Ultra-Premium, Long-Range Business Jet

HR One

AME 261 – Flight Dynamics - 2025 Design Project

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HR One

Contents

Table 1: Nomenclature

Parameter	**Symbol**
Fuselage diameter [m]	d
Fuselage length [m]	l
Airfoil type	NACASC(2) - 0610
Wing span [m]	b
Wing chord [m] (or MAC)	c
Wing area [m^2]	S
Aspect ratio	AR
Horizontal Tail area [m^2]	S_H
Vertical Tail area [m^2]	S_V
Parasite drag coef.	C_{D_0}
Parasite drag area for fuselage [m^2]	A_{fuse}
Parasite drag area for plane [m^2]	A_{total}
Stall velocity @ cruise [m/s]	V_{stall}
Maximum Lift-to-Drag ratio	E_{max}
Maximum Rate of Climb [m/s]	RoC_{max}
Cruise Velocity [m/s]	V_∞
Max lift coef. (with flaps deployed)	$C_{L_{\text{max}}}$
Cruise lift coef.	$C_{L,\text{cr}}$
Chord Reynolds number	Re_c
Drag [N]	D_{cruise}
Minimum drag [N]	D_{min}
Velocity at D_{min} [m/s]	$V_{D_{\text{min}}}$
Minimum power required [W]	$P_{R_{\text{min}}}$
Velocity at $P_{R_{\text{min}}}$ [m/s]	$V_{P_{\text{min}}}$
Max power available [W]	P_A
Weight of the aircraft [N]	W_0
Mission total range flown [km]	X_{total}
Max range without refueling [km]	X_{max}
Mission total flight time [hr]	t_{total}
Max endurance [hr]	ξ_{max}
Payload weight [N]	W_{pay}

1 Abstract

The design proposal to create a high-end business jet required a balance between comfort, speed, and cost-efficiency. The aircraft needed to complete long-range missions while remaining lightweight and within FAA regulations. The Highroller 1 features advanced materials and cutting edge technology to maximize aircraft capability. The Highroller-1 seats 8 passengers in lay-flat reclinable seats and has a hot water shower. With a maximum range of over 9000 nautical miles and a cruise velocity of Mach 0.92, the HR One can complete the Aspen Mission, Napa Mission, and more. The HR One boasts a wingspan of 30.7 meters and a maximum take off weight of 40425 kilograms.

The HR One was designed through extensive trade studies and sizing methods. Individual trade studies were ran on MATLAB to select the NACA SC(2) - 0610 supercritical airfoil and the sweep angle of the aircraft. Then, the Raymer weight sizing method [4] was used to size the weight fractions of the aircraft. Finally, an extensive MATLAB script was utilized to trade the remaining parameters such as aspect ratio (AR = 9.68) and wingspan (b = 30.987m). Additional features such as HYTEC Hybrid Engines and folding wingtips were added to optimize the aircraft. The resulting aircraft is an industry leading aircraft that maximizes performance and efficiency with zero sacrifices in luxury in and comfort.

2 Introduction

The rapid globalization of business and the ever-increasing demand for uninterrupted, transcontinental travel have driven a corresponding evolution in the design of ultra-long-range business jets. Today's corporate and private travelers demand not only exceptional comfort and onboard amenities, but also the ability to fly nonstop over greater distances at near-transonic speeds. By eliminating intermediate stops, operators save time, reduce logistical complexity, and offer a seamless door-to-door experience. In this context, the Highroller One (HR One) project seeks to push the boundaries of range and efficiency while maintaining the highest standards of luxury and safety.

Early milestones in the long-range business-jet arena include the late 1960s introduction of the Gulfstream II and the subsequent evolution to the G650ER, which demonstrated a nonstop capability of up to 8,379 nm—an industry record at the time [1]. Bombardier's Global 7500, entering service in 2018, further extended that envelope to 7,700 nm [2], while Dassault's Falcon 8X offers a 6,450 nm range in a three-zone cabin layout [3]. Despite these advances, no current turbofan business jet combines both a cruise speed approaching Mach 0.92 and a nonstop range exceeding 9,000 nm—the gap that the HR One is designed to fill.

The design challenges begin with stringent regulatory requirements under FAA Part 25 for structural integrity, handling qualities, and system redundancies. Airport infrastructure imposes gate-width limits (e.g., the 28.96 m maximum wingspan at Aspen), runway-length constraints at high-elevation fields, and noise-abatement procedures. Environmental considerations—particularly life-cycle emissions—are also paramount, as operators and regulators move toward carbon-neutral flight. Finally, mission requirements dictate cabin comfort (seating for eight in lay-flat reclinable berths, private shower facilities), payload-range trade-offs, and integration of advanced propulsion technologies such as NASA's Hybrid Thermally Efficient Core (HyTEC) engines. [5]

To meet these demands, the HR One design is organized into six core workstreams:

