations) which generate oblique shocks that compress the air more efficiently than normal shocks. Additionally, ramp inlets must account for variable geometry to change their deflections as these are functions of the freestream Mach. At higher speeds the flow would require greater deflection which in turn comes with higher cowl drag [14]. Because of this, inlets for hypersonic speeds require a mix of external and internal compression (internal reflected oblique shocks), resulting in mixed compression inlets. Although mixed compression inlets provide higher recovery with less drag penalties, they introduce unstart problems due to the internal portion having a throat. To alleviate this problem, the internal portion must account for variable throat geometry to satisfy the Kantrowitz criterion at a given speed [14,15]. Based on this, the inlet design for the propulsion unit of this paper will consist of a mixed compression inlet and additional considerations will be pointed out in the methods section.

To achieve maximum efficiency, the engine must expand the high pressure/temperature exhaust gases via a nozzle so that the gas pressure matches that of the ambient pressure at a given flight condition. Following isentropic relations, as the freestream Mach number increases so does the required expansion area, however this results in longer nozzles that may become impractical for adequate vehicle integration [4]. Due to this, a compromise between expansion and nozzle geometry must be made to avoid large geometries at the expense of under expansion of the exhaust flow [4,16].

Regarding fuel type, the fuel of choice tends to be between hydrocarbon and liquid hydrogen fuels. On one hand, Hydrocarbon fuels have lower but adequate hPR (in the range of 42,800 kJ/kg) along with high density (in the range of 800 kg/m³) and high flashpoints which has made it the ideal selection for high-speed vehicles. A critical aspect of the fuel selection is the implementation of regenerative cooling for the engine, and as stated by Hunt et al [17], hydrocarbon fuels have an operational limit at around Mach 8 where they can no longer provide adequate cooling. On the other hand, liquid hydrogen has been considered as an alternate fuel for high-speed missions due to its high hPR (around 120,000 kJ/kg which almost triples that of hydrocarbons) as well as higher cooling capacity for regenerative cooling systems beyond Mach 8. The high hPR of hydrogen allows for more efficient performance by having a higher Isp as evidenced in Figure 2 which means a 3rd of the mass is required for the same amount of energy, however, the lower density of liquid hydrogen (74 kg/m³) results in having to allocate about 4 times more volume for the same amount energy as hydrocarbon fuels. In addition to this, liquid hydrogen requires heavy and complex cryogenic

tanks which contribute to the drawbacks for this type of fuel.

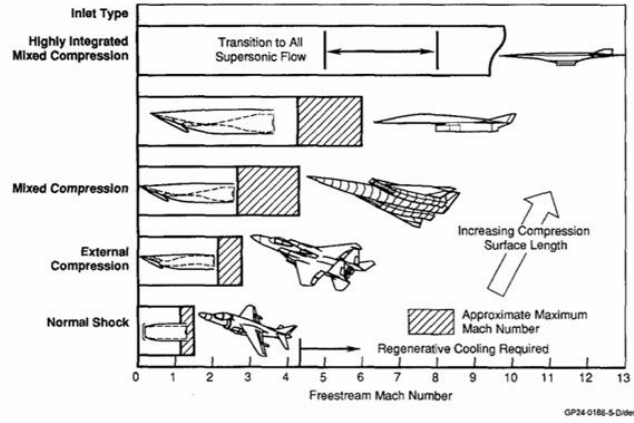
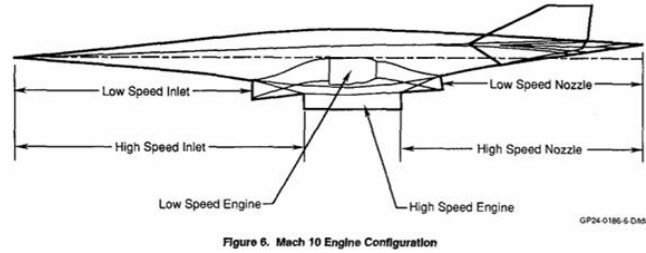


Figure 3. TBCC Engine Over-Under Layout [4].

2.2 Conceptual Design Methods

This section introduces the methodologies applicable to conceptual design (CD) throughout its different phases: Parametric Sizing (PS), Configuration Layout (CL) and Configuration Evaluation (CE). PS refers to the process of translating mission requirements into preliminary size characteristics of the aircraft such as weight, planform area, thrust required, etc. selected from a design space that is constrained by certain performance criteria that each of the parametric sizing variable combinations must meet to consider the design to be feasible. In CL the parametric sizing variables are used as initial inputs to determine more detailed aspects of the configuration such as planform shape, powerplant design, preliminary weights estimation and balance to provide the first iteration of the concept. Once a configuration has been developed, the design is then evaluated via low order analytical, numerical, and semi-empirical methods to evaluate if the proposed concept meets the expected requirements for the relevant disciplines and can complete the mission.

2.2.1 Hypersonic Convergence - PS

Hypersonic Convergence (HC) ([15] Chapter 16) is a parametric sizing methodology developed by Czysz in the 1960's. Czysz identified the need for a different sizing approach for hypersonic vehicles (launchers, accelerators) as these require high levels of systems integration in which the entire airframe impacts propulsion, weights and aerodynamics all while allocating internal volume for systems, payload, and fuel. For this reason, HC offers an iterative logic that relates geometric, systems and structural characteristics of the vehicle such as planform area, fuel volume, engine volume, wetted area, payload volume, structural index (Istr), etc. to the total weight of the vehicle. The fundamental weight equations for HC are

shown in equations(1 and(2. OEW is the weight budget of the vehicle, and it is derived by summing the weights of the structure, systems and engine but in terms of design characteristics such as Kuchemann Tau, planform area (S_{pl}), engine thrust-to-weight (TW0), etc. OWE is the volume budget equation which gives the weight of the vehicle (minus payload and crew), and it is derived by summing the volumes of the airframe, crew, payload, engine and propellant and includes density terms that relate them to their respective weights. For a fixed Tau and remaining variables, the planform is iterated until the weight and budget equations yield the same weights. If this point is reached the designed is said to have converged and becomes a feasible design point. For additional details see APPENDIX 12.

$$OEW = \frac{I_{str} \cdot K_w \cdot S_{pl} + C_{sys} + W_{cprv} + \frac{(\frac{T}{W})_{TO} \cdot WR}{E_{TW}} (W_{pay} + W_{crw})}{\left(\frac{1}{1+\mu_a}\right) - f_{sys} - \frac{(\frac{T}{W})_{TO} \cdot WR}{E_{TW}}} \approx W_{dry} \quad (1)$$

$$OWE = \frac{\tau \cdot S_{pl}^{1.5} (1 - k_{vv} - k_{vs}) - (v_{crw} + k_{crw}) \cdot N_{crw} \cdot \frac{W_{pay}}{\rho_{pay}}}{\frac{WR-1}{\rho_{ppl}} + k_{ve} \cdot \left(\frac{T}{W}\right)_{TO} \cdot WR} = OEW + W_{pay} + W_{crw} \quad (2)$$

Kuchemann's Tau is a term that relates the volume of a vehicle to its planform area, and it is given by equation(3. Equation(4 shows the US industry convention of the same relationship. An additional term of interest is k_w (equation (5)) which relates the aircraft's wetted area to its planform area for a given Tau (Figure 5). Tau is a key parameter for the sizing logic as for a given Tau and planform, the total volume and wetted area of the airframe can be obtained and hence the remaining weights, so Tau is analogous to the classical wing-loading (WS) term for this approach [18].

$$\tau = \frac{V_{total}}{S_{pl}^{1.5}} \quad (3)$$

$$t = \tau^{\frac{2}{3}} \quad (4)$$

$$k_w = \frac{S_{wet}}{S_{pl}} \quad (5)$$

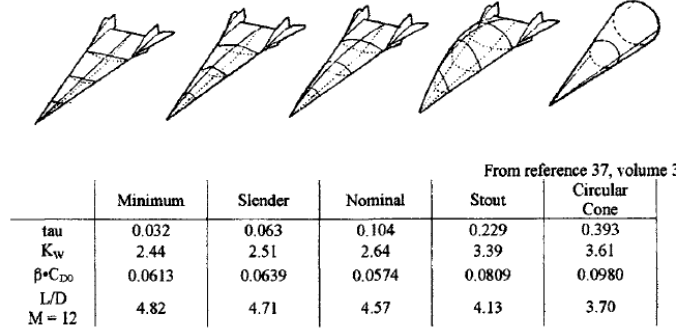


Figure 4. Vehicle Characteristics as Function of Tau [15].

Propulsion Integrated Configurations:

Blended Body	$K_W = -62.217 \cdot \tau^3 + 29.904 \cdot \tau^2 - 1.581 \cdot \tau + 2.469$
Wing-Body	$K_W = -93.831 \cdot \tau^3 + 59.920 \cdot \tau^2 - 5.648 \cdot \tau + 2.821$
Nonweiler Waverider	$K_W = -533.451 \cdot \tau^3 + 220.302 \cdot \tau^2 - 22.167 \cdot \tau + 3.425$

Figure 5. K_w as function of Tau for Different Bodies [15].

The HC code has 3 built-in modules, one for geometry to obtain k_w (estimated using Figure 5) and the vehicle length (obtained using the method from APPENDIX 1), one to estimate the TW0 based on the required acceleration through the transonic region (M 0.9 – 1.2) which was assumed to be the highest thrust-to-weight ratio requirement for the gas generators, and one to estimate the fuel fraction for a general trajectory. The transonic constraint (equation (6) which is a general thrust-to-weight constraint used by Hassan et al [19], includes propulsion coefficients for lapse rate and Mach effects, namely α and β which were calculated for a low-bypass-ratio turbofan with after burner using the method by Howe (see APPENDIX 2) [20]. To obtain, LD_{max} , $CD0$ and CDW at PS phase, the aerodynamic methods used by Coleman [18] (originally by Czysz) was used (see section 2.2.7 and APPENDIX 3 & 4). For the fuel fraction module (see APPENDIX 5), the weight ratios for take-off, accelerated climb, descend and landing were estimated using recommended values from Raymer [21], Nicolai et al [22], and Roskam [23]. For the cruise weight ratio, the Breguet range equation (equation (7) presented by Anderson [24] was used.

$$\left(\frac{T}{W}\right)_{TO} = \frac{\beta}{\alpha} \left\{ K_1 \left(\frac{\beta W_{TO}}{qS} \right) + \frac{C_{D0} + C_{DW}}{\beta \frac{W_{TO}}{qS}} + \frac{1}{g_0} \frac{\Delta V}{\Delta t} + \sin(\theta) \right\} \quad (6)$$

$$\frac{W_i}{W_f} = e^{\left(\frac{-R}{\left(\frac{V LD_{max}}{c g} \right)} \right)} \quad (7)$$

A general N-S diagram of the HC MATLAB code logic is presented in Figure 7. The result of HC is a solution space that differs from the TW0 vs WS approach. Instead of having a feasible design area bounded by performance constraints, HC presents a curved line composed of points at which the code has converged by for a given Tau. This converged curve implies that the weights and volumes of the systems, payload, fuel, crew, etc. from the mission requirements and design inputs have been properly integrated into a vehicle that satisfies the estimated fuel fraction and transonic acceleration constraints. Additionally, as recommended by Chudoba et al [25], a WS constraint for stall during landing can be imposed onto the solution space to further constrain it. For the stall at landing constraint, the FAR 23 approach speed relationship was used to determine the stall speed following Roskam's method [23] as shown in equations (8 and (9. For this analysis, data for the SR-71 obtained from Mixon et al [2] was used and the weight ratio $W_{correction}$ was included to correct the wing-loading for landing conditions. Figure 6 shows a generic behaviour for the HC solution space as seen in Coleman [18], Chudoba et al [25], and Ingenito et al [8].

$$V_s = \frac{V_a}{1.3} \quad (8)$$

$$\left(\frac{W}{S}\right)_{TO} = \frac{1}{2} \rho V_s^2 C_{Lmax} W_{correction} \quad (9)$$

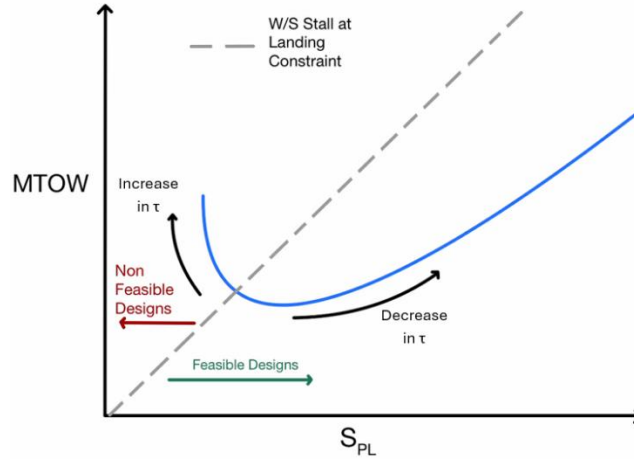


Figure 6. Generic HC Solution Space.

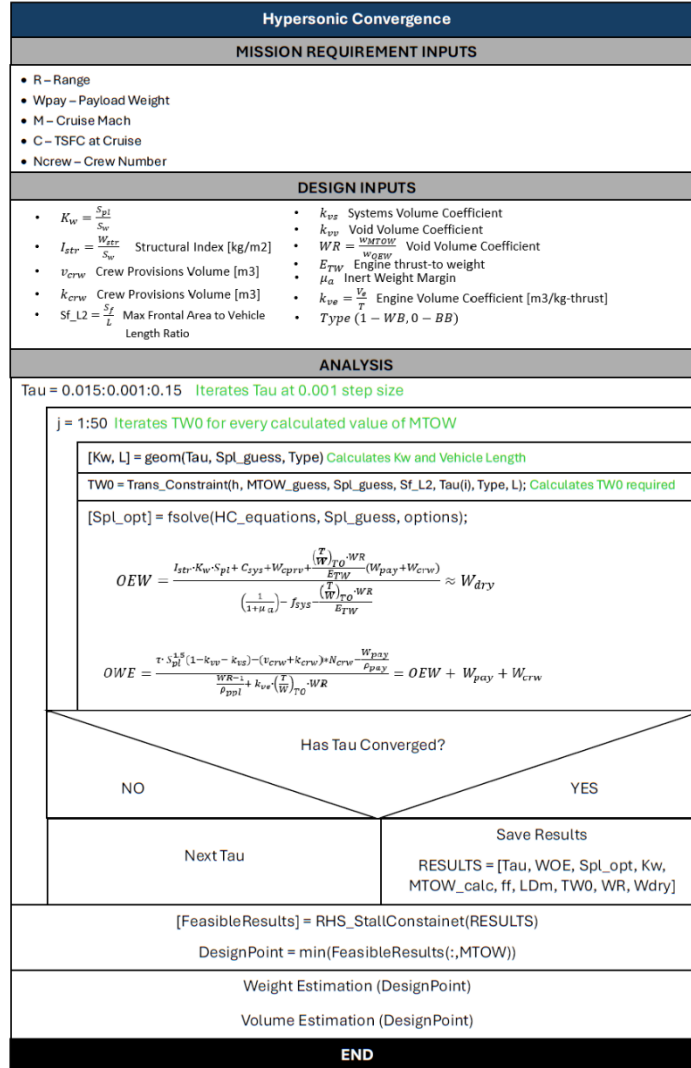


Figure 7. HC Code NS Diagram

2.2.2 Mixed Flow Turbofan with Afterburner Real Cycle Analysis

To reproduce the thrust and TSFC values for the gas generator, NASA's engine simulator [26] was used with its F-100 turbofan model. This approach was taken as the engine simulator calculated the real performance of the F-100 taken into consideration component efficiency and mass flowrate variations with Mach Number and altitude.

2.2.3 Real Ramjet Cycle Analysis

To reproduce the thrust and TSFC values for the Ramjet, Mattingly's [27] real turbojet cycle analysis was used to simulate a Ramjet engine using a compressor ratio of 1 as suggested by Farokhi [14]. For full set of equations see APPENDIX 6. The cycle analysis MATLAB code was used as a function of Mach and Altitude to generate the data for trajectory analysis.