1. **Weight Estimation Mission Performance:** Establish takeoff, empty, payload, and fuel weights; analyze three distinct missions; develop detailed weight breakdowns; and map the center-of-gravity envelope alongside range and fuel-volume calculations.
2. **Aerodynamics Wing Design:** Select and optimize supercritical airfoils; define wing aspect ratio, sweep, taper, and area; compute zero-lift drag (C_{D0}), Oswald efficiency, and $C_{L_{max}}$; and generate V–n and turning-radius envelopes.
3. **Propulsion Powerplant:** Identify an engine; chart power required vs. available; quantify fuel burn per flight segment; and ensure seamless integration within airframe weight and volume constraints.
4. **CAD Geometry / Internal Layout:** Craft the fuselage geometry and cabin layout; allocate space for galley, lavatories, berths, and systems; produce fully dimensioned three-view drawings.
5. **Cost Emissions Analysis:** Project development, flyaway, and operating costs; calculate life-cycle CO_2 and N_2O emissions; evaluate material-choice impacts; and justify trade-offs between cost, luxury, and performance.
6. **Trade Studies Configuration Optimization:** Conduct parametric studies (e.g., wing size vs. range, Mach vs. fuel burn); down-select optimal configurations; perform sensitivity analyses; and identify the best design compromises.

Team Member Responsibilities

- **Diego (Weight Estimation Mission Performance):** Takes ownership of all weight-and-fuel calculations, detailed mission-profile analysis, and CG-envelope modeling.
- **Gui (Aerodynamics Wing Design):** Leads airfoil selection, drag-buildup studies, and performance envelope generation.
- **Seojoon (Propulsion Powerplant):** Selects and integrates the powerplant, producing thrust and HyTEC Technology.
- **Sammy (CAD Geometry / Internal Layout):** Develops interior layouts, three-view drawings, and fuel-tank volume models.
- **Katie (Cost Emissions Analysis):** Estimates costs and life-cycle emissions while assessing material and performance trade-offs.
- **Jana (Trade Studies Configuration Optimization):** Runs major trade studies, down-selection exercises, and sensitivity analyses to refine the overall configuration.

With this framework in place, the following sections delve into the analytical details and quantitative results that validate the HR One's potential as the next-generation leader in ultra-long-range business aviation.

3 Theoretical Presentation

3.1 Aerodynamic Characteristics and Aerodynamic Performance

3.1.1 Level Unaccelerated Flight

In steady, level flight, the aircraft is in equilibrium:

$$L = W, \qquad T = D \tag{1}$$

From this equilibrium condition, lift can be expressed in terms of the dynamic pressure $q = \frac{1}{2}\rho V^2$ and wing reference area S:

$$C_L = \frac{W}{qS} \tag{2}$$

3.1.2 Drag Buildup

The total drag of the aircraft can be decomposed into three specific components of drag: parasite drag C_{D_0}, induced C_{D_i} and compressible ΔC_{D_c}. This buildup provides the following equation:

$$C_D = C_{D_0} + C_{D_i} + \Delta C_{D_c} \tag{3}$$

Parasite Drag The parasite drag on an aircraft can be found by the equation

$$D_0 = q\Big(A_{D_\text{wing}} + A_{D_\text{fuse}} + A_{D_{h\text{-tail}}} + A_{D_{v\text{-tail}}} + A_{D_\text{aux}}\Big)$$

where A_D is the equivalent parasite drag area of each component, defined by

$$A_D = C_{D_0}\, S_\text{wet},$$

with S_wet the wetted surface area of the component.

Skin-friction coefficient The parasite drag coefficient C_{D_0} is a function of the skin-friction coefficient C_f. Assuming fully turbulent flow,

$$C_f = \frac{0.074}{Re_x^{0.2}}, \qquad Re_x = \frac{\rho V_\infty x}{\mu},$$

where Re_x is the Reynolds number based on characteristic length x, V_∞ is the free-stream velocity, ρ the air density, and μ the dynamic viscosity.

Fuselage For ease of calculation, the fuselage is idealised as a prolate ellipsoid:

$$C_{D_{0,\mathrm{fuse}}} = \left[1 + 1.5\left(\frac{l_f}{d_f}\right)^{-1.5} + 7\left(\frac{l_f}{d_f}\right)^{-3}\right] C_f, \qquad S_{\mathrm{fuse}} = \frac{\pi d_f}{2}\,(d_f + l_f),$$

where l_f is fuselage length and d_f is fuselage diameter. For the Reynolds-number calculation the characteristic length is $x = l_f$.

Wing and Empennage For the wing and tail surfaces,

$$C_{D_0} = 2C_f, \qquad S_{\mathrm{wet}} = c\,b,$$

where c is the mean aerodynamic chord and b is the span of the wing. Tail sizing follows the wing: the horizontal tail span is $0.4\,b$ with chord $0.5\,c$; the vertical tail span and chord are each half those of the horizontal tail.

Auxiliary Components The equivalent parasite drag area due to auxiliary items (antennas, control surfaces, etc.) is assumed to be $A_{D_{\mathrm{aux}}} = 0.02\ \mathrm{m}^2$. Any contribution these parts make to the parasite drag coefficient itself is neglected.