2.2.4 Inlet Design

For the mixed compression inlet design, the method by Sandeep [28] was followed. The method consists of calculating the external ramp angles using oblique shock relations for 3 deflections (ramps) to achieve a desired Mach reduction from freestream to the cowl entrance. For the internal compression, the same oblique shock relations with the addition of a final normal shock were calculated to achieve a desired isolator Mach number having the Cowl Mach as the incoming Mach number. A MATLAB code was developed to calculate these relations and vary different parameters (M_∞ , M_{cowl} , $M_{isolator}$, N_{ramp} , $N_{internal-shocks}$) to obtain a desired pressure recovery as well as a $M_{isolator}$. The shock angles were determined following the Oswatitsch criteria for maximizing recovery. Once the ramp angles are determined, the geometry of the ramp inlet can be constructed using trigonometry for the 1st ramp as its shock is constrained to impinge on the cowl lip. For the remaining ramps, SolidWorks 2025 was used to constrain the geometry using dimensioning via equations in sketch mode to determine the ramp lengths and Cowl Height. The internal portion was assumed to be a convergent duct that would satisfy the Kantrowitz criterion at the design point (Mach 5) to avoid inlet unstart.

$$CR_{Kantrowitz} = \frac{A}{A^*} = \left(\frac{(\gamma + 1)M^2}{(\gamma - 1)M^2 + 2} \right)^{0.5} \left(\frac{(\gamma + 1)M^2}{2\gamma M^2 - (\gamma - 1)} \right)^{\left(\frac{\gamma + 1}{2(\gamma - 1)} \right)} \quad (10)$$

MachBetaTheta Function
INPUTS
<ul style="list-style-type: none"> • Beta – Shock Angle • Theta – Deflection Angle • M1 – Incoming Mach • gamma – Ratio of Specific Heats
$A = \tan(\theta)$ $B = 2 \cot(\beta) \left(\left(\frac{M1^2 \sin^2(\beta) - 1}{M1^2(\gamma + \cos(2\beta))} \right) + 2 \right)$ Residual = A – B calculates the difference for fsolve
END

Figure 8. M-β-θ Relations Function NS Diagram

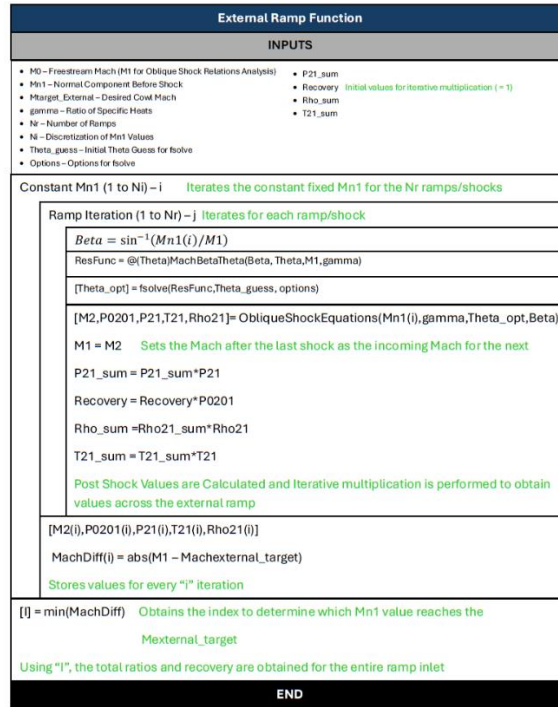


Figure 9. External Ramp Function NS Diagram

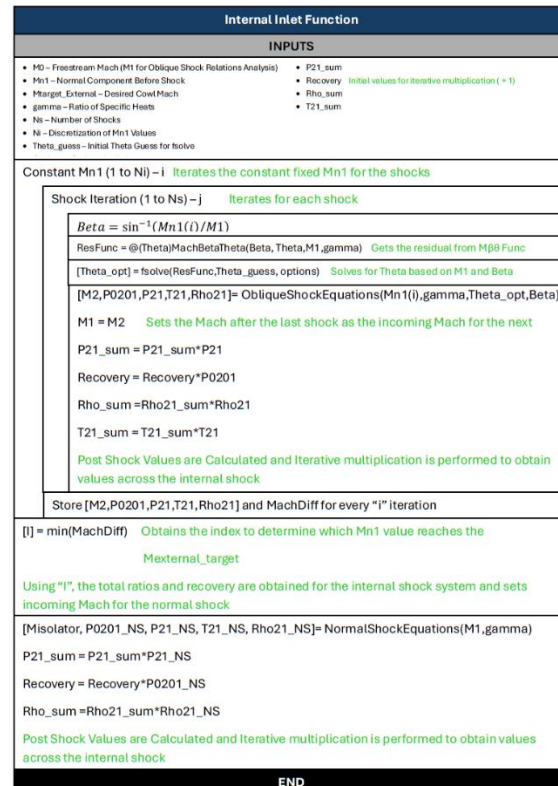


Figure 10. Internal Inlet Function NS Diagram

2.2.5 Combustor Geometry

When the Mach number entering the combustor is at a high subsonic Mach (0.5 – 0.7), 1-D Rayleigh flow analysis (constant area duct) estimated a choking condition downstream which is unwanted since having a fixed mass flowrate downstream while having a higher incoming mass flow rate it increases the back pressure from the combustor damaging the inlet shock system upstream causing unstart. To avoid this, the zero-dimensional analysis approach by Ananthkrishnan et al [29] was implemented to determine the required inlet-outlet divergent combustor area ratio to achieve a desired subsonic combustor exit Mach number to avoid choked flow. The method consists of solving the conservation equations (energy, momentum, and mass) using fsolve in MATLAB whilst using non dimensional parameters such as area (AR), velocity (u) and density (r). This method accounts for frictional losses (via a friction coefficient) as well as heat transfer through the combustor's walls, however for this study the conditions were assumed to be adiabatic and frictionless to simplify the analysis. For details see APPENDIX 7.

$$r \cdot u \cdot AR = 1 + f \quad (11)$$

$$r \cdot u^2 \cdot AR \left(\frac{1}{\gamma_2 M_2^2} + 1 \right) = \left(AR \cdot \frac{1}{\gamma_1 M_1^2} + 1 \right) - \left(\frac{1}{2} \right) Cf \overline{A_w} \quad (12)$$

$$\frac{1}{\gamma_2 - 1} \left(u^2 \frac{1}{M_2^2} - \bar{R} \bar{\gamma} * \frac{1}{M_1^2} \right) + \left(\frac{1}{2} \right) (u^2 - 1) = \frac{\bar{Q}_c + \bar{Q}_w}{1 + f} \quad (13)$$

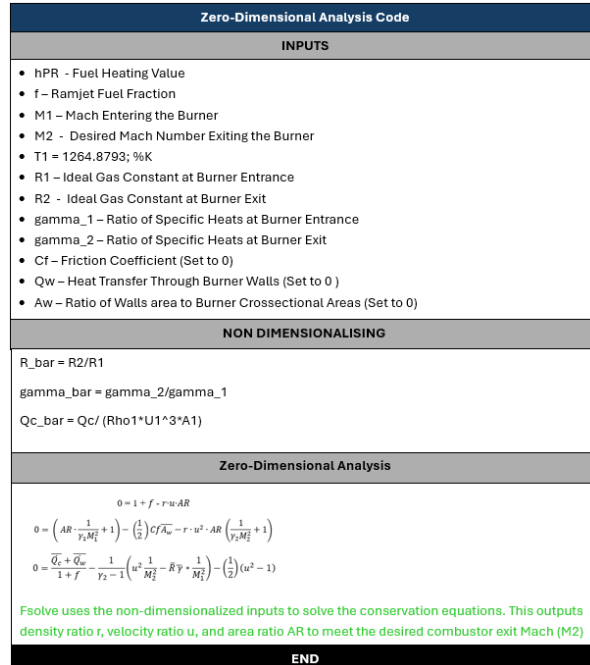


Figure 11. Zero-Dimensional Analysis Code NS Diagram

2.2.6 Additional Geometry Aspects

Additional dimensions such as isolator and combustor length were obtained from Anderson et Al ([15] Chapter 6) and Dutt [30]. These were relationships used in their respective studies, but detailed analyses including numerical methods and testing are required to better determine the component lengths. For the nozzle geometry, the isentropic area ratio equation [28] was used to determine the throat at the nozzle as well as the exit area as suggested by La Sorsa et al [31].

$$l_{isolator} = 4.7 H_{throat} \quad (14)$$

$$l_{combustor} = 6 H_{throat} \quad (15)$$

$$\frac{A}{A^*} = \frac{1}{M} \left[\left(\frac{2}{\gamma + 1} \right) \left(1 + \frac{\gamma + 1}{2} M^2 \right) \right]^{\left(\frac{\gamma + 1}{2(\gamma + 1)} \right)} \quad (16)$$

2.2.7 Aerodynamics - PS

At the early stage of PS, several geometric parameters that are key for aerodynamics analysis (S_{pl} , AR) are yet to be defined as they belong to CL. The method provided by Coleman [18] (see APPENDIX 3 & 4) was implemented using empirical data for highly swept wings. From this data, L' (induced drag factor), LD_{max} and C_{DW} were obtained. C_{D0} was then calculated using the following relationship:

$$C_{D0} = \frac{1}{4 \cdot L' \cdot LD_{max}^2} \quad (17)$$

2.2.8 Subsonic Aerodynamics - CE

For subsonic aerodynamics, two methods applicable to delta wings were used. The first method consisted of a hybrid between numerical potential flow solutions obtained with OpenVSP paired with a nonlinear vortex lift term provided by Kuchemann following the methodology implemented by Abraham et al [32]. As seen in equation(18, the normal coefficient (in this case lift coefficient) is composed by a C_{L-P} obtained from a potential flow of the wing-body and a nonlinear vortex lift term C_{L-V} (used for calculations at high AoA). The C_N coefficient was corrected for compressibility effects via the compressibility factor β . Additionally, this method allows for the entire wing geometry to be captured in the potential flow calculations. The second method developed by Polhamus [33], follows a similar approach where potential flow characteristics were calculated using panel methods and then an additional vortex lift term is also introduced to provide the overall lift coefficient as seen in equation(19. The K_P and K_V terms were obtained from charts which were a function of planform geometry and Mach number.

$$\frac{c_N}{s} = \beta \left(C_{L-Potential} + 4.9 \left(\frac{\alpha}{\left(\frac{s}{l} \right)} \right)^{1.7} \right) \quad (18)$$

$$C_L = K_p \sin^2(\alpha) \cos(\alpha) + K_v \sin(\alpha) \cos^2(\alpha) \quad (19)$$

One characteristic of highly swept wings is that lift is enhanced when they have an increased span at the rear portion also known as a double delta configuration or cranked wing as pointed out by Polhamus [34] and Hopkins et al [35], which increases lift and minimize drag allowing the achieve higher LD. For this reason, during CL, a double delta wing configuration was employed but to use the aforementioned methods in which it is required to use a delta wing equation (20 was used to calculate an effective leading edge sweep angle whilst keeping the same aspect ratio for the CL cranked wing.

$$\Lambda_{eff} = \cos^{-1} \left(\frac{\cos(\Lambda_{LE-inboard})S_{pl-inboard} + \cos(\Lambda_{LE-outboard})S_{pl-outboard}}{S_{pl-inboard} + S_{pl-outboard}} \right) \quad (20)$$

For C_{D0} and C_{DW} the modules built into OpenVSP were used. For C_{D0} , OpenVSP performs the component build up method using methods by Hoerner, Torenbeek, and DATCOM and considers Mach number as well as Altitude effects while using the Spalding-Chi and Schlichting friction coefficient models for subsonic ($M < 0.8$) and high speed ($M > 0.8$) Mach numbers respectively. For C_{DW} , the Eminton-Lord method which OpenVSP uses, provided better variation of wave drag at higher Mach numbers, something that other surveyed methodologies did not account for since they were only functions of maximum cross-sectional area. These drag methods were also employed to obtain the drag polars at low and high speeds.

For subsonic C_{Di} , the classical approach shown in equation(21 was used for low-speeds with the efficiency factor for swept-wing aircraft provided by Raymer [21] shown in equation(22. The study made by MCAIR [5] suggests an efficiency factor calculated using the Weissinger method but this involves dedicated codes not available for this study. Instead, the method provided by Raymer [21] which stems from leading-edge suction analogy was used to calculate the K factor at high-speeds (equation (23).

$$C_{Di} = \frac{C_L^2}{\pi \cdot AR \cdot e} \quad (21)$$

$$e = 4.61(1 - 0.045AR^{0.68}) \cos(\Lambda_{LE})^{0.15} - 3.1 \quad (22)$$

$$K = \frac{1}{C_{L\alpha}} \quad (23)$$

2.2.9 High-speed Aerodynamics - CE

For high-speed aerodynamics, graphical/analytical method 70012 and graphical/semi-empirical method 90013 [36] by ESDU were used to determine the lift curve slope for angles of attack of 0°- 9° and 10°-20° respectively. These methods were based of empirical and analytical databases which provided the C_N and $C_{L\alpha}$ as functions of planform geometry and Mach Number. These methods are applicable for Mach numbers 1.2 – 5, hence their applicability to this study.

2.2.10 Take-off Analysis - CE

For the take-off analysis, the method by [23] was used to validate that the resulting values from PS (MTOW, S_{pl} , TW0) along with the aerodynamic data meets the mission requirements.

$$S_{TOG} = \frac{k_1 \left(\frac{W}{S} \right)_{TO}}{\rho \left(C_{LmaxTO} \left\{ k_2 \left(\frac{T}{W} \right)_{TO} - \mu_G \right\} - 0.72 C_{D0} \right)} \quad (24)$$

2.2.11 Ascend Trajectory Analysis – CE

For the climb trajectory analysis, the energy state approximation method presented by Bryson et al [37] was used to develop a MATLAB code to assess if the propulsion and aerodynamic data paired with the vehicle sizing values (MTOW and S_{pl}) could perform the mission to ascend to Mach 5 at 80,000 ft. This method calculates the specific excess power (equation (25) and excess power fuel parameter (equation (26) at different energy levels and calculates the minimum time (MTC) and minimum fuel (MFC) to climb trajectories by determining the location of maximum rate of change of energy at each level and then making the vehicle follow those locations until the desired cruise Mach number is achieved. Additionally, the code logic accounted for the mass of the fuel burned at each energy level to constantly update the vehicle's weight as it climbs. Furthermore, a maximum constant dynamic pressure constraint of 1000 psf [38] was enforced to constraint the altitude vs Mach plot to avoid excessive structural loads in the trajectory. For details on the methodology, see APPENDIX 8. Additionally, the codes include an international standard atmosphere model using the method provided by Smith [39].

$$\dot{E} = \frac{V(T - D)}{W} \quad (25)$$

$$FuelParam = \frac{V(T - D)}{f} \quad (26)$$

3.0 Analysis and Results

This section presents the results obtained for each of the disciplinary tasks performed for this study. The order of this chapter's contents match the methodology's order which started with developing the preliminary propulsion layout to determine k_{VE} which was then used as one of the inputs in PS. Following the results from PS, the sizing values were used to obtain the preliminary vehicle design during the CL phase. Once the vehicle's characteristics were defined in CL, the aerodynamic characteristics were obtained and then used alongside the results from propulsion and PS in the performance trajectory analysis to determine if the vehicle could complete the mission.

3.1 Propulsion

3.1.1 F100-PW-229 Results

The thrust and TSFC results were obtained using NASA's engine simulator [26] for the Mach number range of 0 -2 and an altitude range of 0-60,000 ft as these were the limits from the model. Based on the F-15 Mach 2.5 precedence, the values at Mach 2 were linearly extrapolated to Mach 3 to capture the intermediate behaviour at Mach 2.5. The results for thrust and TSFC are presented in APPENDIX 9.

3.1.2 Ramjet Real Cycle Analysis

For the Ramjet analysis, the design point (Mach 5 at 80,000ft) was evaluated using Hydrocarbon and Hydrogen fuel to determine their performance and potential benefits and drawbacks from each configuration. Considering that a perfectly expanded nozzle would yield an unreasonable geometry for integration as mentioned in 2.1.2, and under expanded value for P_9/P_0 of 0.5 was used. Additionally, the maximum Tt_4 temperature was set higher for hydrogen as its storage temperature would allow for better cooling and hence higher achievable Tt_4 's. The results presented in Table 3, show an overall better performance for the hydrogen fuel as it provides 6.69% more specific thrust than hydrocarbon, which for the same flow rate (capture area) hydrogen produces more thrust or for the same thrust using hydrogen would result in a smaller configuration. The greatest improvement is the fuel consumption in which hydrogen yields 62.62% less consumption than hydrocarbon due to its higher energy density which in principle would result in lighter vehicles due to reduced fuel weight, however, the impact for fuel volume using hydrogen was assessed during parametric sizing (3.2), which provided configurations with poorer aerodynamic performance due to increased cross-sectional area than hydrocarbons. Due to the unavailability of Ramjet data, the validity of this analysis was evaluated via comparing I_{sp} which for each configuration was within the expected values given by Figure 2. Based on this, the results for the Ramjet analysis were deemed appropriate for the next steps in the design.

During the engine integration portion of CL, the capture area and the achievable expansion for proper airframe integration were determined as 5.16 m^2 and $P_9/P_0 = 0.574$. The results for the final hydrocarbon Ramjet engine are summarized for the design point in Table 4 and the Thrust/TSFC vs Altitude data in Figure 12 and Figure 13. One thing to notice from these figures is the drop in performance that occurs at Mach 5, which not only can be attributed to the lower density of the air but also (and primarily) to the change in pressure recovery estimated from MIL-STD-5008D at speeds of Mach 5 or greater. For more information on the design refer to sections 3.1.4 and 3.3.

Table 2
Preliminary Ramjet INPUTS – SI Units

Variable	Hydrocarbon	Hydrogen
M_∞	5	5
$\gamma_{burner\ inlet}$	1.4	1.4
$\gamma_{burner\ exit}$	1.36	1.36
$Cp_{burner\ inlet}$	1.004	1.004
$Cp_{burner\ exit}$	1.239	1.239
h_{PR}	42800	120000
π_{burner}	0.98	10.98
$\pi_{diffuser\ Max}$	0.4	0.4
π_{nozzle}	0.96	0.96
$\pi_{compressor}$	1	1
$e_{compressor}$	0.9	0.9
$e_{turbine}$	0.9	0.9
η_{burner}	0.95	0.95
η_{mixer}	0.99	0.99
Tt_4	2200	2400
$P9/P0$	0.5	0.5

Table 3
Preliminary Ramjet Results – SI Units

Altitude [kft]	Mach	Specific Thrust[N/(kg/s)] (Hydrocarbon)	TSFC [kg/N-h] (Hydrocarbon)	Isp [s]	Specific Thrust[N/(kg/s)] (Hydrogen)	TSFC [kg/N-h] (Hydrogen)	Isp [s]	Specific Thrust %Diff	TSFC %Diff
80	5	552.412	0.239	1532.1	589.42	0.08932	4108.47	+6.69	-62.62

Table 4
Final Hydrocarbon Ramjet Results – SI Units

Altitude [kft]	Mach	Thrust[kN]	TSFC[kg/N-h]	Isp [s]	$\eta_{overall}$	P9/P0
80	5	188	0.236	1558.15	0.532	0.574

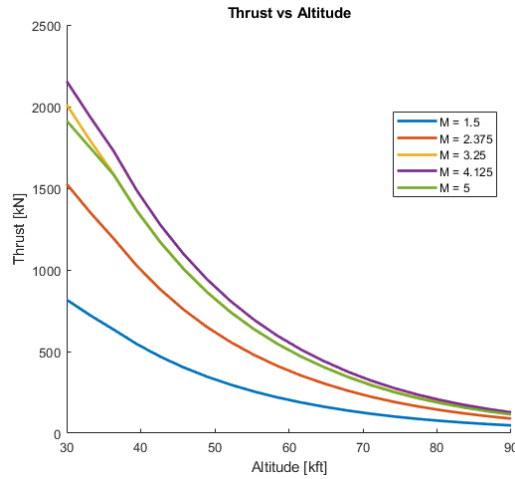


Figure 12. Hydrocarbon Ramjet Thrust vs Altitude

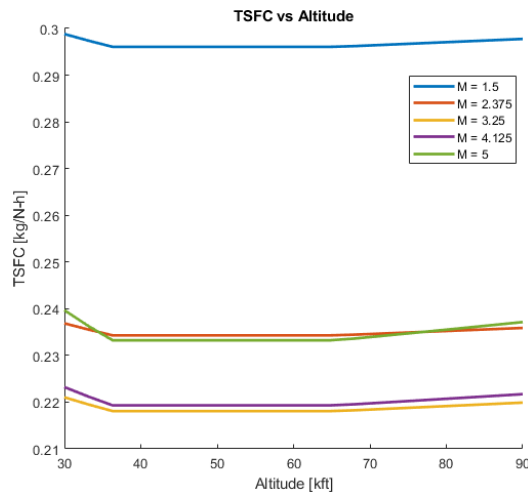


Figure 13. Hydrocarbon Ramjet TSFC vs Altitude

3.1.3 TBCC Cycle Results

The generated cycle analysis results were used to obtain the values for thrust and fuel consumptions required for trajectory analysis for altitudes ranging from seal-level to 105,000 ft and Mach numbers from 0 to 6. The upper bounds of this data are not considered for the operation of the vehicle, however, they had to be increased from 80kft and Mach 5 to increase the energy level calculations. The data was generated at 100% throttle for both engines as no optimization for this variable was considered for this study and the turbofans were constrained to operate from M 0 – 2.5 (Maximum Mach for the F-15) and the Ramjet for M 1.5 +. Additionally, when both engines are running simultaneously a correction factor of 0.5 to the mass flowrate of the ramjet was used to account for the mass flowrate that is going to the turbofans. These results are presented in APPENDIX 11.