Induced Drag The induced drag arises from lift generation and depends on wing geometry through the span-efficiency factor e and the aspect ratio AR:

$$C_{D_i} = \frac{C_L^2}{\pi e AR}. \tag{4}$$

Compressible Drag At the high cruise Mach numbers relevant to this aircraft design, compressibility effects significantly impact drag. The compressible drag increment is calculated as follows:

$$\Delta C_{D_c} = y \cos^3 \Lambda, \tag{5}$$

where

$$y = 3.97 \times 10^{-9} e^{12.7x} + 10^{-40} e^{81x}, \quad x = \frac{M_\infty}{M_{cc,\Lambda}}, \tag{6}$$

$$M_{cc,\Lambda} = \frac{M_{cc,\Lambda=0}}{\cos^m \Lambda}, \quad m = 0.83 - 0.583 C_L + 0.111 C_L^2, \tag{7}$$

$$M_{cc,\Lambda=0} = 0.87 - 0.175 C_L - 0.83\left(\frac{t}{c}\right). \tag{8}$$

The Mach number M itself is determined by:

$$M = \frac{V}{a} = \frac{V}{\sqrt{\gamma R T}}, \tag{9}$$

with $\gamma = 1.4$, $R = 287\,\mathrm{J/kg \cdot K}$, and temperature T based on altitude conditions.

Critical-Mach Number As air travels over the suction surface near the airfoils leading edge, the local Mach number of the flow may exceed the free stream value M_∞. The lowest free-stream Mach number at which any point over the surface first reaches supersonic speed ($M_{local} = 1$) is the critical mach number M_{cc} [4]. This concept is illustrated below:

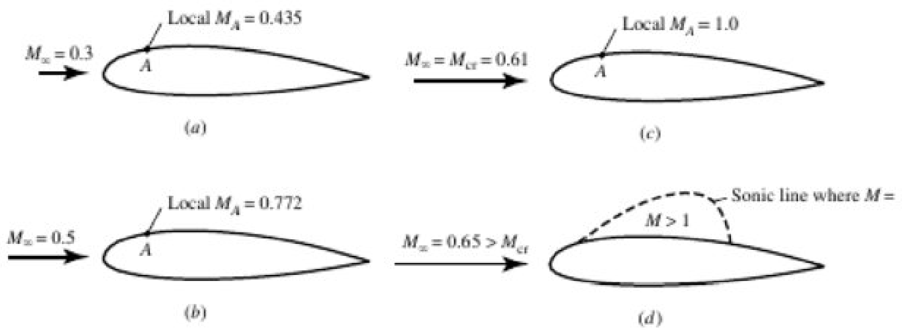


Figure 1: Critical Mach number for a notional airfoil showing development of local supersonice flow

This leads to the following relationship for the coefficient of pressure due to compressibility correction:

$$C_p = \frac{C_{p_0}}{\sqrt{1 - M_\infty^2}} \tag{10}$$

where $C_{p,0}$ is the incompressible (low-Mach) minimum pressure coefficient. The coefficient of lift faces a similar correction:

$$C_L = \frac{C_{L_0}}{\sqrt{1 - M_\infty^2}} \tag{11}$$

Supercritical Airfoils Supercritical airfoils are a specific type of airfoil designed to minimize the effects of compressibility at high Mach numbers to increase efficiency. The main visual difference between a conventional airfoil to a supercritical one is that supercritical airfoils have a flatter top (suction) surface and a rounded bottom with the trailing edge accented downward to restore lift lost. The supercritical airfoil moves the drag divergence Mach number closer to $M_\infty = 1$, which delays the onset of shock-induced wave drag and improves efficiency. It also reduces shock-induced flow separation and allows for thicker wings to hold fuels and less sweep to save weight. The following diagram demonstrates the location and magnitude of the compressible shock effects on conventional and supercritical airfoils:

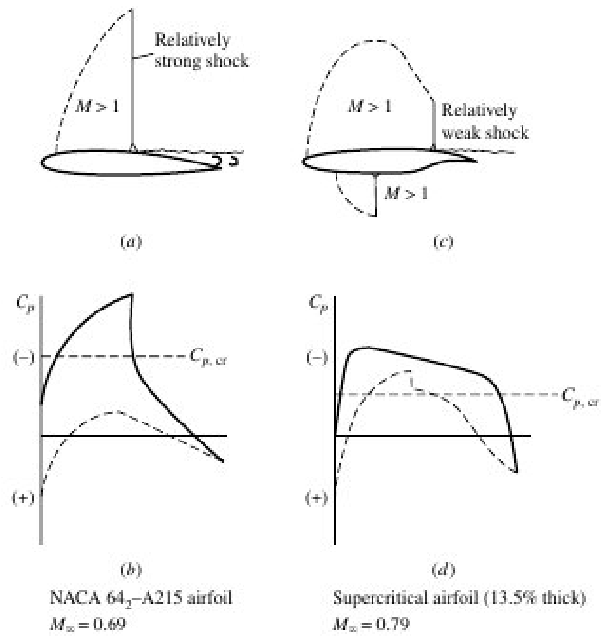


Figure 2: Comparison of traditional airfoil with supercritical airfoil showing differences in shock

The equations for calculating the compressible effects are specifically made for conventional. However, since this airplane will use supercritical, there needs a correction. According to Raymer's Aircraft Conceptual Design 6th Edition, the value of the thickness to chord ratio needs to be multiplied by 0.6 to correct for a supercritical airfoil [4].