3.1.4 Engine Geometry Design

The inlet design process began by determining which M_{cowl} and $M_{\text{pre-NS}}$ would maximize the pressure recovery for the inlet for the design point at Mach 5. The analysis was conducted using shock relations which is an ideal case and to account for viscous losses and effects such as spillage and bleed, a 10% decrease correction factor was applied to the ideal recovery values. A first attempt at maximizing recovery was undertaken by assuming a low supersonic Mach before the normal shockwave as to minimize the losses through it. Although, this increases the pressure recovery further, this causes high subsonic Mach numbers (0.7 – 0.8) to enter the combustor, and with heat addition, Rayleigh flow predicted choked flow ($A/A^* > 1$) inside the combustor which in turn causes unstart condition for the inlet as mentioned in section 2.1.2. Figure 14 shows a pressure recovery surface plot as a function of Mach at the cowl entrance and pre normal shock, color-coded with the Mach number entering the combustor (post normal shock). As can be seen, high recovery values are achieved alongside high subsonic Mach numbers in the combustor. As means to prevent choked flow, the desired Mach number was chosen to be $\sim$ Mach 0.5, whose Max efficiency values are intersected with the MIL-STD-5008B value of 0.5127. Based in this a higher M_{cowl} was chosen as to not have high deflection angles to slow the flow lower than that making the inlet increase in height and hence, cross-sectional area. For the internal compression, 3 internal shocks were considered ($\beta_4, \beta_5, \beta_6$) and the Kantrowitz ratio was calculated taking the cowl Mach as the reference. The final values for the inlet are summarized in Table 5.

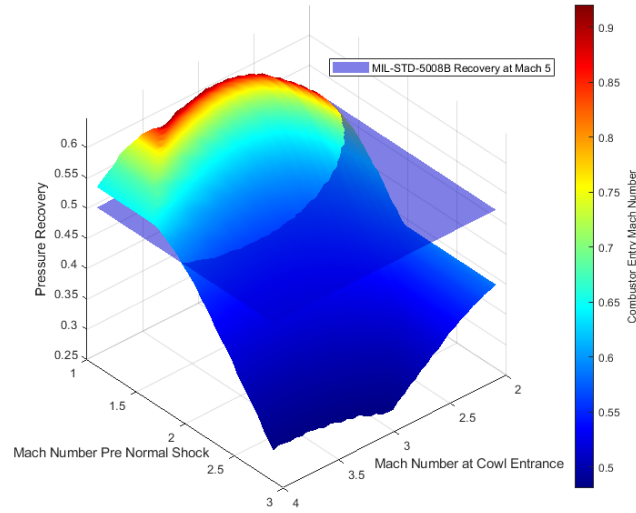


Figure 14. Pressure Recovery Surface as Function of Mach Numbers at Cowl Entrance and Pre-shock

Table 5

External Inlet Data – SI Units

M_{∞}	5
M_{cowl}	3.55
M_{pre-NS}	2.1
$M_{combustor}$	0.5127
γ	1.4
θ_1	5.061
θ_2	5.619
θ_3	6.2610
β_1	15.121
β_2	16.898
β_3	18.995
β_4	22.595
β_5	26.599
β_6	32.109
Recovery	0.5127
$T_{21-overall}$	5.644
$P_{21-overall}$	243.4
$\rho_{21-overall}$	43.121
$CR_{Kantrowitz}$	1.451

The next step was to determine the overall lengths and heights along the inlet. For this, SolidWorks 2025 was used to sketch the design and constrain it using the deflection and shock angles obtained from the inlet analysis to make sure all the shocks impinged at the cowl lip as well as constraining the cowl height so that the first internal reflected shock impinged at the end of ramp 3. Having these constraints as functions, the total inlet height could be varied, and the rest of the dimensions were scale accordingly. After various iterations during CL (based on capture area required and propulsion system integration) the inlet height was defined as 2 m.

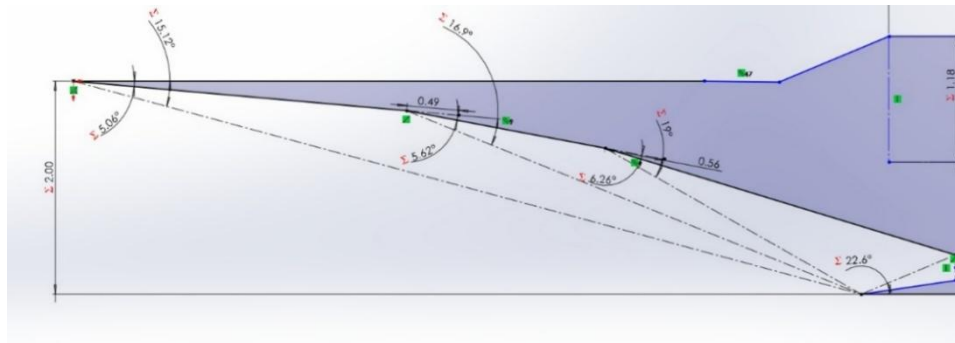


Figure 15. Inlet Constrained Sketch

For the internal dimensions, these were dominated by the throat height for the internal inlet. The isolator and combustor lengths were determined using the relations from section 2.2.6. For the Area ratio required to have subsonic flow at the exit of the combustor, the method from section 2.2.5 was employed with desired Mach Number of 0.8 at the exit. This Mach number was selected as to have a buffer if the flow was choked at other conditions. The results for this analysis are summarised in Table 6. To obtain a subsonic combustor exit Mach number, the combustor is required to diverge by 1.347 times the entry area to avoid a choked condition due to heat addition. The combustor divergent geometry is consistent with the reviewed literature.

Table 6
Combustor Geometry Results

$\bar{r}$	0.375
$\bar{U}$	2.045
A	1.347

With the geometry of the burner defined, the subsonic flow must be accelerated to Mach 1 so it can then be supersonically expanded to the freestream. To determine the throat area at the nozzle, the isentropic area relation was used using $M = 0.8$. With the throat defined, the expansion ratio was calculated using the exit under-expanded conditions mentioned in 3.1.2 and using the nozzle throat, the nozzle exit height was calculated. It is imperative to emphasize that the geometry was determined at the design point to obtain the volume of the engine for PS but in practice, the design must account for variable geometry at the inlet ramps, inlet throat as well as the nozzle throat to account for changes in the flight conditions. Additionally, the nozzle geometry would be best determined using exact numerical solutions such as method of characteristics as the isentropic relation was used as an approximation. The final engine geometry is shown in Figure 16. The final volume a single engine module (1 Turbofan + 1 Ramjet) was found to be 20.91 m^3 and was calculated by extruding the sketch for a width of the F100-PW-229 with an additional allowance of 10%. This was done to have the estimate k_{ve} for the engine which is the total volume of the engine divided by its thrust at sea level which is used to scale the engine based on the T/W ratio required at take-off.

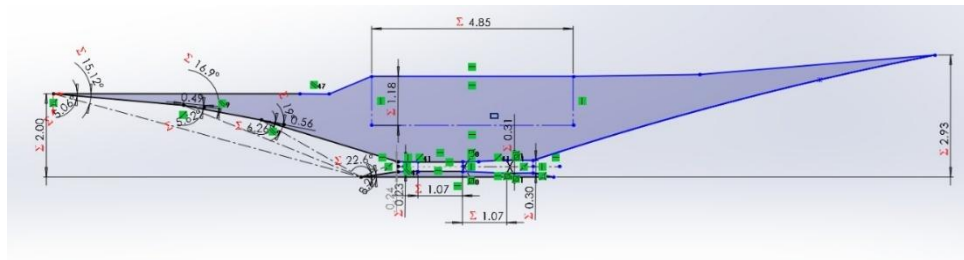


Figure 16. Final Engine Dimensions.

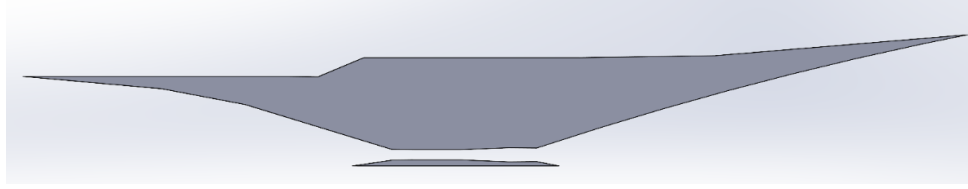


Figure 17. Final Engine Geometry.

3.1.5 Engine Weight Estimation

For the engine's weight estimation, as these are custom configurations which vary on a study-to-study basis and due to their classified nature no available weights databases to estimate relationships were found, the hypersonic vehicle study by Pegg et Al [1] was used to find a ratio of the ramjet weight and fuel system to the turbojet weight. Using the F100-PW-229's weight of 1735 kg, a single engine module would have a weight of 2933.12 kg.

$$W_{Engine\ Total} = W_{F100}(1 + 0.07526 + 0.6153) \quad (27)$$

3.2 Parametric Sizing

For parametric sizing **Table 7** summarizes the inputs for the analysis. The inputs were obtained by suggestions by Czysz et al ([15] Chapter 16), Chudoba et al [25] and by the propulsion analysis.

Table 7

HC Inputs

	Hydrocarbon	Hydrogen
Type	1-WB, 0-BB	
$I_{str} \left[\frac{kg}{m^2} \right]$	21	
f_{sys}	0.2	
$C_{un}[kg]$	2000	
$f_{md}[kg]$	1000	
$W_{crpv}[kg]$	70	
$W_{pay}[kg]$	4536	
μ_a	0.1	
N_{crw}	1	
$W_{crw}[kg]$	133	
$W_{engine}[kg]$	2993	
$V_{eng}[m^3]$	20.91	
E_{TW}	4.5	
$k_{ve} \left[\frac{m^3}{kg} \right]$	0.0016	
$\rho_{ppl} \left[\frac{kg}{m^3} \right]$	804	74.6
k_{vv}	0.01	
k_{crw}	2	
$\rho_{pay} \left[\frac{kg}{m^3} \right]$	140	
M_{cruise}	5	
$C \left[\frac{kg}{Ns} \right]$	6.52E-05	2.49E-05
$R [nm]$	3500	
$V_a \left[\frac{m}{s} \right]$	90	
$C_{Lmax-landing}$	1.15	
S_f/L^2	0.015	0.03
$h_{transonic}[ft]$	37000	

3.2.1 Body Type and Fuel Trade Study

Each body type possesses different aerodynamic characteristics (critical for performance), k_w relationships which influence the weight of the structure and the required amount of fuel to complete the trajectory. Fuel is also linked to its density which in turn affects the required volume to be stored affecting the converged results. To assess these interconnected parameters, the HC code was executed using the same inputs but varying the Type (WB, BB) as well as the propulsion and fuel characteristics based on hydrocarbon and hydrogen characteristics. The I_{str} was set at 21 kg/m^2 for all configurations per Coleman's recommendation for demonstrators [18]. The design points are converged values with the lowest MTOW to the right of the stall constraint. Figure 18 shows the general MTOW trends for all vehicle types. As can be seen, hydrocarbon vehicles tend to be significantly lighter than their hydrogen counterparts, which initially would seem counter intuitive given that due to the higher energy density of hydrogen it would result with less mass for the same energy required compared to hydrocarbon, however, hydrogen's required volume is the key factor that ends up driving the weights up for these vehicles.

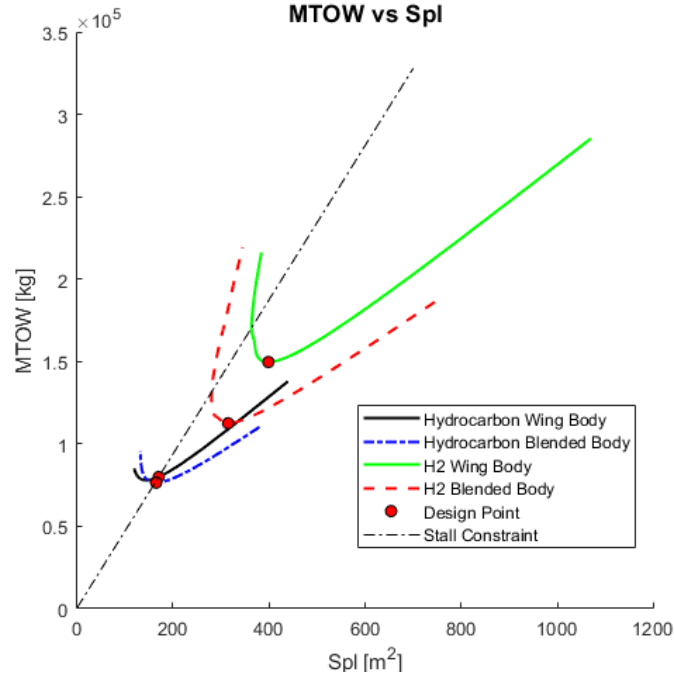


Figure 18. HC Design Space for Body and Fuel Types

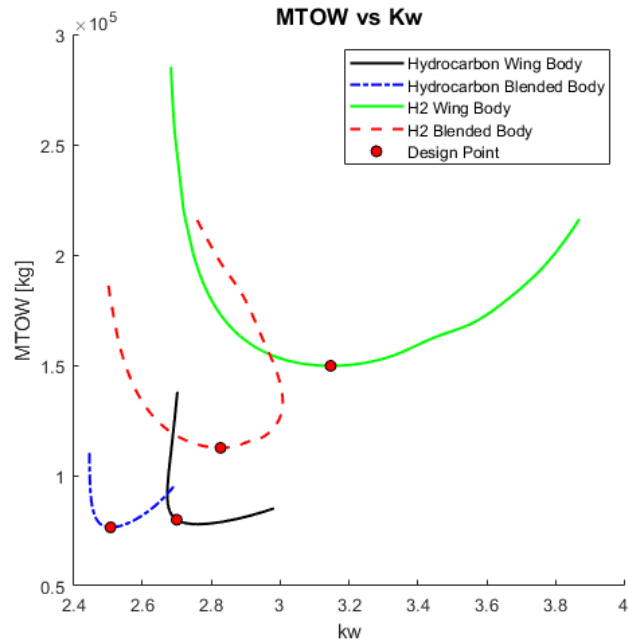


Figure 19. MTOW vs Kw

As shown in Figure 20 the volume requirement goes up, so does the converged Tau values which result in higher k_w and hence higher S_{wet} , as shown in Figure 21. Higher S_{wet} require more structural weight per the I_{str} definition, yielding overall higher dry weights for hydrogen vehicles as shown in Figure 22.

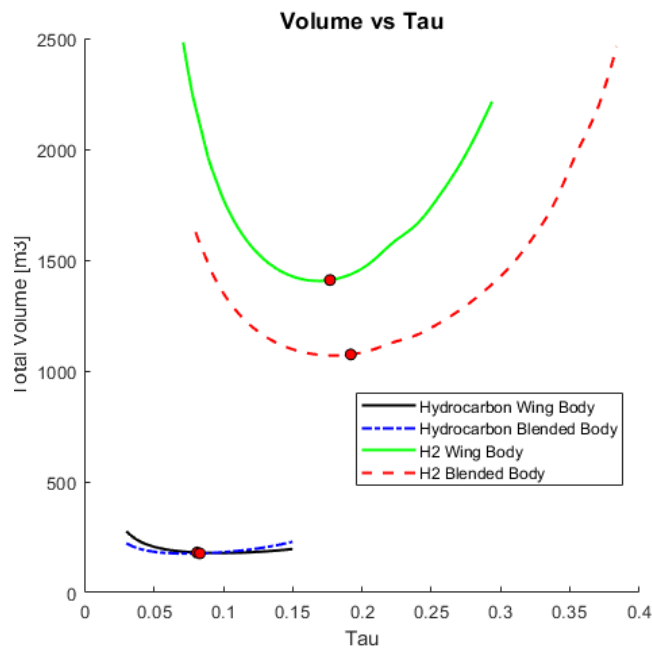


Figure 20. Total Volume vs Tau

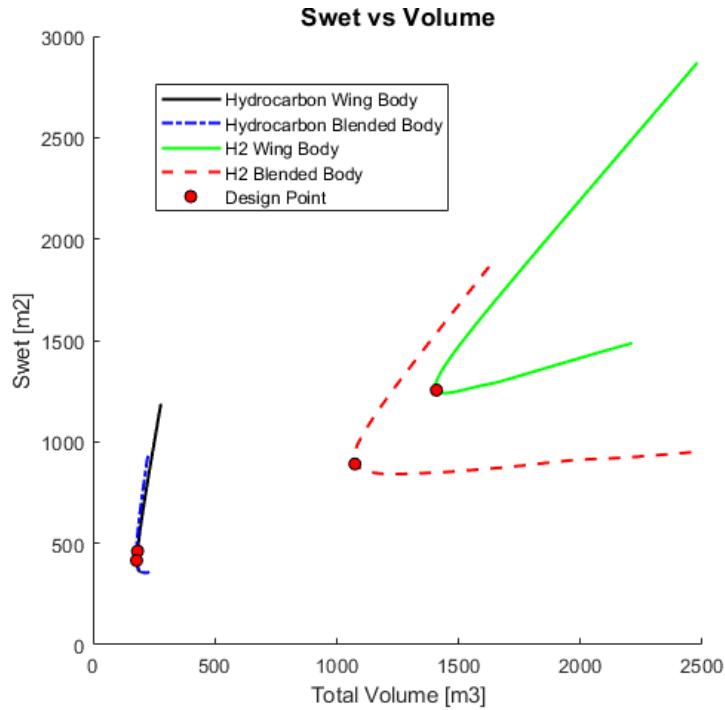


Figure 21. Wetted Area vs Total Volume

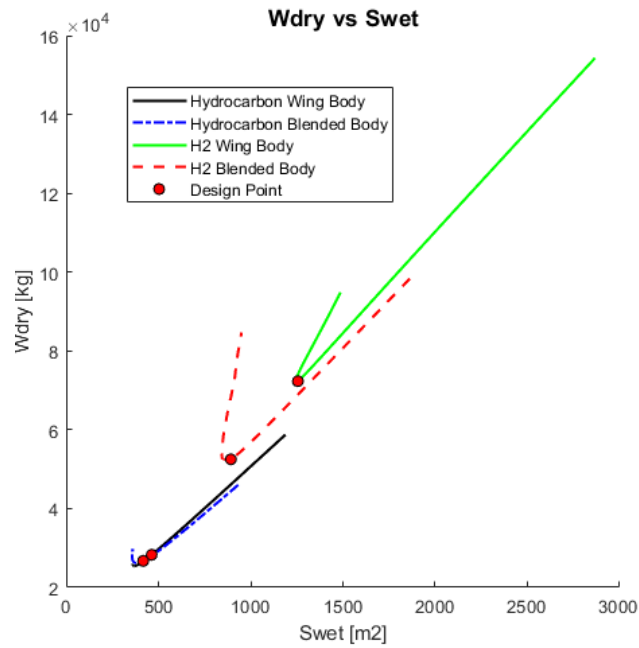


Figure 22. Dry Weight vs Wetted Area

Furthermore, although the dry weight is increased, hydrogen vehicles do present lower fuel fractions over their hydrocarbon counterparts as shown in Figure 23. One would expect that the lower fuel mass evidenced by the lower fuel fraction required would compensate for this structural weight increase due to increased S_{wet} , however, the increased volume also increases the cross-sectional area which in turn requires higher thrust for transonic acceleration, with

hydrogen vehicles requiring up to 8 turbofan engines (more engine weight) compared to 3 for hydrocarbon vehicles. This, paired with higher dry weights yields the higher MTOW for hydrogen vehicles. Additionally, even though hydrogen vehicles have lower fuel fractions, the higher Tau values for hydrogen vehicles also results in poorer aerodynamic performance (lower LDmax) as shown in Figure 24, making them consume more fuel during cruise (higher drag requiring higher thrust and hence higher fuel consumption).