Wing Sweep Another method of delaying the wave drag due to compressible effects is by sweeping the wings of the aircraft. The sweep reduces the speed experienced by the wing depicted as the speed of the flow is now $M_{eff} = M_\infty \cos(\Lambda)$ where Λ is the sweep of the wing. This directly impacts the critical mach number that is now reduced by a factor of $\frac{1}{\cos(\Lambda)}$.

3.2 Aircraft Performance and Flight Envelope

3.2.1 Turning Flight

Level, coordinated turn performance follows directly from the level, steady flight assumption and introduces the load factor:

$$n = \frac{L}{W} = \frac{1}{\cos\phi} \tag{12}$$

Where ϕ is the bank angle. Using the load factor, the following equation will be to calculate the turn radius:

$$r = \frac{V^2}{g\sqrt{n^2 - 1}} \tag{13}$$

The turn rate of the aircraft will be calculated using:

$$\omega = \frac{g\sqrt{n^2 - 1}}{V}. \tag{14}$$

There are three main limitations in an aircraft's turn ability: structural load n_{struct}, thrust limit T_{max} and stall considerations $C_{L_{max}}$. Structural design must satisfy $n \leq n_{\text{limit}}$ per FAR 25, while engine thrust must exceed drag at the elevated C_L and ΔC_{D_c}. The structural limit is merely a constant that can be placed into the turn radius equation below:

$$r = \frac{V^2}{g\sqrt{n_{struct}^2 - 1}}$$

The coefficient of lift limitation equation is:

$$r = \frac{2(W/S)}{\rho g \left(C_L^2 - \left(\frac{W}{qS}\right)^2\right)^{1/2}}$$

Lastly, the thrust limitation is given by firstly solving for the load factor based on thrust:

$$n = \sqrt{(T_A - C_{D_0} qS)\frac{qS}{kW^2}} \tag{15}$$

Then, using this to plug into equation (12).

3.2.2 Take-Off and Landing

Due to margin and safety, the following condition also must be met:

$$C_{L_{max},TO} = 0.8 C_{L_{max}}$$

The ground-roll is calculated using:

$$d_{\text{LO}} = \frac{1.44\, W^2}{\rho\, g\, S\, C_{L_{\max}} \big(T - D_\phi + \mu_r (W - L)\big)} \tag{16}$$

For the condition in which $T_{max} \gg D$, the following approximation can be used:

$$\frac{1.44\, W^2}{\rho\, g\, S\, C_{L_{\max}}\, T}$$

These calculations are done over the following conditions:

$$V_{TO} = 1.2 V_{stall} = 1.2\sqrt{\frac{2W}{\rho S}}$$

$$T_{TO} = 0.9 \Pi T_A$$

The take-off field length will be calculated using:

$$f_{TO} = (0.217\chi + 183) \tag{17}$$

where:

$$\chi = \frac{W^2}{\rho^2 S C_{L_{max}} (0.9 \Pi T_{A_{SL}})}$$

The landing performance will be evaluated using:

$$d_G = \frac{1.69W^2}{\rho g S\left[T_{rev} + (D + \mu_r(W - L))\right]_{0.7V_T}} \tag{18}$$

The FAR requires to be calculated using:

$$V_{LO} = 1.3V_{stall}$$

3.2.3 Rate of Climb

For a jet, the rate of climb is obtained using:

$$\text{RoC} = V\,\frac{T_A - D}{W}, \tag{19}$$

where V is true airspeed, T_A thrust available and D drag required. The aircraft must satisfy FAR25 first-segment climb (RoC ≥ 0 on one engine inoperative) at MTOW.

3.2.4 Range and Endurance

Range can be calculated using a multitude of equations, depending on desired constant factors. For constant h and C_L, Maximum range occurs at $\frac{\sqrt{C_L}}{C_D}$, so that coefficients of lift and drag are calculated as follows:

$$C_L = \sqrt{\frac{\pi e A R C_{D0}}{3}} \tag{20}$$

$$C_D = \frac{4}{3}C_{D0} \tag{21}$$

Then, following the Lecture 11 derivation, the Breguet jet range equation for level cruise at constant Mach is

$$\text{Range} \;=\; \frac{V}{\text{SFC}_j}\,\frac{L}{D}\,\ln\!\left(\frac{W_i}{W_f}\right), \tag{22}$$

with specific fuel consumption $\text{SFC}_j = c_T$, and L/D computed using the full, compressible drag model. However, FAR conditions hold V and h constant. The equation for the weight factor of this best range equation is shown in equation 15:

$$W_0^* = \frac{W_{TO}}{\sqrt{C_{D0}/k} * q * S} \tag{23}$$

The range is calculated as follows, with $E_m = (L/D)_{max}$.

$$\text{Range} = \frac{2VE_m/SFC * arctan(W_0^*\xi)}{1 + W_0^{*2}(1 - \xi)} \tag{24}$$

The endurance is related closely to range in both range equations, as

$$\epsilon = \frac{Range}{V} \tag{25}$$

In this situation, E_m is held constant as a design parameter, but can be varied to account for compressible effects.

3.3 Aircraft Weight Statement, Aircraft CG Envelope, and FWD/AFT CG Limits

3.3.1 FWD/AFT CG Limits and CG Envelope

To establish safe limits for the CG to shift, first calculate the neutral point, h_n, which is the AFT limit the CG can shift. Define the equation for h_n:

$$h_n = h_{ac} + \frac{a_t}{a_w}(1 - \frac{\partial \varepsilon}{\partial \alpha})\eta_t V_t \tag{26}$$

The aerodynamic center is calculated to be a quarter of the mean aerodynamic chord ($\bar{c}$) behind the leading edge of the wing. In dimensional form (meters), $x_{ac} = 0.25(\bar{c})$. The dimensionless form is known as h_{ac}, which is normalized by $\bar{c}$, therefore,

$$h_{ac} = \frac{0.25(\bar{c})}{\bar{c}} = 0.25. \tag{27}$$

The partial of the downwash angle, ε, can be defined as:

$$\frac{\partial \varepsilon}{\partial \alpha} = \frac{2a_w}{e\pi AR} \tag{28}$$

The tail efficiency factor, η_t, is known as 0.9, which accounts for flow interference due to the aircraft, forward of the tail.

The tail volume coefficient, V_t, gauges the longitudinal (pitch) stability and control authority of the horizontal tail, and is known from when the target CG was calculated.

The $C_{L\alpha}$ slopes of the tail and wing, a_w and a_t, are already known from aerodynamic calculations.

To find the FWD CG limit, take HR One's maximum elevator deflection to be 15 degrees, for both positive and negative deflections. The maximum forward shift occurs when the elevator is deflected to -15 degrees.

To maintain static stability, $\frac{\partial C_{M_{cg}}}{\partial C_L} < 0$. $\frac{\partial C_{M_{cg}}}{\partial C_L}$ definition is:

$$\frac{\partial C_{M_{cg}}}{\partial C_L} = h_{cg} - h_{ac} - \frac{a_t}{a_w}(1 - \frac{\partial \varepsilon}{\partial \alpha})\eta_t V_t = -SM, \tag{29}$$

But the definition changes when elevator deflection is introduced. To find the new value for this slope, define the equation for $C_{M_{CG}}$:

$$C_{M_{cg}} = C_{M_o} + \frac{\partial C_{M_{cg}}}{\partial C_L} \cdot C_L + \frac{\partial C_{M_{cg}}}{\partial \delta_e} \cdot \delta_e \tag{30}$$

Where $\frac{\partial C_{M_{cg}}}{\partial \delta_e}$ is defined as:

$$\frac{\partial C_{M_{cg}}}{\partial \delta_e} = -a_t \eta_t V_t \tau \tag{31}$$

Without true elevator sizing data, assume the elevator efficiency parameter, $\tau = 0.5$.

At the forward CG limit, $C_L = C_{L_{max}}$, and the aircraft is in longitudinal equilibrium, therefor, $C_{M_{cg}} = 0$. Once the new slope is found, the FWD limit is $h_{cg} - h_{ac}$, which is defined as:

$$h_{cg} - h_{ac} = \frac{\partial C_{M_{cg}}}{\partial C_L} + \frac{a_t}{a_w}(1 - \frac{\partial \varepsilon}{\partial \alpha})\eta_t V_t. \tag{32}$$

In this theoretical presentation, the CG envelope will simply consist of the FWD and AFT limits that the CG can move, which are presented as two vertical lines on a graph.

3.3.2 Initial Sizing and Weight Estimate

Initial sizing was conducted in order to estimate maximum takeoff weight and verify the HR One's ability to satisfy mission requirements. Preliminary weight estimates were done using methods described in Raymer's *Aircraft Design - A Conceptual Approach*. The method begins by calculating a fuel fraction (W_f/W_0) based on specific mission segment fuel burn, along with a statistical empty weight (W_e/W_0), so that the gross take-off weight can be found iteratively from equation (33).

$$W_0 = \frac{W_{crew} + W_{payload}}{1 - \frac{W_f}{W_0} - \frac{W_e}{W_0}} \tag{33}$$

Fuel fraction is given below by equation (34), where W_x/W_0 is the fuel fraction of the entire mission, and $x = 5$ for distinct mission segments: warm-up and take-off, climb, cruise, loiter, and landing.

$$\frac{W_f}{W_0} = 1.06\left(1 - \frac{W_x}{W_0}\right) \tag{34}$$

Mission segments 1, 2 and 5 are taken directly from Raymer. Meanwhile segment 3, cruise, comes from the Breguet Range Equation, rearranged.

$$\frac{W_3}{W_2} = \exp\left(-\frac{R \cdot SFC_j}{V \cdot L/D_{cruise}}\right) \tag{35}$$

For the HR One, flying at Mach 0.92, $R = 8000nm = 14,816km$. Raymer provides historical data to estimate L/D for similar aircraft, which yielded $L/D = 20$. An SFC_j of 0.6099 was also used. Exact calculation of SFC_j will be discussed in a later section.