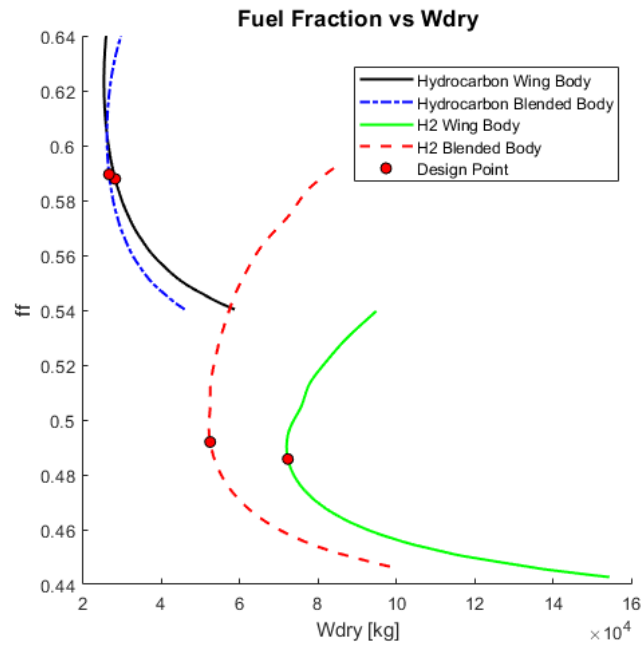


Figure 23. Fuel Fraction vs Dry Weight

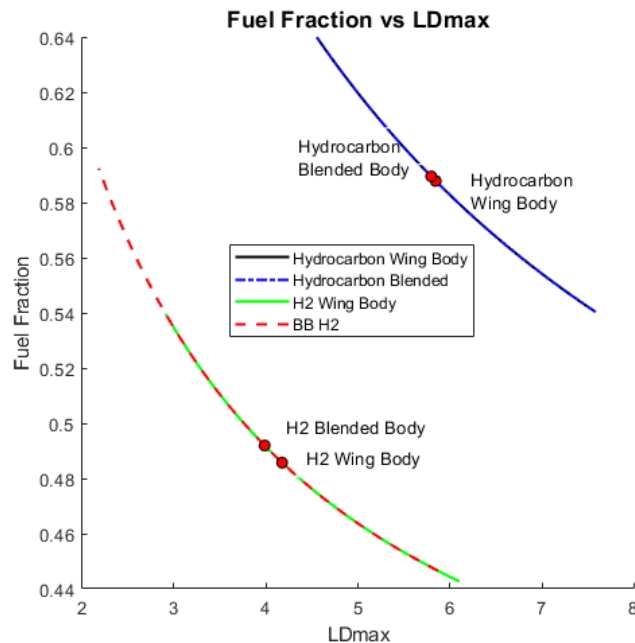


Figure 24. Fuel Fraction vs LDmax

Table 8

Fuel and Body Type Trade Study Results

	BB	WB	BB H2	WB H2	Hydrocarbon	H2	BB
	Hydrocarbon	Hydrocarbon			Comparison	Comparison	Comparison
MTOW [kg]	76,490.5	79,849.4	112,523	149,715	-4.206	-33.052	-47.107
S_{pl} [m ²]	165.44	170.455	315.137	398.808	-2.942	-26.550	-90.484
τ	0.083	0.081	0.192	0.177	2.469	7.8125	-131.325
k_w	2.5	2.7	2.827	3.146	-7.407	-11.284	-13.08
V [m ³]	176.627	180.644	1074.11	1409.68	-2.223	-31.241	-508.123
S_{wet} [m ²]	414.97	460.92	891.038	1255.01	-9.969	-40.848	-114.723
W_{dry} [kg]	26731.2	28283	52501.2	72326.9	-5.486	-37.762	-96.404
LD_{max}	5.79	5.844	3.98	4.176	-0.924	-4.924	31.260
ff	0.58949	0.5878	0.4919	0.4857	0.287	1.260	16.554
Fuel [kg]	45,090	46,935	55,350	72,716	-3.931	-31.375	-22.753

On a specific comparison for each fuel type, BB configurations present the lowest MTOW compared to their WB counterparts mainly due to the lack of blending between the fuselage and wings which results in a higher wetted surface area, which is evidenced by the k_w equations by Czysz ([15] Chapter 16). As discussed earlier, S_{wet} surface areas are linked to higher structural weight. For hydrocarbon vehicles, the difference in MTOW is not large at -4.2% between BB and WB, which also results in small differences for total volume (-2.223%) and dry weight (-5.486%) which are somewhat proportional to their Tau difference of 2.469%. The latter has a slight impact on the achievable LD_{max} at cruise, in which WB show better performance (+0.924%), reducing the WB fuel fraction by 0.287%. The slight increase in LD_{max} and slight reduction in the fuel fraction do not provide the expected fuel savings as the WB ends up being a heavier configuration meaning that although it has better aerodynamic performance it requires an additional 1,800 kg of fuel.

Moving on to hydrogen vehicles, the differences between BB and WB are more pronounced. For the MTOW, BB presents a -33% difference compared to WB. This significant weight reduction is proportional to their total volume (-31.2%) and wetted surface area (-40.8%) differences which seem to be greater when using such low-density fuels as these differences are not large for hydrocarbon. Additionally, when it comes to performance, hydrogen vehicles follow the same trend for BB vs WB for hydrocarbon vehicles where WB have higher LD_{max} and lower fuel fractions. However, the difference for these factors is increased to 4.9% for LD_{max} and 1.2% for fuel fraction which at first would still seem low but considering how low-density fuel types can amplify the impact variations in Tau and k_w have on the weights and volumes of a given configuration (as using hydrogen produces converged solutions with higher Tau values which then amplify the nonlinear behaviour of the Kw relationships for WB and BB which does not occur for hydrocarbon which yield converged solutions with lower Tau values) results in the WB actually requiring about 17,300 kg more fuel than BB.

3.2.2 Final Design Point

Based on the PS analysis, the hydrocarbon BB was the chosen configuration for its lower size (weights, volumes, and surfaces) and less fuel required which would result in a more cost-effective vehicle which is critical for the viability of its development program. The final breakdown of the design is shown in Table 9. The weights and volumes breakdown was obtained using the equations from Czysz et al ([15] Chapter 16) included in APPENDIX 12. These sizing results were used as inputs for the CL phase to determine the planform and airframe shape, component layout, centre of gravity and propulsion integration.

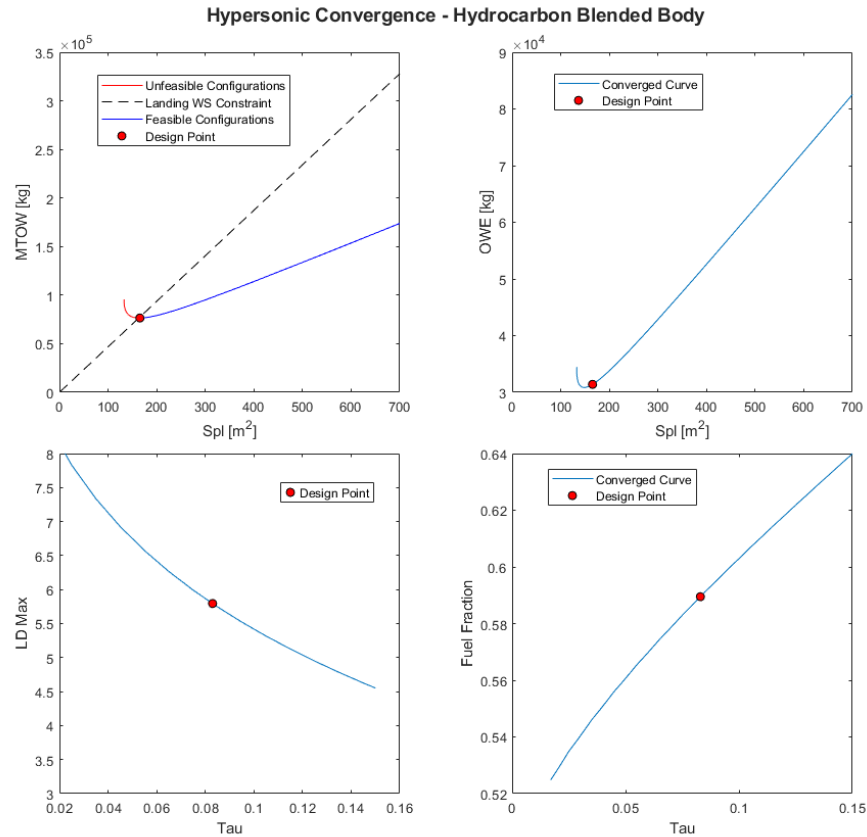


Figure 25. Final Design Point Solution Spaces

Table 9

Hydrocarbon BB Design Results

τ	0.083
$S_{pl}[m^2]$	165.44
$S_{wet}[m^2]$	414.96
k_w	2.5
$R_{cruise}[nm]$	3500
M_{cruise}	5
ff	0.5895
TW_{TO}	0.4225
$\left(\frac{W}{S_{TO}}\right)_{TO} \left[\frac{kg}{m^2}\right]$	462.33
LD_{cruise}	5.7953
$V_{total}[m^3]$	176.626
$V_{fuel}[m^3]$	56.082
$V_{sys}[m^3]$	28.260
$V_{eng}[m^3]$	51.117
$V_{void}[m^3]$	1.76
$V_{payload}[m^3]$	32.4
$V_{crew}[m^3]$	2
$V_{crpv}[m^3]$	5
$MTOW[kg]$	76,491
$OWE[kg]$	31,4
$W_{str}[kg]$	8,784
$W_{sys}[kg]$	8,346
$W_{eng}[kg]$	7,17
$W_{payload} [kg]$	4,536
$W_{fuel} [kg]$	45,090
$\frac{W_{dry}}{MTOW}$	0.411
N_{eng}	3
N_{crew}	1

3.3 Configuration Layout

3.3.1 Vehicle Geometry

The vehicle geometry process consisted of developing the shape of the vehicle while matching the planform area and total volume values obtained from PS. In the first step, the planform shape was defined. For this an initial sweep angle of 78° for the forward part of the airframe was used with a 74° sweep angle starting at 8.73 m to increase the span of the vehicle with the aim of improving its aerodynamic characteristics. Once the planform shape was defined, 8 cross-section sketches were created and then lofted to provide the volume of the airframe. Using this airframe, a shell was created, and the remaining internal volume was the divided to allocate the different systems.

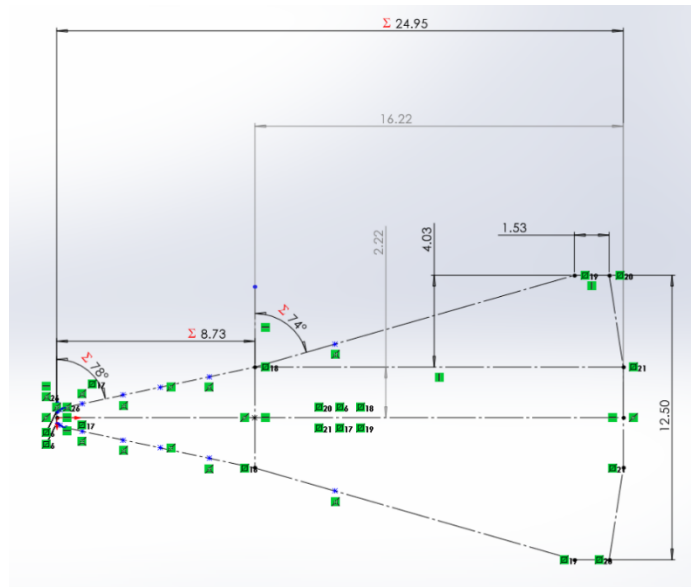


Figure 26. Vehicle Planform Sketch

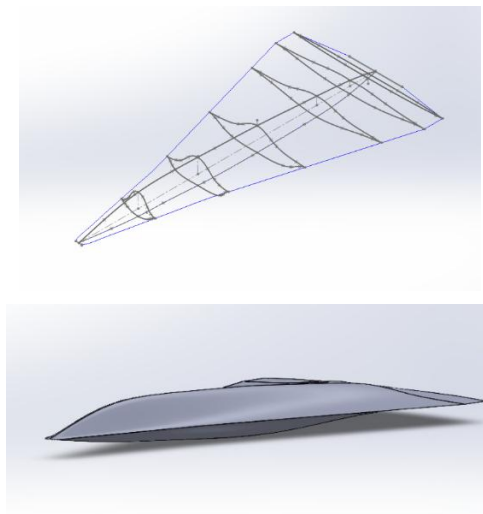


Figure 27. Final Airframe Geometry

The next step consisted of integrating the propulsion system defined in section 3.1. To properly integrate the inlet, the external ramp had to be truncated so it would fit in the underside of the airframe. And iterative process for positioning the engine yield the required expansion height and capture area described in that same section. The final geometry of the vehicle (excluding control surfaces) is shown in Figure 28 and a dimensions summary in Table 10.

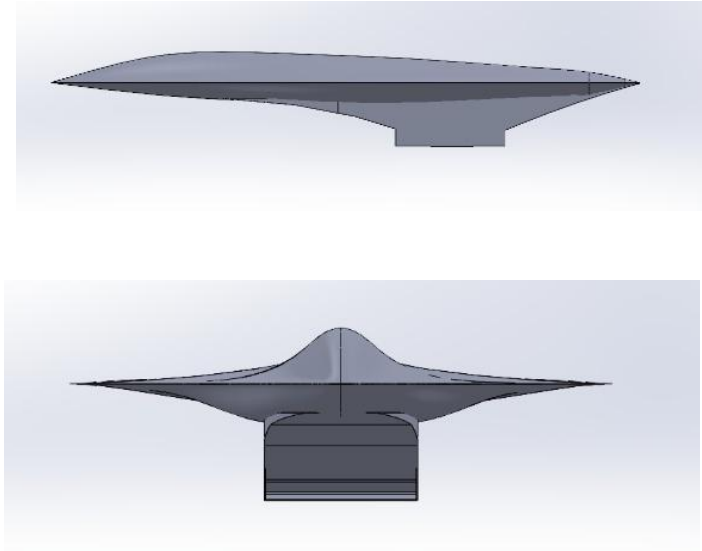


Figure 28. Final Airframe and Propulsion Integration

Table 10

Airframe Geometry

<i>length</i> [m]	24.95
<i>b</i> [m]	12.5
<i>S_{pl}</i> [m ²]	165.44
<i>AR</i>	0.9444
<i>C_{tip}</i> [m]	1.54
λ	0.062
<i>MAC</i> [m]	16.69
<i>A_{LE-eff}</i> [deg]	76.3
$\left(\frac{t}{c}\right)_{average}$	0.04

3.3.2 Weights and Balance

The centre of gravity (CG) was determined using assembly mode in SolidWorks 2025 by applying the estimated weights obtained from PS. The breakdown of the system weights was further refined using the methods described in APPENDIX 13 by Cranfield University [40,41], splitting the total weight into forward, centre, and aft sections using system-oriented placement for the controls, fuel, and electrical systems. The systems weights and distribution are presented in APPENDIX 13 and the general internal layout shown in Figure 29. Furthermore, the fuel tank was split into 3 sections to allow fuel transfer throughout the trajectory by taking into consideration the shift in CG that occurs with increasing speeds. Overall, the MTOW CG was aimed to lie in the geometric centroid of the wing and with the fuel tanks allowing a forward and rearward shift with ample margin as needed. The physical CG locations as well as the forward and rearward limits were obtained by “emptying” the respective fuel tanks (setting their masses to 0). Additional analysis is required from stability and control to assess the actual CG limits throughout the trajectory as it was not performed in this study. As can be seen with the current layout, the maximum CG travel that can be achieved by fuel transfer (full forward tank vs full rearward tank) is of 11% of the vehicle length or 16.5% of the MAC.

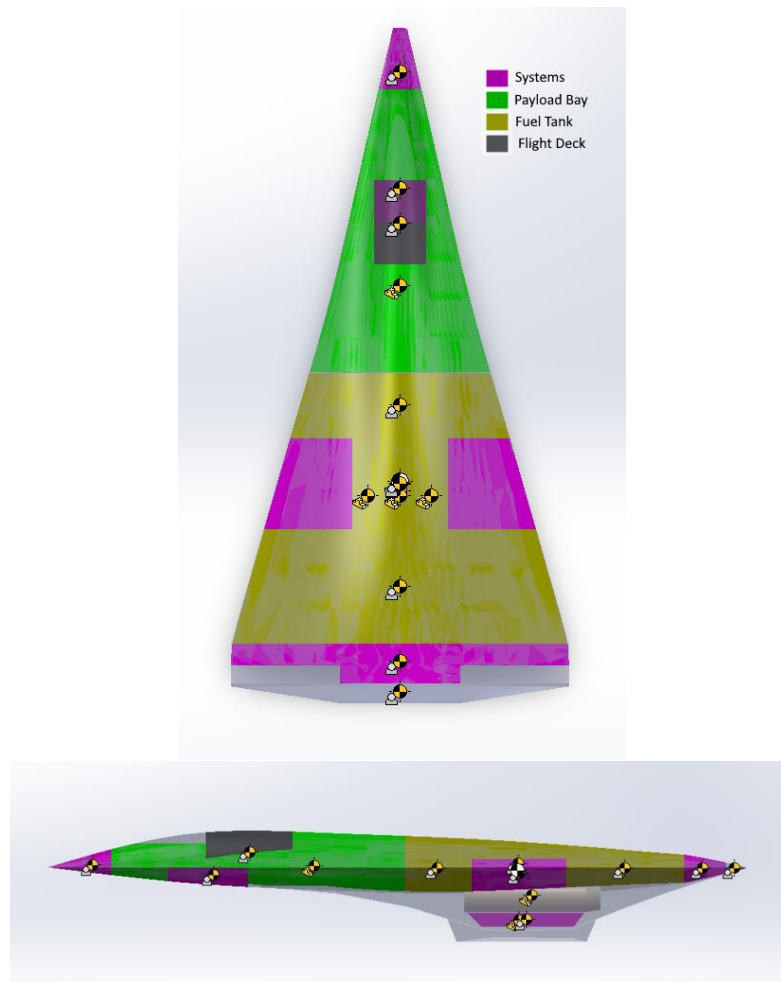


Figure 29. Final Vehicle Layout

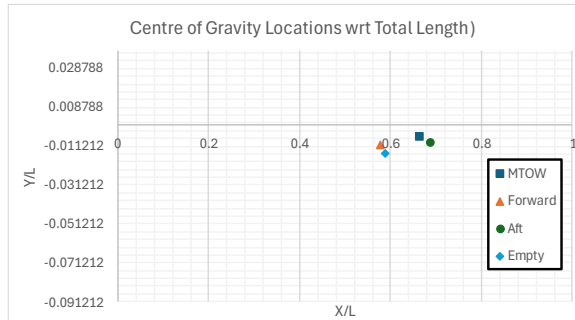


Figure 30. Centre of Gravity Locations with Respect to Total Length

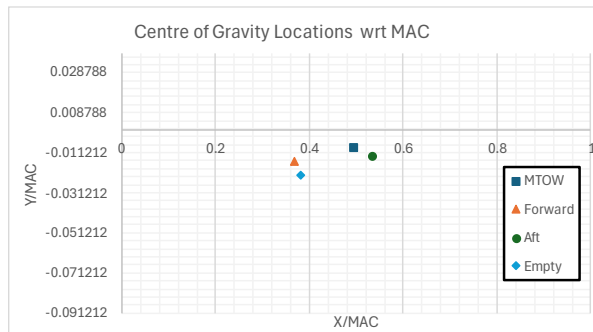


Figure 31. Centre of Gravity Locations with Respect to MAC

Table 11

Centre of Gravity Locations Results with Respect to Total Length

Vehicle	X/L	Y/L	Z/L
MTOW	0.662525	0	-0.00561
Forward	0.577555	0	-0.01042
Aft	0.688176	0	-0.00882
Empty	0.586774	0	-0.01483

Table 12

Centre of Gravity Locations Results with Respect to MAC Leading Edge

Vehicle	X/MAC	Y/MAC	Z/MAC
MTOW	0.495438955	0	-0.00838
Forward	0.368434047	0	-0.0155
Aft	0.533780059	0	-0.0131
Empty	0.382212882	0	-0.0221

3.3.3 Control Surface Sizing

For control surface sizing, the volume coefficient method was used using the estimated coefficients for the fin and elevons for the XB-70 and Concorde. The values were obtained from a previous study by the author of this paper [42] as shown in Table 13 which were calculated using the total sum of elevon areas and the fins (for the case of the XB-70). Using the estimated CG locations for the forward, MTOW, empty and aft positions, it was determined that the critical sizing condition for the elevons would be the MTOW position to ensure enough elevon control power to perform the take-off rotation. For the vertical tails, it was determined that the critical sizing condition would be aft location since this would provide the lesser directional control power specially under crosswind conditions. Using SolidWorks 2025, a preliminary sketch for the fin and elevons was created which was then used to vary their dimensions while estimating the aerodynamic centre location for these surfaces, this allowed to determine the moment arm with respect to the CG locations. Through an iterative process, the surface areas and dimensions were determined as shown in Table 14.

$$\bar{V}_{DICE} = \frac{S_{tail} l_{tail}}{S_{pl} b} \quad (28)$$

$$\bar{V}_{LoCE} = \frac{S_{elevon} l_{elevon}}{S_{pl} MAC} \quad (29)$$

Table 13
High Speed Vehicle Control Surface Volume Coefficients [42]

Vehicle	$\bar{V}_{DICE}$	$\bar{V}_{LoCE}$
Concorde	0.059644	0.033342
XB-70	0.034017	0.04442

Table 14
Control Surfaces Geometry Summary

Control Surface	Number	C_{root} [m]	C_{tip} [m]	$Span$ [m]	l_{arm} [m]	Area Per Surface [m ²]	Total Area [m ²]
Elevon	2	2.18	1.56	4.44	23.25	8.43	16.86
Fin	2	4.19	1.47	2.35	22.42	6.22	12.44

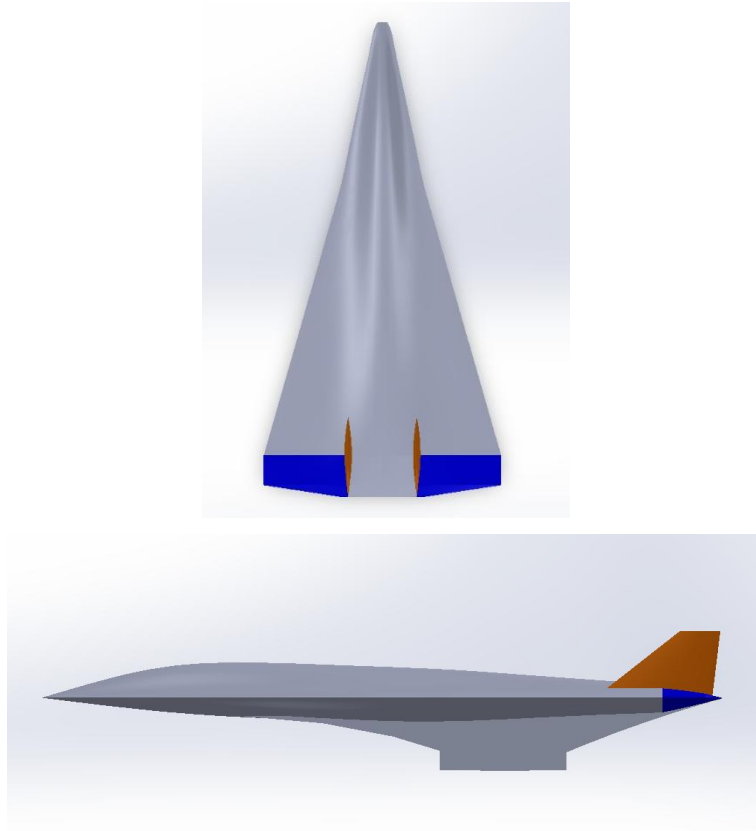


Figure 32. Final Control Surfaces

3.4 Configuration Evaluation

3.4.1 Aerodynamics

Starting with subsonic/transonic aerodynamics, the process described in section 2.2.8 was followed. Using OpenVSP, the geometry obtained in CL was generated as shown in Figure 33. This wing was created using a biconvex aerofoil profile per the recommendation by Hamm et al [4] using thickness-to-chord ratios of 0.06, 0.05, 0.01 for the root chord, cranked wing start location, and wing tip respectively. The fins used a biconvex aerofoil with a thickness-to-chord ratio of 0.05. The fuselage geometry was generated using a fuselage component with a scaled sideview of the airframe as reference. Using this model the C_{D0} was calculated at different Mach numbers. The engine was not included in the model as its geometry could not be modelled properly and because the form factor methods were mostly applicable to streamlined bodies. Hence, the influence of the engine on drag must be analysed in the later stages of the design process with high-order CFD methods.

Furthermore, the C_{Dw} was estimated using the wave-drag module for speeds ranging from Mach 1.2 to 5. The absence of engine geometry was not critical since, as suggested by Hamm et al [4] and MCAIR [5], the engine does not account for wave-drag as it is assumed the incoming flow is going through it. As suggested by Coleman [18], one can assume that wave-drag begins at Mach 0.9 with a peak at Mach 1.2, hence, a linear interpolation between zero wave-drag at Mach 0.8 and the peak at Mach 1.2 was used to determine the wave-drag at Mach 0.9. As shown in Figure 34, the variation for the “fixed drag” (Parasite plus wave-drag, excluding induced drag) shows the expected variation with an exponential increase in the transonic region and then decaying with increase in Mach.

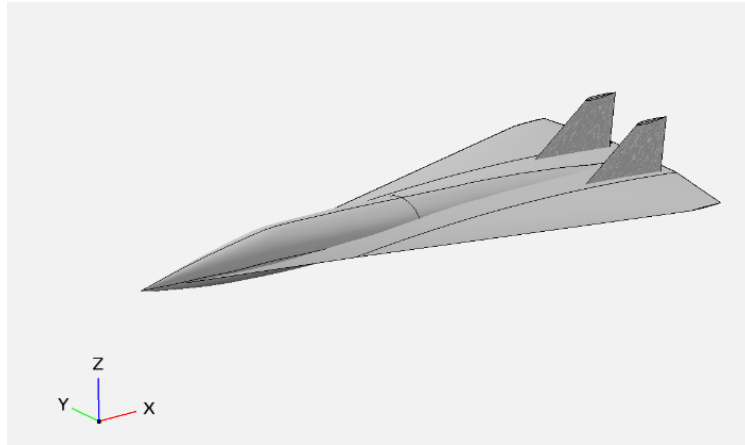


Figure 33. OpenVSP Geometry

Table 15
Parasite and Wave-drag Variation with Speed

Mach	C_{D0}	C_{Dw}	$C_{D\ Total}$
0	0.00592	0	0.00592
0.2	0.00542	0	0.00542
0.4	0.00517	0	0.00517
0.6	0.00513	0	0.00513
0.8	0.00517	0	0.00517
0.9	0.00532	0.00255	0.00787
1.2	0.00511	0.0102	0.01531
2	0.00505	0.0079	0.01295
3	0.00555	0.0063	0.01185
4	0.00562	0.0058	0.01142
5	0.00559	0.0055	0.01109

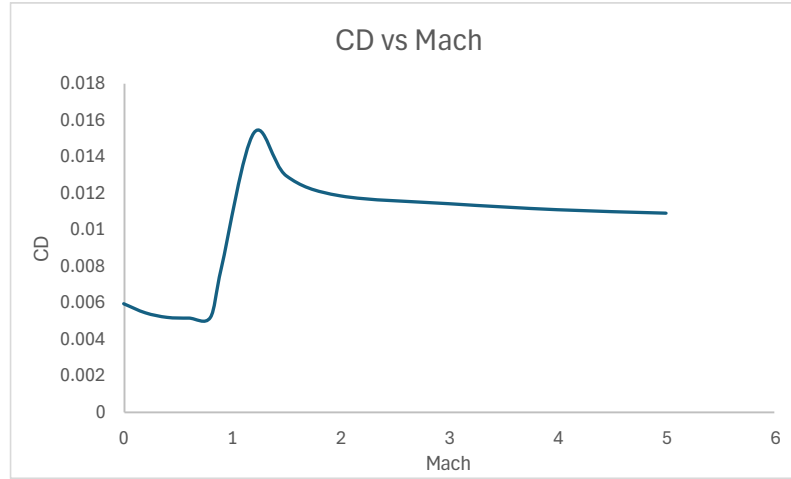


Figure 34. CD vs Mach Number

For induced drag, the efficiency factor was estimated to be 0.453 following the subsonic approach (classical) and the K factor was calculated using the method for high-speeds, both provided by Raymer [21] using the effective leading-edge sweep angle of 76.3° estimated following the method by Hopkins et al [35] and the AR of 0.9444.

Table 16
Parasite and Wave-drag Variation with Speed

Mach	K
0 – 0.9	0.74402
1.2	0.64977
2	0.67159
3	0.72463
4	0.83963
5	0.99009

For the subsonic lift-curves, the procedures by Abraham et al [32] in parallel with Polhamus [33] were followed. For the first approach, the potential flow $C_{L\alpha}$ was obtained using VSPAERO. The C_{L-P} values were then corrected for Mach numbers 0.2 to 0.9 using the Prandtl-Glauert correction factor, as the potential flow solver did not account for it and without it all lift-curves would have remained invariant. For C_{L-V} , Kuchemann's equation was used and then corrected for compressibility effects using the approach given by Polhamus [33]. As shown in Figure 36, the results by both methodologies show good agreement at low Mach numbers (0 – 0.6), however, the corrected approach by Abraham et al shows higher estimates for Mach numbers 0.8 and 0.9 at higher AoA, where the vortex-lift dominates. As it is reasonable that compressibility tends to improve the lift performance, a regression curve was obtained for each Mach to better capture the behaviours from both methods. The final low-speed lift-curves are shown in Figure 37. As can be seen, due to the reliance on vortex-lift for delta wings, where at lower angles of attack it is small, as well as the thin thickness-to-chord ratio required for high-speed vehicles, the lift-coefficient values remain relatively

low compared to conventional wings. Additionally, the $C_{L\alpha}$ increases with increases in Mach number which follows the same behaviour for the SANGER II presented by Weiland et al [43] (see APPENDIX 14). Having the C_L and C_D values, the lift-to-drag ratio values were calculated. The LD_{max} show an increasing behaviour from Mach 0 to 0.6, due to the reduction of C_{D0} at higher Reynolds number, after which decreases due to compressibility effects and wave-drag.

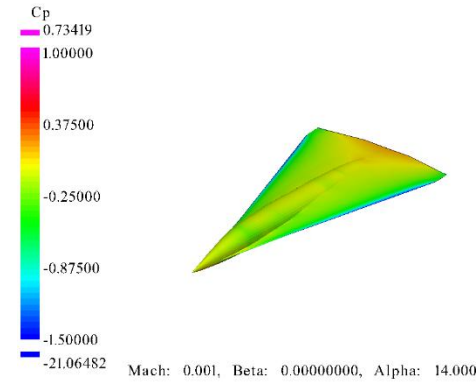


Figure 35. VSPAERO Viewer Sample

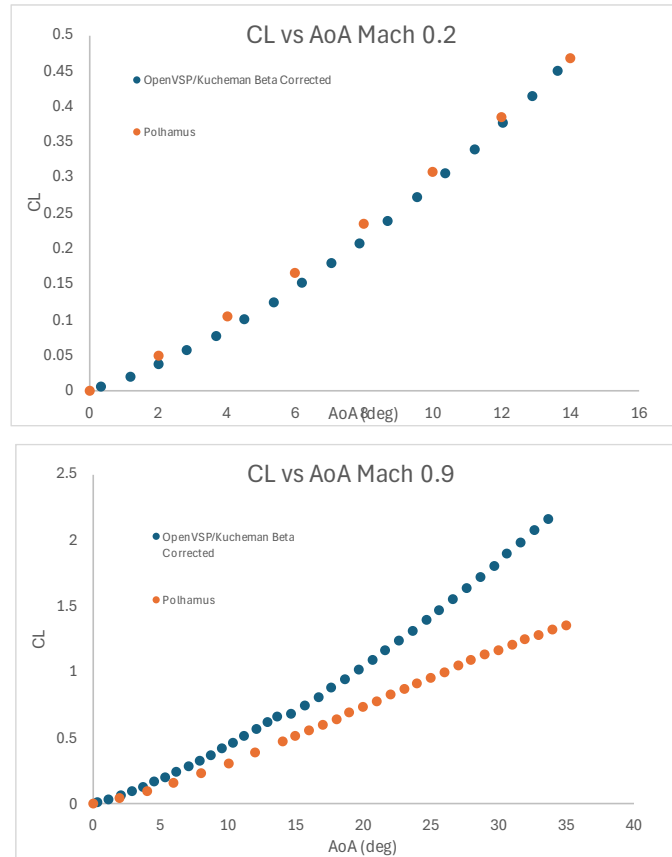


Figure 36. Abraham vs Polhamus Methods Comparison

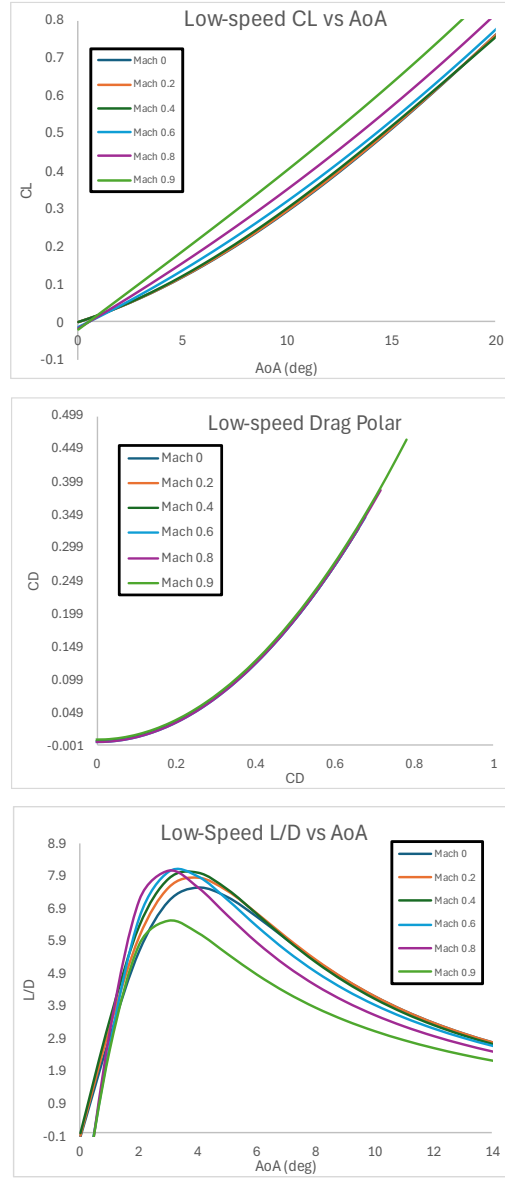


Figure 37. Low Speed Aerodynamic Data

For high speed aerodynamics, the methods found in ESDU 70012 [36] and 90013 [44] were employed using the AR and taper-ratio values obtained in CL. The final high-speed lift-curves are presented in Figure 38 including the drag polar and lift-to-drag ratio variation with Mach number. As can be seen, the $C_{L\alpha}$ decrease with increasing in Mach number, in a similar behaviour to SAENGER II [43], mainly due the narrowing of the generated Mach cone (which also reduces vortex-lift) and the reduction of the influence of the upper surface as described by the Newtonian method.