Lastly, segment 4, loiter, is governed by FAA regulation that all passenger aircraft be equipped with 30 minutes of additional cruise fuel [4]. Derived from the range equation in Chapter 17 of Raymer [4], loiter weight fraction is given by the following:

$$\frac{W_4}{W_3} = \exp\left(-\frac{E \cdot SFC_j}{L/D}\right) \tag{36}$$

where

E = loiter time [s]

Table 2: Mission Segment Fuel Fractions [1]

Fuel Fraction [-]	Estimated Value	Mission Segment
W_1/W_0	0.97*	Warm-up and Take-off
W_2/W_1	0.985*	Climb
W_3/W_2	0.553	Cruise
W_4/W_3	0.983	Loiter
W_5/W_4	0.995*	Landing

With segments 1,2 and 5 dervied from Raymer, the following mission segment fuel fractions were calculated.

The statistical empty weight fraction is also derived from historical data for similar aircraft. A 5% composites weight reduction is applied [cite raymer].

$$\frac{W_e}{W_0} = (0.85)(1.02)W_0^{-0.06} \tag{37}$$

The iterative process begins with an initial guess for W_0, which yields an initial empty weight fraction. Gross take-off weight is then found iteratively from equation (33). A value between the guess and calculated MTOW is used to recalculate gross take-off weight until both converge. An initial guess of $45,000$ kg, yielded an MTOW of $38,745$ kg.

3.4 Stability and Control

Static Margin Theory

At this point, the theory behind calculating the static margin is complete. Static margin values tend to be fairly standard across aircraft, and selecting a static margin before knowing these values can provide a strategic way to formulate other values. Static margin quantifies an aircraft's longitudinal stability by measuring the distance between its center of gravity (CG) and its neutral point (NP), normalized by the mean aerodynamic chord. A positive SM means the CG lies ahead of the NP, creating a restoring pitching moment whenever the aircraft's angle of attack deviates from trim. The greater the SM, the stronger this restoring moment—and thus the more inherently stable the aircraft—but excessive SM increases tail size, trim drag, and degrades pitch control authority. Conversely, too small an SM yields twitchy, lightly damped behavior that can challenge pilot or autopilot control. By selecting an SM in a mid-range "sweet spot", the aircraft balances stability and maneuverability, ensuring predictable handling qualities, and provides resilience against CG shifts due to fuel burn.

4 Results

4.1 Payload-Range Charts

4.1.1 Cruise Altitude and Velocity

A cruise altitude of 43000 ft (13100 m) and cruise velocity of Mach 0.92 were selected based on a detailed analysis of performance requirements and comparable aircraft historical research. This cruise point satisfies the mandatory RFP requirement for a minimum cruise Mach of 0.85 and targets the enhanced Mach 0.92 performance goal for premium transcontinental operations. At 43000 ft, the reduced air density at high altitudes enables decreased parasite drag, improving overall fuel efficiency, while remaining below the maximum assumed service ceiling (45000 ft (13700 m)) to preserve margin for altitude maneuvering and ensure safe pressurization levels. The selected altitude also ensures that the aircraft operates within the compressible drag rise threshold, given the chosen airfoil geometry and sweep angle, minimizing transonic drag divergence effects. This cruise condition was cross-validated against thrust available curves and dynamic pressure limits to verify steady, level flight with acceptable stall margin and aerodynamic efficiency. The Mach 0.92 cruise velocity allows significant time savings over long-range missions without incurring prohibitive increases in specific fuel consumption or total drag, and thus represents an optimized balance between performance and operational cost. These values were used in all subsequent mission performance calculations, including range, aircraft geometry, fuel consumption, and drag analysis, and reflect the finalized baseline cruise condition for this design.

4.1.2 Range Trade Studies

Two range studies were completed for the HR One. The range was first traded against the AR and V. The range calculations are based on equation 38 below, the Breguet-Range Equation as it applies to jets, specifically to meet FAR requirements. Those requirements include holding cruise altitude, h, and velocity, V, constant for each calculation. Forced values included a minimum wing area of S = 99.2 m^2, simply calculated by

$$S_{min} = W_{TO}/qC_{Lcr} \tag{38}$$

and b = 30.95 m, found by a trade study described later in this report. These were enforced to find range at each AR and V, given a minimum wing size. Velocity was traded from 40 to 350 m/s, and AR from 5 to 15. Ultimately, AR = 9.68 and V = 0.92M = 271 m/s were chosen as points of interest for the HR One. Since this range was computed at cruise and to ensure the aircraft can withstand the worst conditions,

$$W_{fuel} = W_{TO} * \xi \tag{39}$$

and maximum payload weight, which occurs during the Napa Economic Mission at 1231.5 kg, were held constant. The coefficient of lift here was calculated using equation 12, the best-range lift equation. Since the aircraft is cruising above 0.3M, compressibility effects on drag must be considered, as discussed further in section 4.2.1. Therefore, E_m, which represents the lift to drag ratio, was found by looping through payload weight and calculating C_L and C_D for each weight step, as seen in the MATLAB code in Appendix A.