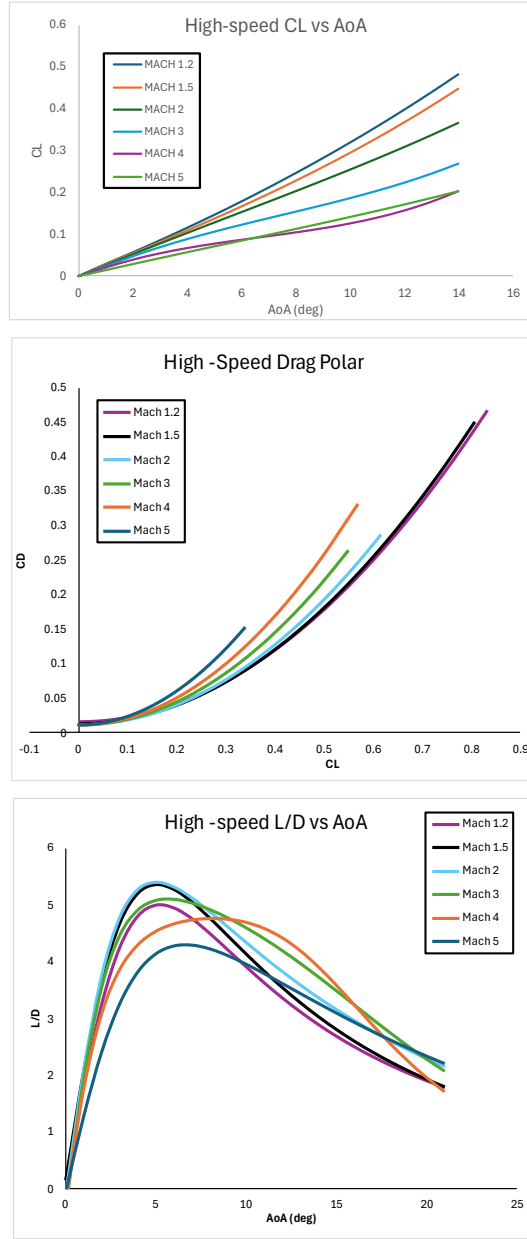


Figure 38. High-speed Aerodynamic Data

The results for high-speed LD show a significant decrease in LD_{max} from the predicted value used for PS. This was expected considering the semiempirical data for PS [18] consisted of regression curves based on multiple experiments, which potentially overpredicts the LD_{max} . The LD_{max} of 4.3 was still deemed reasonable for the slenderness of this vehicle also considering that its shape was not optimized to maximize aerodynamic performance. Additionally, the thrust required for cruise using this new LD_{max} , was calculated to be 143.9 kN, which for the thrust calculated in section 3.1.2 still yields an excess thrust of 44.1 kN, providing an ample margin for additional engine drag considerations.

3.4.2 Performance & Trajectory Analysis

For the take-off analysis, the aerodynamic results paired with a corrected TW0 for the 3 F100-PW-229 at max thrust were used to determine the CL required to meet the S_{TOG} requirement. The results summarised in Table 17, show that an AoA of 23.5° is required to produce a C_L of 0.933 which gives an STOG of 4600 ft, therefore meeting the take-off distance requirement.

Table 17
Take-off Analysis Inputs and Results

$\left(\frac{W}{S}\right)_{TO} [psf]$	94.692
$\left(\frac{T}{W}\right)_{TO}$	0.5187
$C_{L_{TO}}$	0.933
C_{D0}	0.00567
BPR	0.36
k_1	0.0447
k_2	0.92201
μ_G	0.03
$\rho \left[\frac{slugs}{ft^3}\right]$	0.00222126
$S_{TOG} [ft]$	4600

After the preliminary configuration was defined in CL, alongside its propulsion and aerodynamic data, it was evaluated in the trajectory module to determine if the aerodynamic and propulsion characteristics allowed the aircraft to reach Mach 5 at 80,000 ft. The data obtained in the previous sections was used to create look-up tables as functions of Mach number and altitude to be used for calculations at each energy level. Figure 39 shows the calculated MTC trajectory. As can be seen the vehicle follows along the constant Max q Limit as these where the locations in the Mach vs Altitude plane with the highest excess power with a dynamic pressure lower than 1000 psf. At about 42,000 ft, the aircraft performs a constant energy level climb to 53,900 ft, decreasing its Mach number while gaining height, after which accelerates through a small dive to 52,000 ft where it accelerates to Mach 2.5. At this point the aircraft initiates a what appears to be a constant dynamic pressure climb from to Mach 5 at 80,000 ft. Furthermore, the weight fraction for the ascend for MTC was 0.8248, yielding a fuel burn of 13,397 kg.

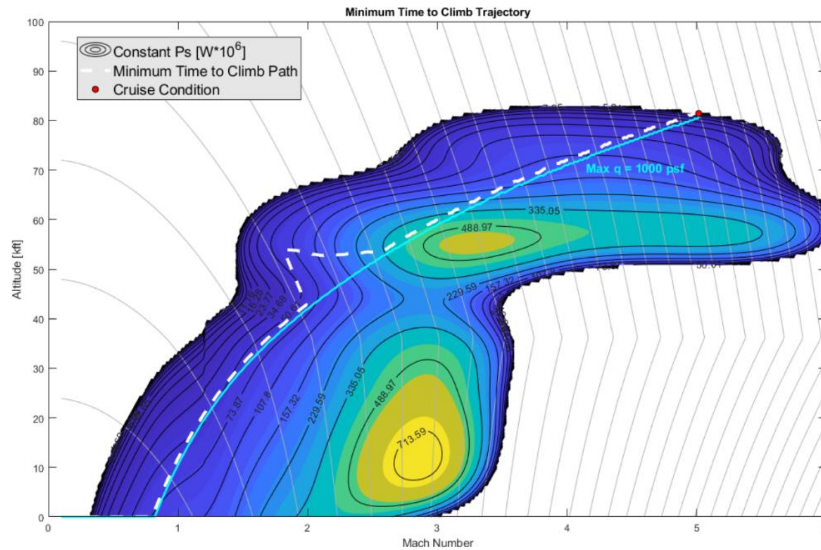


Figure 39. Minimum Time to Climb Path

Figure 40 shows MFC trajectory. This path looks similar the MTC, except for 2 constant Mach number climbs at 42,700 ft and 54,159 ft respectively, followed by a constant energy level climb at 67,987 ft to 74,000 ft. After this final climb, the vehicle slightly descends accelerating to Mach 4.1 after which it follows a constant dynamic pressure climb to 80,000 ft while accelerating to Mach 5. The execution of these 3 climb segments optimizes the trajectory gaining a slight fuel savings of 130.8 kg. This is small saving is expected since the contours were generated by operating at 100% throttle with afterburner on (for the turbofans) leaving a small margin to find lower fuel parameter locations. The trajectory results for the ascend are summarized in Table 18. In virtue of reducing the time the aircraft spends ascending to avoid unnecessary heating, the MTC trajectory results were chosen to continue with the cruise analysis.

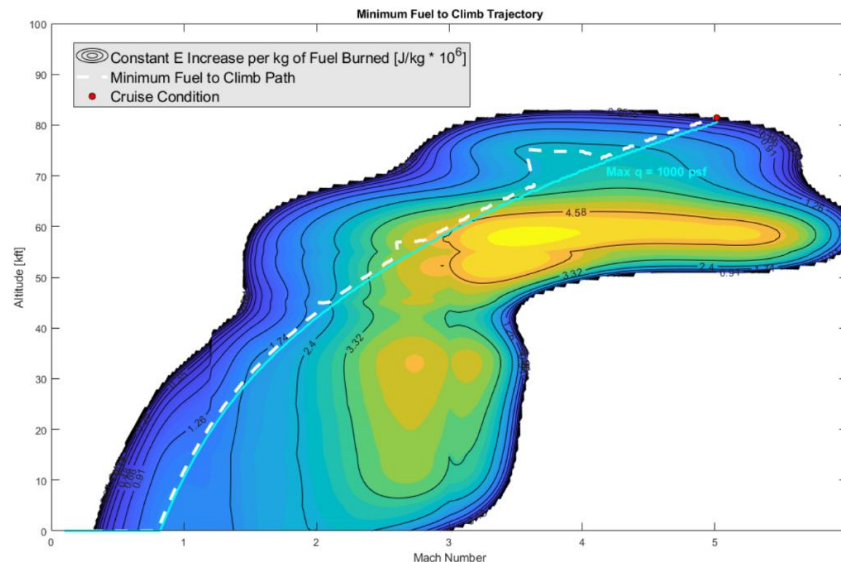


Figure 40. Minimum Fuel to Climb Path

Table 18
Ascend Trajectory Results

Trajectory	Weight Ratio	Fuel Burned [kg]
MTC	0.8248	13,397.8075
MFC	0.8266	13,267.0250

For the cruise analysis, the results from the previous sections was used to calculate the cruise using the Breguet Range equation for 3500 nm. The inputs and results are summarized in Table 19.

Table 19
Cruise Analysis Inputs and Results

Mach	5
Altitude [ft]	80,000
V [m/s]	1490.08
LD_{max}	4.3
R [nm]	3500
C [kg/N-s]	6.61E-05
$\frac{W_{end-cruise}}{W_{start-cruise}}$	0.51899
Fuel Burned [kg]	39,651

Finally, a detailed analysis for the descend was not performed. Instead a weight fraction of 0.98 was used as suggested by Roskam [23]. Multiplying the calculated ratios to find the total trajectory fuel burn yields a fuel burn of 44,606 kg, leaving about 393 kg of fuel remaining in the fuel tank. From this we can conclude that the aircraft can achieve Mach 5 at 80,000 ft and complete the 3500 nm cruise while still having 393 kg spare fuel. It is important to note that this margin is small as ullage fuel must be taken into consideration. Additionally, the lower LD_{max} for cruise does not impact significantly the results from PS as the additional fuel burned during cruise due to lower aerodynamic performance was compensated in PS since the ascend weight fraction was more conservative (0.679 compared to the calculated 0.8248 for MTC), which gave the aircraft more fuel in the sizing process.

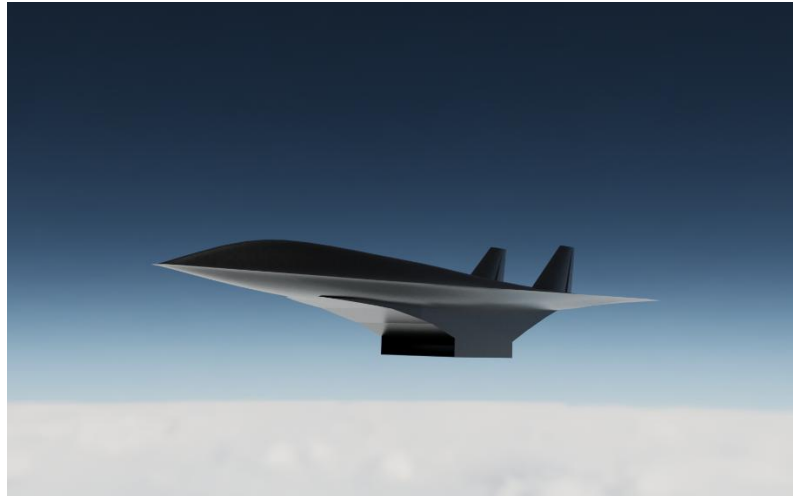


Figure 41. Final Hypersonic Vehicle Demonstrator Concept

4.0 Conclusions

4.1 Study Conclusions

- Regarding Propulsion, the ramjet cycle analysis was developed which allowed the study of the effects of using hydrocarbon fuel vs hydrogen. From this analysis, hydrogen exhibits 62.62% lower fuel consumption and a 6.69% increase in specific thrust compared to hydrocarbon fuel.
- The geometry of the TBCC was defined using tools applicable to conceptual design to develop the preliminary inlet, burner and nozzle geometry meeting the MIL-STD-5008B recovery specifications.
- During PS, the fuel and body type trade study showed better characteristics for the blended body hydrocarbon configuration, influenced by its lower converged k_w and volume required which produced lower wetted surface areas (compared to the other configurations) resulting in lower dry weights. The Blended Body Hydrocarbon configuration provided the lowest MTOW, dry weight, and fuel required for the same structural, systems, payload and mission characteristics.
- A benefit in aerodynamic performance was found for Wing Bodies for a given fuel type, as these configurations exhibited lower Tau values which improved their LD_{max} values by 4.29% and 31.26% (for hydrocarbon and hydrogen respectively) compared to their Blended Body counterparts. However, as Wing Bodies were overall heavier than Blended Bodies, these improvements did not show actual savings in the required fuel as they required 3.9% and 31.37% more fuel (for hydrocarbon and hydrogen respectively) compared to their Blended Body counterparts.
- Significant differences were found between Blended Body and Wing Body types when using hydrogen as fuel. The lower density of hydrogen resulted in these vehicles converging with higher Tau values which resulted in higher k_w values. Due to the high Tau values, the nonlinear behaviour of k_w was amplified in which a relatively small

differences of 7.8% and 11.28% (for Tau and Kw respectively) produced increases of 33%, 31.24%, 37.7%, and 31.37% for MTOW, volume required, dry weight, and fuel required respectively, for hydrogen Wing Bodies over hydrogen Blended Bodies. These differences were not large for the hydrocarbon vehicles as only increases of 4.2%, 2.22%, 5.48, and 3.91 for MTOW, volume required, dry weight, and fuel required respectively were found for Wing Body over Blended Body.

- Even though hydrogen provides a more efficient engine cycle, its lower density has a detrimental impact in the sizing of the vehicle resulting, for its smallest configuration (Blended Body), in 47 % higher MTOW, 96.4% higher dry weight, 508% more volume, and 22.75% more fuel required for the same mission compared to the Blended Body Hydrocarbon vehicle. All these factors add significant complexities to the development of the design as well as increased costs associated to manufacturing and fuel required which can put at risk the success of a program following this configuration. From this analysis, the hydrocarbon blended body was chosen as the preferred design.
- The conceptual design methods for aerodynamics, provided reasonable results that align to common behaviours for delta wings in both low-speed and high-speed regimes.
- Based on the results from PS, aerodynamics and propulsion, the vehicle meets the STOG requirement for take-off.
- The trajectory analysis for the ascend provided optimistic results for fuel consumption by calculating a weight ratio of 0.8248 for the MTC trajectory which yields a lower fuel burn for the ascend (13,397 kg) compared to the conceptual design estimate of 0.679 (24,524 kg).
- This overestimation for the ascend compensated for the decrease in aerodynamic performance during cruise, which for the vehicle developed in CL showed a LDmax of 4.3 when for PS its estimated value was of 5.79.
- Due to the full throttle condition for the trajectory, there was a small fuel savings of 393 kg for the MFC trajectory over the MTC trajectory. However, to reduce the aerodynamic heating, the MTC trajectory is preferred.
- The integration of the results from PS, aerodynamics, and propulsion provided a vehicle that can preliminarily complete the mission and meet the mission requirements.

4.2 Future Work and Recommendations

- The use of a real ramjet analysis code that includes off-design characteristics is recommended to capture real effects on specific thrust and fuel consumption. Furthermore, the inclusion of a detailed chemistry-based combustion model is recommended to accurately model the combustor's thermodynamic characteristics and their impact in the cycle performance.
- A variable geometry design for the inlet ramps, inlet throat and nozzle throat must be developed to enhance performance at off-design conditions.
- It is recommended to validate the developed engine geometries with high-order CFD to account for real effects such as viscous phenomena and spillage. CFD can also provide a better insight into the engine's drag characteristics and the pressure distributions on the nozzle surface which has a great impact on the aerodynamics of the vehicle due to its large area. This would assess the aerodynamic performance as well as the stability

characteristics of the vehicle.

- Regarding Parametric Sizing, the hypersonic convergence tool can be improved to include additional design constraints (estimated cost, other performance characteristics, etc.) and body types (such as Waverider families) to increase the scope of hypersonic vehicle sizing.
- Improvements in the PS fuel fraction estimation can be made to provide accurate weight ratios throughout the different trajectory phases.
- The use of dedicated CFD software for hypersonic flow that accounts for viscous, compressibility and aerothermal heating effects must be used to capture the full behaviour of the surveyed designs obtained from PS.
- A complete aerothermal analysis is required as the effects of heating must be carefully defined to select the design of the TPS, materials and the cooling systems included in the design. The aerothermal analysis should also be used to constraint the trajectory to avoid heat flux limits that may affect the integrity of the structure.
- Structural loads analysis must be conducted to determine the actual structural limits of the vehicle throughout the trajectory as the 1000 psf constraint was used just as an estimate found in the reviewed literature.
- The issue raised by Hamm et al [4] regarding the integration of the APU and the energy source for the secondary power systems when the gas generators are inoperative must be assessed from the onset of CL.
- Even though the preliminary centre of gravity locations were defined for the vehicle, it is critical to pair these results with a dedicated stability and control analysis to determine the control power and trim characteristics of the vehicle.
- A landing gear design assessment must be conducted to determine If the component volumes and their locations presented in this study can provide enough clearance and stability during take-off.

III.ACKNOWLEDGEMENTS

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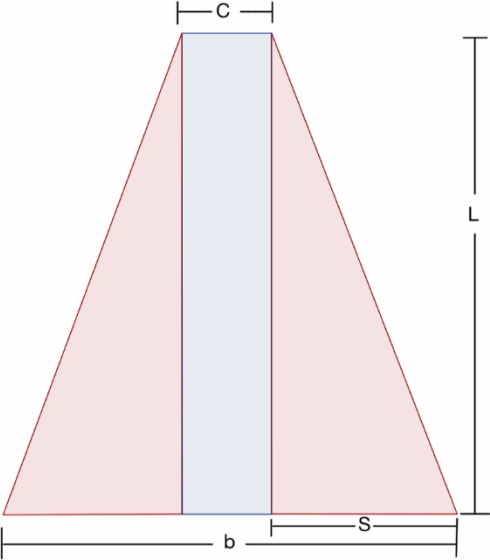
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V.APPENDIX

APPENDIX 1 – Geometry Module				
Discipline	CD Phase	Type	Author	Reference
Geometry	PS	Geometrical	Czysz, P.	[15]
Inputs: S_{pl} , Λ_{LE} , $\frac{c}{s}$, τ				
<p>Description</p> <p>Propulsion Integrated Configurations:</p> <p>Blended Body $K_W = -62.217 \cdot \tau^3 + 29.904 \cdot \tau^2 - 1.581 \cdot \tau + 2.469$</p> <p>Wing-Body $K_W = -93.831 \cdot \tau^3 + 59.920 \cdot \tau^2 - 5.648 \cdot \tau + 2.821$</p> <p>Nonweiler Waverider $K_W = -533.451 \cdot \tau^3 + 220.302 \cdot \tau^2 - 22.167 \cdot \tau + 3.425$</p> <ul style="list-style-type: none"> • $L = \sqrt{\frac{S_{pl} \tan(\Lambda_{LE})}{1 + \frac{c}{s}}}$ • $S = \frac{L}{\tan(\Lambda_{LE})}$ • $c = \frac{c}{s} * S$ • $b = 2S + c$ 				
Outputs: L , b , c , s , k_w				

APPENDIX 2: Howe Thrust Method

Discipline	CD Phase	Type	Author	Reference
Propulsion	PS	Semi-empirical	Howe, D.	[20]

Inputs: R (Engine Bypass Ratio), σ (Density Ratio)

Description

Table 3.2 Powerplant thrust parameters

Bypass ratio R	Mach number range	Operating condition	$K_{I\tau}$	$K_{2\tau}$	$K_{3\tau}$	$K_{4\tau}$	s^*
1 or lower	0-0.4	Dry	1.0	0	-0.2	0.07	0.8
		Wet	1.32	0.062	-0.13	-0.27	0.8
	0.4-0.9	Dry	0.856	0.062	0.16	-0.23	0.8
		Wet	1.17	-0.12	0.25	-0.17	0.8
	0.9-2.2	Dry	1.0	-0.145	0.5	-0.05	0.8
		Wet	1.4	0.03	0.8	0.4	0.8
3 to 6	0-0.4	Dry	1.0	0	-0.6	-0.04	0.7
	0.4-0.9	Dry	0.88	-0.016	-0.3	0	0.7
8	0-0.4	Dry	1	0	-0.595	-0.03	0.7
	0.4-0.9	Dry	0.89	-0.014	-0.3	+0.005	0.7

* Strictly s values apply up to 11 km altitude above which the factor is unity based on the 11 km altitude condition as a datum.

$$F_{\tau} = \left(\frac{T_w}{T_D} \right) / (1.32 + 0.062R) \quad (3.8)$$

For $0 < M_N \leq 0.9$:

$$\tau = F_{\tau} [K_{I\tau} + K_{2\tau}R + (K_{3\tau} + K_{4\tau}R)M_N] \sigma^s \quad (3.7a)$$

For $M_N > 0.9$:

$$\tau = F_{\tau} [K_{I\tau} + K_{2\tau}R + (K_{3\tau} + K_{4\tau}R)(M_N - 0.9)] \sigma^s \quad (3.7b)$$

$$\alpha = \tau$$

Outputs: α (Thrust Lapse Rate)

APPENDIX 3: AERODYNAMICS PS				
Discipline	CD Phase	Type	Author	Reference
Aerodynamics	PS	Empirical	Czysz, P. (Found in Coleman)	[18]

Inputs: t (US convention for τ)

- Low-Speed

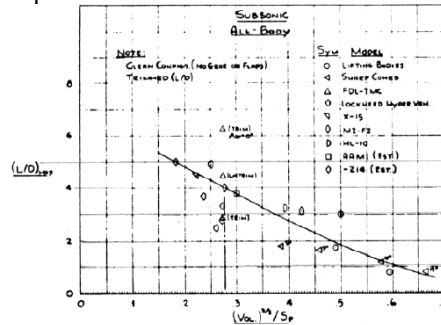


Figure 161. Subsonic Blended Body L/D Correlations

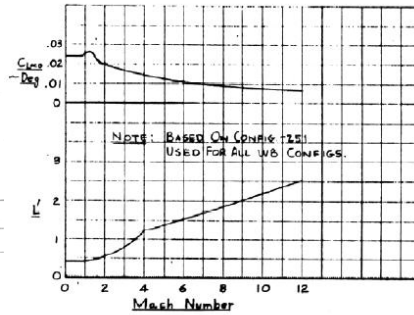


Figure 147. Wing-Body Configurations

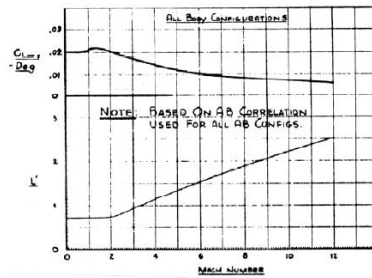
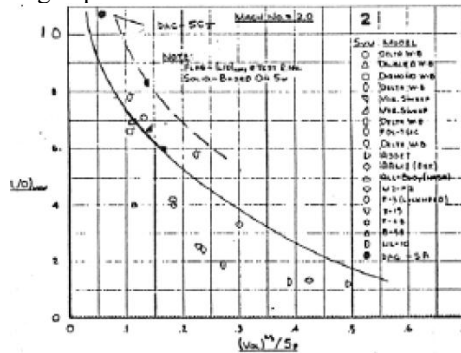
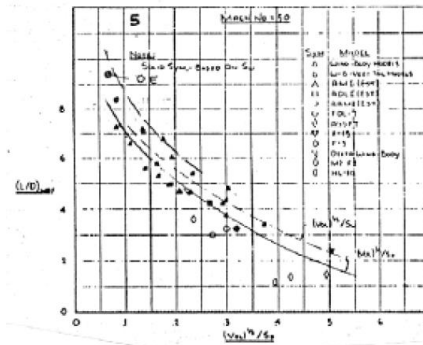


Figure 148. Blended-Body Configurations

- High-speed



- $$C_{D0} = \frac{1}{4 \cdot L' \cdot LD_{max}^2}$$

Outputs: C_{D0} , LD_{max}

APPENDIX 4: TRANSONIC DRAG				
Discipline	CD Phase	Type	Author	Reference
Aerodynamics	PS	Empirical	Czysz, P. (Found in Coleman)	[18]
Inputs: S_{front}/L^2 (Frontal area to vehicle length ratio)				
<p>Description</p> <p>If $S_{front}/L^2 < 0.015$ then</p> $(C_{Dwave})_{max} = \frac{S_{front}}{S_{pln}} \left[1.3862 \left(\frac{S_{front}}{L^2} \right) + 0.067 \right]$ <p>If $S_{front}/L^2 > 0.015$ then</p> $(C_{Dwave})_{max} = \frac{S_{front}}{S_{pln}} \left[0.9536 \left(\frac{S_{front}}{L^2} \right)^3 - 1.916 \left(\frac{S_{front}}{L^2} \right)^2 + 1.3651 \left(\frac{S_{front}}{L^2} \right) + 0.1119 \right]$				
Outputs: $C_{DW max}$				

APPENDIX 5: FUEL FRACTION MODULE				
Discipline	CD Phase	Type	Author	References
Aerodynamics	PS	Semi-empirical	Raymer, Nicolai et al, Roskam, Anderson	[21–24]
Inputs: M, LDmax, R, C, Altitude (cruise), g				
Description <ul style="list-style-type: none"> • $MF_{TO} = 0.97$ (Raymer) • $MF_{Accelerated\ Climb} = 0.7$ (Nicolai et al) • $MF_{Descend} = 0.985$ (Roskam) • $MF_{Landing} = 0.992$ (Raymer) • $MF_{cruise} = e^{\left(\frac{-R}{\left(\frac{V LD_{max}}{c g}\right)}\right)}$ (Anderson) • $ff = 1 - (MF_{TO} * MF_{Accelerated\ Climb} * MF_{Descend} * MF_{Landing} * MF_{cruise})$ 				
Outputs: ff				

APPENDIX 6: Real Ramjet Cycle Analysis				
Discipline	CD Phase	Type	Author	Reference
Propulsion	PS, CE	Analytical	Mattingly, J	[27]
Inputs: See below				
<p>Description</p> <p>INPUTS:</p> $M_0, T_0(\text{K}, ^\circ\text{R}), \gamma_c, c_{pc} \left(\frac{\text{kJ}}{\text{kg} \cdot \text{K}}, \frac{\text{Btu}}{\text{lbm} \cdot ^\circ\text{R}} \right), \gamma_t, c_{pt} \left(\frac{\text{kJ}}{\text{kg} \cdot \text{K}}, \frac{\text{Btu}}{\text{lbm} \cdot ^\circ\text{R}} \right)$ $h_{PR} \left(\frac{\text{kJ}}{\text{kg}}, \frac{\text{Btu}}{\text{lbm}} \right), \pi_{d\max}, \pi_b, \pi_n, e_c, e_t, \eta_b, \eta_m, P_0/P_9, T_{i4}(\text{K}, ^\circ\text{R}), \pi_c$ $R_c = \frac{\gamma_c - 1}{\gamma_c} c_{pc} \quad (7.20a)$ $R_t = \frac{\gamma_t - 1}{\gamma_t} c_{pt} \quad (7.20b)$ $a_0 = \sqrt{\gamma_c R_c g_c T_0} \quad (7.20c)$ $V_0 = a_0 M_0 \quad (7.20d)$ $\tau_r = 1 + \frac{\gamma_c - 1}{2} M_0^2 \quad (7.20e)$ $\pi_r = \tau_r^{\gamma_t/(\gamma_t - 1)} \quad (7.20f)$ $\eta_r = 1 \quad \text{for } M_0 \leq 1 \quad (7.20g)$ $\eta_r = 1 - 0.075(M_0 - 1)^{1.35} \quad \text{for } M_0 > 1 \quad (7.20h)$ $\pi_d = \pi_{d\max} \eta_r \quad (7.20i)$ $\tau_\lambda = \frac{c_{pt} T_{i4}}{c_{pc} T_0} \quad (7.20j)$ $\tau_c = \pi_c^{(\gamma_c - 1)/(\gamma_c e_c)} \quad (7.20k)$ $\eta_c = \frac{\pi_c^{(\gamma_c - 1)/\gamma_c} - 1}{\tau_c - 1} \quad (7.20l)$ $f = \frac{\tau_\lambda - \tau_r \tau_c}{h_{PR} \eta_b / (c_{pc} T_0) - \tau_\lambda} \quad (7.20m)$ $\tau_t = 1 - \frac{1}{\eta_m (1 + f)} \frac{\tau_r}{\tau_\lambda} (\tau_c - 1) \quad (7.20n)$ $\pi_t = \tau_t^{\gamma_t/(\gamma_t - 1) e_t} \quad (7.20o)$ $\eta_t = \frac{1 - \tau_t}{1 - \tau_t^{1/e_t}} \quad (7.20p)$ $\frac{P_{t9}}{P_9} = \frac{P_0}{P_9} \pi_r \pi_d \pi_c \pi_b \pi_t \pi_n \quad (7.20q)$ $M_9 = \sqrt{\frac{2}{\gamma_t - 1} \left[\left(\frac{P_{t9}}{P_9} \right)^{(\gamma_t - 1)/\gamma_t} - 1 \right]} \quad (7.20r)$ $\frac{T_9}{T_0} = \frac{\tau_\lambda \tau_t}{(P_{t9}/P_9)^{(\gamma_t - 1)/\gamma_t} c_{pt}} \quad (7.20s)$ $\frac{V_9}{a_0} = M_9 \sqrt{\frac{\gamma_t R_t T_9}{\gamma_c R_c T_0}} \quad (7.20t)$ $\frac{F}{\dot{m}_0} = \frac{a_0}{g_c} \left[(1 + f) \frac{V_9}{a_0} - M_0 + (1 + f) \frac{R_t T_9 / T_0 (1 - P_0/P_9)}{R_c V_9 / a_0 \gamma_c} \right] \quad (7.20u)$ $S = \frac{f}{F/\dot{m}_0} \quad (7.20v)$ $\eta_T = \frac{a_0^2 [(1 + f)(V_9/a_0)^2 - M_0^2]}{2 g_c f h_{PR}} \quad (7.20w)$ $\eta_P = \frac{2 g_c V_0 (F/\dot{m}_0)}{a_0^2 [(1 + f)(V_9/a_0)^2 - M_0^2]} \quad (7.20x)$ $\eta_O = \eta_P \eta_T \quad (7.20y)$				
Outputs: T, TSFC				

APPENDIX 7: ZERO-DIMENSIONAL ANALYSIS				
Discipline	CD Phase	Type	Author	Reference
Propulsion	PS, CE	Analytical	Ananthkrishnan et al	[29]
Inputs: $h_{PR}, \eta_b, c_{p \text{ compressor}}, c_{p \text{ turbine}}, \gamma_{\text{compressor}}, \gamma_{\text{turbine}}, f, A_1, M_1, M_2, T_1, \rho_1, R_1, R_2$				
<ul style="list-style-type: none"> h_{PR} – Fuel Heating Value η_b – Combustor Efficiency $c_{p \text{ compressor}}$ - Specific heat at constant pressure (Combustor Entrance) $c_{p \text{ turbine}}$ - Specific heat at constant pressure (Combustor Exit) $\gamma_{\text{compressor}}$ – Ratio of specific heats (Combustor Entrance) γ_{turbine} – Ratio of specific heats (Combustor Exit) f – Engine Fuel-to-air mass ratio A_1 – Combustor Entry Area M_1 - Combustor Entry Mach M_2 – Combustor Exit Mach T_1 – Combustor Entry Static Temperature ρ_1 – Combustor Entry Density R_1 – Combustor Entry Gas Constant R_2 – Combustor Exit Gas Constant <i>Altitude</i> $r = \frac{\rho_2}{\rho_1}$ $u = \frac{V_2}{V_1}$ $AR = \frac{A_2}{A_1}$ $V_1 = M_1 * \sqrt{\gamma_{\text{compressor}} R_1 T_1}$ $\dot{m}_f = \rho_1 * V_1 * A_1$ $\dot{Q}_c = \eta_b * h_{PR} * \dot{m}_f$ – Heat Released During Combustion $\bar{Q}_c = \frac{\dot{Q}_c}{\rho_1 * V_1^3 * A_1}$ $\bar{A}_W$ – Wall area to Entry area ratio $\dot{Q}_W$ – Heat Transfer Through Walls $\bar{Q}_W = \frac{\dot{Q}_W}{\rho_1 * V_1^3 * A_1}$ $\bar{R} = \frac{R_2}{R_1}$ $\bar{\gamma} = \frac{\gamma_{\text{turbine}}}{\gamma_{\text{compressor}}}$ <div style="display: flex; align-items: center; margin-top: 20px;"> <div style="flex: 1;"> <ul style="list-style-type: none"> $r \cdot u \cdot AR = 1 + f$ $r \cdot u^2 \cdot AR \left(\frac{1}{\gamma_2 M_2^2} + 1 \right) = \left(AR \cdot \frac{1}{\gamma_1 M_1^2} + 1 \right) - \left(\frac{1}{2} \right) C_f \bar{A}_W$ $\frac{1}{\gamma_2 - 1} \left(u^2 \frac{1}{M_2^2} - \bar{R} \bar{\gamma} * \frac{1}{M_1^2} \right) + \left(\frac{1}{2} \right) (u^2 - 1) = \frac{\bar{Q}_c + \bar{Q}_W}{1 + f}$ </div> <div style="flex: 0.5; font-size: 4em; margin: 0 10px;">}</div> <div style="flex: 1; text-align: center;"> <p>Conservation Equations Solved by</p> </div> </div>				
Outputs: u, r, AR				

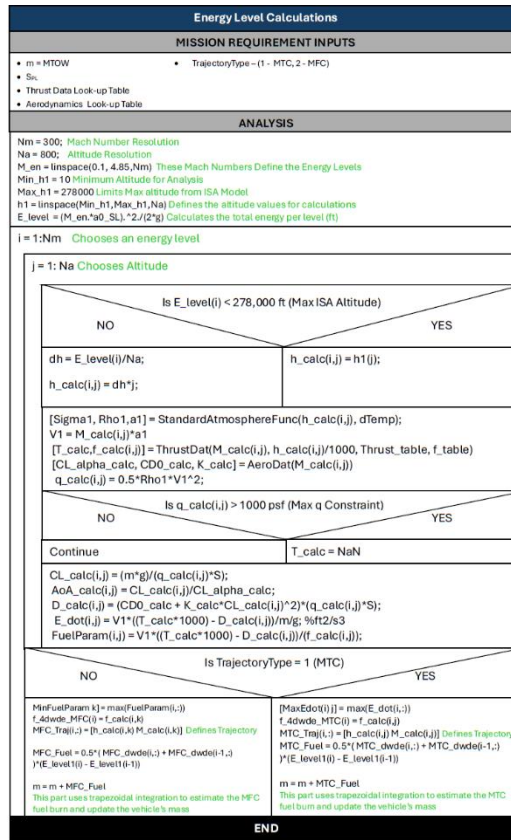
APPENDIX 8: ENERGY STATE APPROXIMATION

Discipline	CD Phase	Type	Author	Reference
Performance and Trajectory	CE	Analytical	Bryson et al	[37]

Inputs: T, f (fuel flow rate), C_L , C_D , MTOW, S_{PL}

The analysis process consists of defining constant energy levels with respect to Mach number and zero altitude and then increasing the altitude in steps while decreasing the Mach number to generate the constant energy lines for each level. Since each level is discretized, the excess power and excess power per mass of fuel (fuel parameter) can be evaluated at each location on the curve for each energy level. Once they are calculated, the code finds the locations that maximize each parameter at each energy level while choosing points whose dynamic pressure is lower than the chosen 1000 psf constraint. The trajectory is then defined by connecting those points for each trajectory type. Additionally, a trapezoidal integration is carried along the obtained trajectories to calculate the fuel burn by integrating the fuel mass from one energy level to the next while constantly updating the vehicle's total mass.