The range here, with maximum payload weight, was 9073 nautical miles (NM). The color bar in the plot represents increasing flight range, in meters, as it gets more yellow. This is highest at velocities in the 200-300 m/s range and at aspect ratios between 9 and 15. This is because low AR result in higher induced drag and a lower L/D, therefore decreasing range. However, at very high AR, the wave drag penalty stars to dominate the induced drag penalty. This also occurs around M > 0.9 At low speeds, the dynamic pressure value is much lower, and there is much higher fuel consumption per meter and a shorter range. The "sweet spot" falls in the range that aspect ratio and cruise velocity were selected at, marked by the red circle on the figure, validating HR One design decisions.

$$W_0^* = \frac{W_{TO}}{sqrt(C_{D0}/k) * q * S} \tag{40}$$

$$\text{Range} = \frac{2VE_m/SFC * arctan(W_0^* \xi)}{1 + W_0^{*2}(1 - \xi)} \tag{41}$$

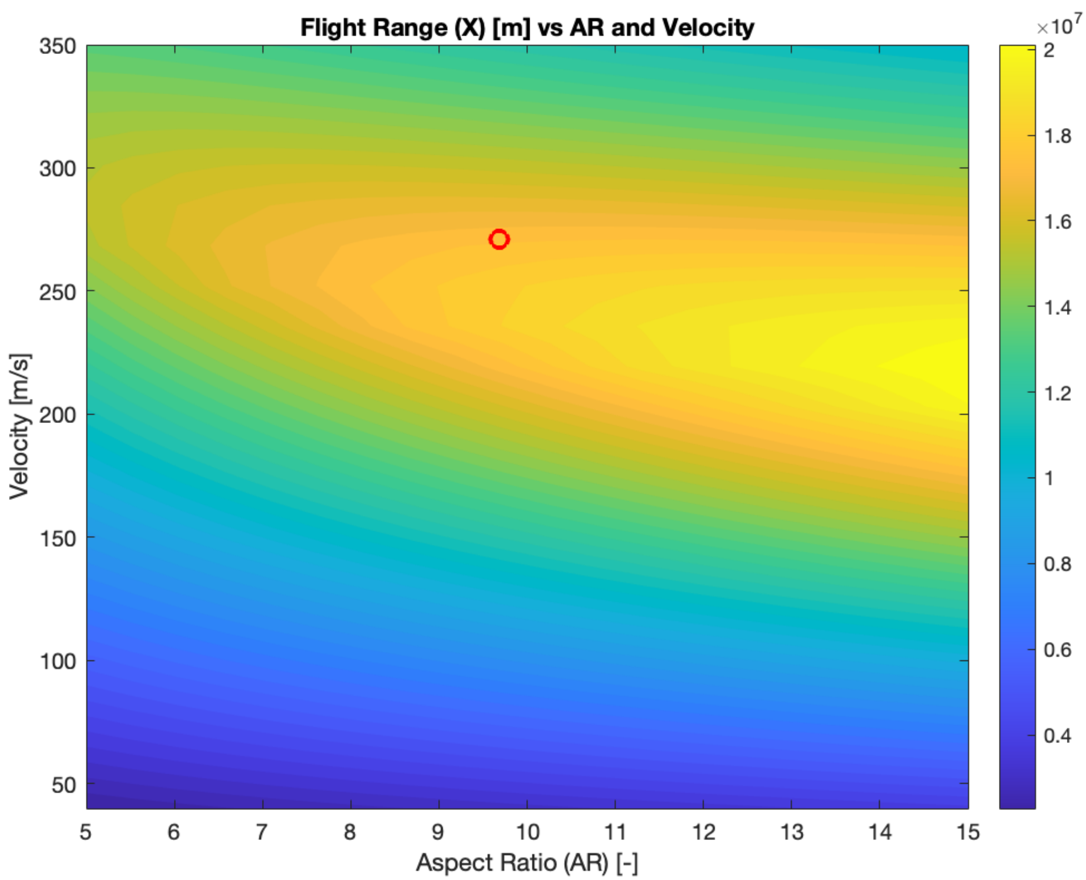


Figure 3: Range vs Payload Weight

Then, in figure 4 below, range was plotted against payload weight carried. The relationship can be seen in figure 4, with range decreasing linearly as more payload weight is carried. This is because to keep takeoff weight (WTO) constant, fuel weight decreases as payload weight increases. As the fuel weight carried changes for each payload weight step, the fuel fraction does as well, and is calculated by

$$\xi = W_{fuel}/W_{TO} \tag{42}$$

Velocity was taken at the cruise condition of 0.92M. Payload mass was calculated from 0 up to the maximum, 1232 kg. Customers can see this relation and understand that higher payload weight missions will decrease the fuel available and therefore range. However, both minimum and maximum range meet the passenger mission requirement of 8000 NM at 9765 NM and 9073 NM, respectively.