- $E_{Level} = h + \frac{(M*a)^2}{2g}$ – Energy of the vehicle at a given speed and altitude
- $\dot{E} = \frac{V(T-D)}{mg}$ – Parameter to be maximized at each level for MTC trajectory
- $Fuel\ Parameter = \frac{V(T-D)}{f}$ – Parameter to be maximized at each level for MTC trajectory



Outputs: Trajectory, Fuel Burn

APPENDIX 9: F100-PW-229 Calculated DATA

Discipline	CD Phase	Type	Author	Reference
Propulsion	CE	Data	NASA	[26]

Inputs: M, Altitude (kft)

Thrust for 3 F100-PW-229 Turbofans (kip) as a function of Mach and Altitude (kft)

3 TURBOFANS - KIPS												
h/Mach	0	0.2	0.4	0.6	0.8	0.9	1.2	1.5	2	3	4	5
0	86.922	84.705	86.94	93.603	105.066	112.779	143.631	186.93	288.909	285.777	0	0
5	74.37	72.579	74.613	80.481	90.504	97.251	124.299	162.615	255.012	389.076	0	0
15	53.379	52.245	53.874	58.299	65.796	70.836	91.152	120.339	193.395	382.479	0	0
25	37.206	36.516	37.764	40.995	46.419	50.064	64.818	86.25	141.405	290.925	0	0
35	25.065	24.666	25.584	27.852	31.638	34.179	44.502	59.631	99.414	210.834	0	0
45	15.519	15.279	15.849	17.262	19.614	21.195	27.612	37.029	61.839	131.523	0	0
55	9.597	9.447	9.801	10.674	12.129	13.104	17.073	22.896	38.238	81.342	0	0
65	7.545	7.428	7.707	8.394	9.537	10.305	13.425	18.006	30.069	0	0	0
75	0	0	0	0	0	0	0	0	0	0	0	0
85	0	0	0	0	0	0	0	0	0	0	0	0
95	0	0	0	0	0	0	0	0	0	0	0	0
105	0	0	0	0	0	0	0	0	0	0	0	0

Fuel Flow Rate (slugs/s) for 3 F100-PW-229 Turbofans as a function of Mach and Altitude (kft)

3 TURBOFANS - slugs/s												
h/Mach	0	0.2	0.4	0.6	0.8	0.9	1.2	1.5	2	3	4	5
0	1.01311	1.03627	1.10789	1.23483	1.42596	1.55011	2.04857	2.77587	4.68435	5.91655	0	0
5	0.86232	0.88228	0.94372	1.0513	1.21504	1.32157	1.74816	2.37409	4.02686	7.58156	0	0
15	0.61294	0.62698	0.67118	0.74845	0.86572	0.94182	1.24971	1.70286	2.91529	6.78267	0	0
25	0.42337	0.43317	0.46395	0.51781	0.59954	0.65267	0.86796	1.18623	2.04978	4.78987	0	0
35	0.28305	0.28962	0.31012	0.34627	0.40126	0.43732	0.58285	0.79902	1.39131	3.271	0	0
45	0.17498	0.17914	0.19184	0.21431	0.24825	0.27064	0.36069	0.49489	0.86224	2.02917	0	0
55	0.10821	0.11076	0.11863	0.13252	0.15352	0.16733	0.22302	0.306	0.53316	1.25497	0	0
65	0.08507	0.08709	0.09329	0.10421	0.12071	0.13159	0.17537	0.24065	0.41926	0	0	0
75	0	0	0	0	0	0	0	0	0	0	0	0
85	0	0	0	0	0	0	0	0	0	0	0	0
95	0	0	0	0	0	0	0	0	0	0	0	0
105	0	0	0	0	0	0	0	0	0	0	0	0

Outputs: T (kip), f (fuel flow rate – slugs/s)

APPENDIX 10: RAMJET DATA

Discipline	CD Phase	Type	Author	Reference
Propulsion	CE	Data	Mattingly, J	[27]

Inputs: M, Altitude (kft)

Ramjet Thrust (kip) as a function of Mach and Altitude (kft)

RAMJET THRUST- KIPS													
h/Mach	0	0.2	0.4	0.6	0.8	0.9	1.2	1.5	2	3	4	5	6
0	0	0	0	0	0	0	0	0	0	0	0	0	0
5	0	0	0	0	0	0	0	0	0	0	0	0	0
15	0	0	0	0	0	0	0	0	0	0	0	0	0
25	0	0	0	0	0	0	0	0	0	0	0	0	0
35	0	0	0	0	0	0	0	0	0	0	0	0	0
45	0	0	0	0	0	0	0	0	0	0	0	0	0
55	0	0	0	0	0	0	0	29.1751	43.7828	136.254	157.125	144.113	95.1954
65	0	0	0	0	0	0	0	18.0417	27.075	84.2584	97.1651	89.1185	58.8682
75	0	0	0	0	0	0	0	11.0732	16.6049	51.5627	59.2306	53.8982	34.7914
85	0	0	0	0	0	0	0	6.83698	10.2443	31.737	36.3042	32.7522	20.5984
95	0	0	0	0	0	0	0	4.24899	6.36142	19.6615	22.3958	20.0266	12.2513
105	0	0	0	0	0	0	0	2.6574	3.97535	12.2577	13.9023	12.319	7.31666

Ramjet Fuel Flow Rate (slugs/s) as a function of Mach and Altitude (kft)

RAMJET FUEL FLOW RATE - slugs/s													
h/Mach	0	0.2	0.4	0.6	0.8	0.9	1.2	1.5	2	3	4	5	6
0	0	0	0	0	0	0	0	0	0	0	0	0	0
5	0	0	0	0	0	0	0	0	0	0	0	0	0
15	0	0	0	0	0	0	0	0	0	0	0	0	0
25	0	0	0	0	0	0	0	0	0	0	0	0	0
35	0	0	0	0	0	0	0	0	0	0	0	0	0
45	0	0	0	0	0	0	0	0	0	0	0	0	0
55	0	0	0	0	0	0	0	0.72903	0.94133	2.56084	2.92322	2.86453	2.27953
65	0	0	0	0	0	0	0	0.45083	0.58211	1.58361	1.8077	1.77141	1.40964
75	0	0	0	0	0	0	0	0.2773	0.35789	0.97206	1.10643	1.07825	0.84687
85	0	0	0	0	0	0	0	0.17162	0.22137	0.60025	0.68111	0.65979	0.51073
95	0	0	0	0	0	0	0	0.1069	0.13782	0.37307	0.422	0.40629	0.30978
105	0	0	0	0	0	0	0	0.06701	0.08635	0.23333	0.26311	0.25173	0.18891

Outputs: T (kip), f (fuel flow rate – slugs/s)

APPENDIX 11: TBCC DATA

Discipline	CD Phase	Type	Author	Reference
Propulsion	CE	Data	NASA. Mattingly, J	[26,27]

Inputs: M, Altitude (kft)

TBCC Thrust (kip) as a function of Mach and Altitude (kft)

TBCC THRUST- KIPS													
h/Mach	0	0.2	0.4	0.6	0.8	0.9	1.2	1.5	2	3	4	5	6
0	86.922	84.705	86.94	93.603	105.066	112.779	143.631	186.93	288.909	285.777	0	0	0
5	74.37	72.579	74.613	80.481	90.504	97.251	124.299	162.615	255.012	389.076	0	0	0
15	53.379	52.245	53.874	58.299	65.796	70.836	91.152	120.339	193.395	382.479	0	0	0
25	37.206	36.516	37.764	40.995	46.419	50.064	64.818	86.25	141.405	290.925	0	0	0
35	25.065	24.666	25.584	27.852	31.638	34.179	44.502	59.631	99.414	210.834	0	0	0
45	15.519	15.279	15.849	17.262	19.614	21.195	27.612	37.029	61.839	131.523	0	0	0
55	9.597	9.447	9.801	10.674	12.129	13.104	17.073	52.07109	82.02081	217.5957	157.1251	144.1129	95.19542
65	7.545	7.428	7.707	8.394	9.537	10.305	13.425	36.04768	57.144	84.25838	97.16508	89.11846	58.86821
75	0	0	0	0	0	0	0	11.07316	16.60494	51.56269	59.2306	53.89819	34.79137
85	0	0	0	0	0	0	0	6.83698	10.2443	31.737	36.30423	32.75216	20.59843
95	0	0	0	0	0	0	0	4.248986	6.36142	19.66153	22.39578	20.02656	12.25132
105	0	0	0	0	0	0	0	2.657402	3.975349	12.25771	13.90233	12.31897	7.316655

TBCC Fuel Flow Rate (slugs/s) as a function of Mach and Altitude (kft)

TBCC FUEL FLOW RATE - slugs/s													
h (kft) /Mach	0	0.2	0.4	0.6	0.8	0.9	1.2	1.5	2	3	4	5	6
0	1.013108	1.036266	1.107895	1.234825	1.425959	1.550114	2.048569	2.775875	4.684347	5.916555	0	0	0
5	0.862316	0.882279	0.943723	1.051295	1.21504	1.321573	1.74816	2.374087	4.026862	7.581558	0	0	0
15	0.612935	0.626978	0.671178	0.748453	0.865719	0.941818	1.24971	1.702855	2.915291	6.782667	0	0	0
25	0.423371	0.433174	0.463954	0.517807	0.599542	0.652672	0.86796	1.186226	2.049783	4.789875	0	0	0
35	0.283053	0.289621	0.310119	0.346267	0.401258	0.437321	0.582851	0.799017	1.391307	3.271005	0	0	0
45	0.174984	0.179138	0.191841	0.21431	0.248252	0.270641	0.360686	0.494886	0.862239	2.029171	0	0	0
55	0.108211	0.110761	0.118634	0.132519	0.153515	0.167326	0.223019	1.035028	1.474497	3.815805	2.923216	2.864533	2.279526
65	0.085074	0.087089	0.093288	0.104213	0.120709	0.131586	0.175366	0.691473	1.001375	1.583606	1.807697	1.771408	1.409644
75	0	0	0	0	0	0	0	0.277305	0.357887	0.972062	1.106434	1.078249	0.846869
85	0	0	0	0	0	0	0	0.171617	0.221372	0.600249	0.681113	0.659789	0.510731
95	0	0	0	0	0	0	0	0.106903	0.137823	0.373066	0.422002	0.406293	0.309777
105	0	0	0	0	0	0	0	0.067013	0.086351	0.233335	0.263106	0.251728	0.188914

Outputs: T (kip), f (fuel flow rate – slugs/s)

APPENDIX 12: Hypersonic Convergence Inputs and Weight/Volume Equations				
Discipline	CD Phase	Type	Author	References
Synthesis	PS	Numerical	Czysz, P. Chudoba et al	([15] Chapter 16) ,[25]
Inputs: See Below				
<ul style="list-style-type: none"> Inputs <ul style="list-style-type: none"> I_{str} – Structural Index $\left(\frac{kg}{m^2}\right)$ $18\left(\frac{kg}{m^2}\right) \leq I_{str} \leq 23\left(\frac{kg}{m^2}\right)$ f_{sys} – Systems Fraction $\left(\frac{ton}{ton}\right)$ $0.16\left(\frac{ton}{ton}\right) \leq f_{sys} \leq 0.24\left(\frac{ton}{ton}\right)$ C_{un} – Unmanned Systems Weight(ton) $1.9(ton) \leq C_{un} \leq 2.1(ton)$ f_{mnd} – Manned Systems Fraction $\left(\frac{ton}{person}\right)$ $1.45\left(\frac{ton}{person}\right) \leq f_{mnd} \leq 1.05\left(\frac{ton}{person}\right)$ W_{cprv} – Crew Provisions Weight (kg) W_{pay} – Payload Weight (kg) μ_a – Inert Mass Fraction N_{crew} – Crew Number W_{crew} – Crew Weight (kg) $C_{sys} = C_{un} + f_{mnd}N_{crew}$ E_{TW} – Engine Thrust – to – Weight Ratio k_{ve} – Engine Volume Coefficient $\left(\frac{m^3}{kg}\right)$ $\rho_{\frac{ppt}{fuel}}$ – Fuel Density $\left(\frac{kg}{m^3}\right)$ k_{vv} – Void Volume Coefficient $\left(\frac{m^3}{m^3}\right)$ k_{vs} – Systems Volume Coefficient $\left(\frac{m^3}{m^3}\right)$ ρ_{pay} – Payload Density $\left(\frac{kg}{m^3}\right)$ $48\left(\frac{kg}{m^3}\right) \leq \rho_{pay} \leq 130\left(\frac{kg}{m^3}\right)$ k_{crw} – Crew Volume $\left(\frac{m^3}{person}\right)$ $0.9\left(\frac{m^3}{person}\right) \leq k_{crw} \leq 2\left(\frac{m^3}{person}\right)$ Weights <ul style="list-style-type: none"> $W_{str} = I_{str}k_w S_{pl} + W_{cprv}$ $W_{sys} = C_{sys} + f_{sys}W_{dry}$ $W_{eng} = \frac{TW_0 WR}{E_{TW}} (W_{dry} + W_{pay} + W_{crw})$ Volumes <ul style="list-style-type: none"> $V_{total} = \tau S_{pl}^{1.5}$ $V_{ppt/fuel} = W_{OE} \frac{(WR-1)}{\rho_{ppt/fuel}}$ $V_{sys} = k_{vs}V_{total}$ $V_{void} = k_{vv}V_{total}$ $V_{pay} = \frac{W_{pay}}{\rho_{pay}}$ $V_{crw} = (V_{cprv} + k_{crw})N_{crew}$ 				
Outputs: Weights, Volumes				

APPENDIX 13: WEIGHT DISTRIBUTION

Discipline	CD Phase	Type	Author	Reference
Weights and Balance	CL, CE	Numerical	SolidWorks 2025, Cranfield University	[40,41]

Inputs: MTOW, Component Masses and Volumes

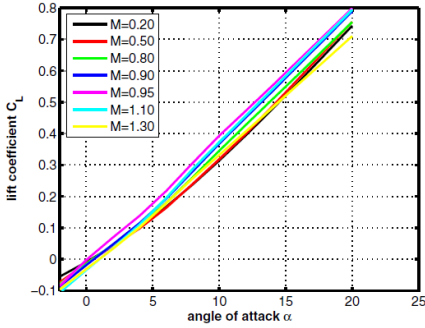
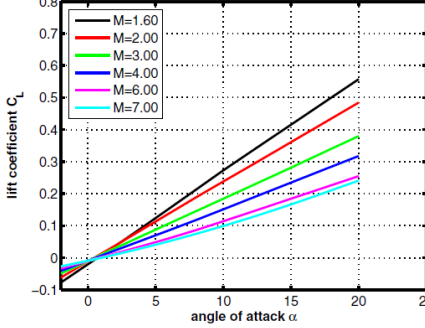
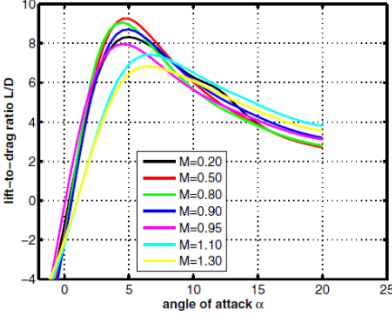
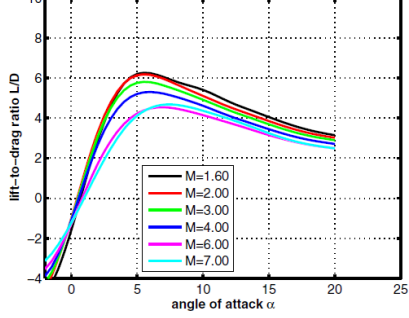
The weights and Volumes from PS were used to develop the reduced-order CAD models of each component and determine their respective CG locations within the aircraft (Origin at Nose). With the MTOW, the methods by Cranfield University were used to calculate the systems weights per component and their distribution is as follows:

- $W_{fuel\ system} = (1.5 * 0.4) * MTOW^{0.8}$
- $W_{flight\ controls} = 0.11 * MTOW^{0.8} + 0.005 * MTOW$
- $W_{electrics} = 0.09 * MTOW^{0.84}$
- $W_{landing\ gear} = 0.077 * MTOW^{0.93}$
- $W_{fuel\ system} = (1.5 * 0.4) * MTOW^{0.8}$
- $W_{miscellaneous} = W_{sys\ from\ PS} - W_{sys\ from\ DaeTC}$

Forward Systems		Weight (kg)
30% Flight Controls		381.0538223
50% Electrics		569.4338635
Nose Landing Gear		Weight (kg)
35% Landing Gear (Nose)		938.2407088
Flight Deck		Weight (kg)
Crew Prv		70
Instruments		170
Crew		133
Centre Systems		Weight (kg)
%20 Flight Controls		254.0358815
30% Electrics		341.6603181
65% LG (MAIN)		1742.447031
50% Fuel System		242.1072021
Engine Systems		Weight (kg)
50% Fuel System		242.1072021
APU		200
Aft Systems		Weight (kg)
%50 Flight Controls		635.0897038
%20 electrics		227.7735454
Miscellaneous		Weight (kg)
Miscellaneous		2332.250722

Component	Weight (kg)	X	Y	Z
Forward Systems	950.49	1.61	0	0.09
Nose Landing Gear	938.24	5.92	0	-0.26
Flight Deck	373	7.23	0	0.54
Payload	4536	9.53	0	0.05
Forward Tank (Full)	13315.74	13.93	0	-0.03
Centre Tank (Full)	10168.87	16.82	0	0.15
Aft Tank (Full)	21284.66	20.62	0	-0.02
Main Gear and Centre Syst	2580.25	16.86	0	-0.22
Engine Systems	442.11	17.11	0	-1.87
Engine Structure	1965	16.83	0	-1.88
F100-PW-229 # 1	1735	17.29	-1.19	-1.01
F100-PW-229 # 2	1735	17.29	0	-1.01
F100-PW-229 # 3	1735	17.29	1.19	-1.01
Aft Systems	862.86	23.41	0	-0.01
Aiframe Structure + Miscel	11116.39	15.89	0	-0.06

Outputs: Systems Weights Breakdown, Vehicle CG

APPENDIX 14: SAENDER II DATA WEILAND				
Discipline	CD Phase	Type	Author	Reference
Aerodynamics	Reference	Data	Weiland, C.	[43]
Inputs: None				
<p>The following aerodynamic data tables for SANGER II were obtained from “<i>Aerodynamic Data of Space Vehicles</i>” by Claus Weiland, Chapter 7 - Section 2. They are included as an appendix to compare their behaviour with the results obtained in this study.</p>				
<div>   </div>				
<div>   </div>				
Outputs: None				