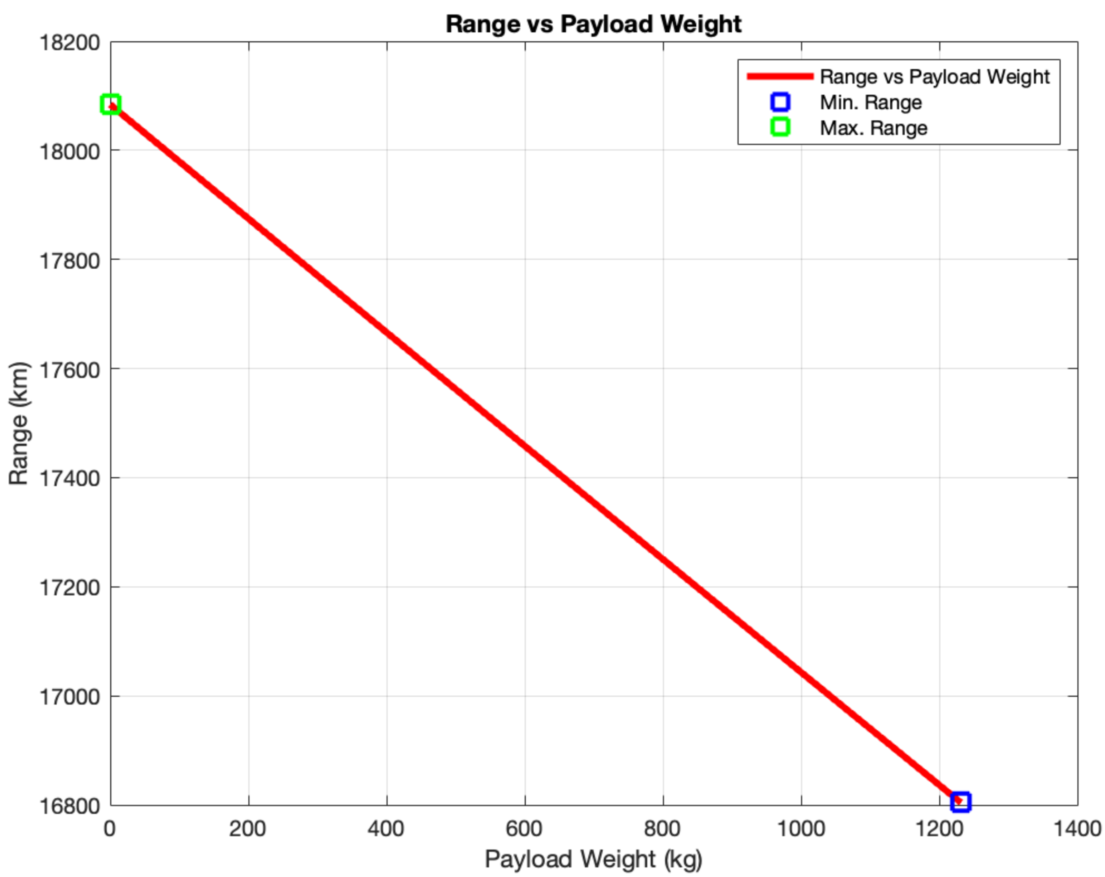


Figure 4: Range vs Payload Weight

4.1.3 Mission-Specific Range and Endurance

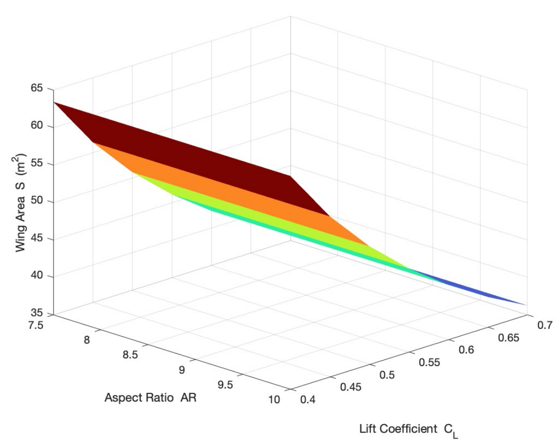
Range and endurance values were calculated using equations 16 and 17 of section 3.2.1. These can be seen in the table below.

Table 3: Endurance and Range for Various Mission Payloads.

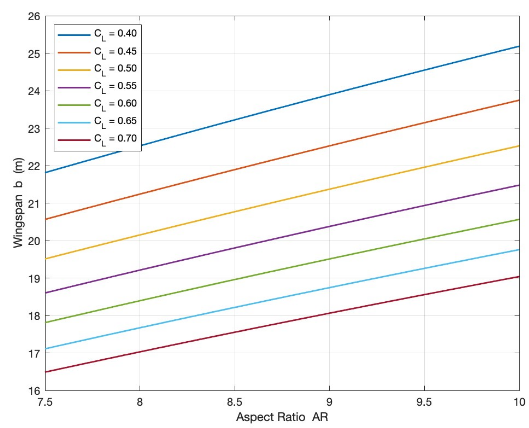
Mission	Payload (kg)	Endurance (hr)	Range (km)	Range (NM)
No payload	0.0	18.54	18084	9765
Aspen mission	573	18.52	18071	9758
Passenger mission	962	18.51	18058	9751
Napa economic mission	1232	18.50	18045	9743

4.2 Aerodynamic Characteristics and Performance

The design process began with an airfoil trade study as this selection provides various important parameters choice sets the allowable thickness–to–chord ratio, the drag-divergence Mach number, and the internal volume available for fuel and structure—parameters that cascade into every subsequent sizing trade. The airfoil family chosen is the NASA SC(2) airfoils, as they are supercritical and created by NASA. Most SC(2) polar data were generated at NASA Langley in the early-to-mid 1980s. Although no modern wind-tunnel or CFD campaign exists the SC(2) database remains the industry benchmark. A trade of the different C_L's of the airfoils using initial estimates of weight, cruise altitude and a velocity of $M = 0.92$ outputted required wingspan (b) and aspect ratio (AR). The following graphs were outputted:



(a) Wing Area vs AR & C_L

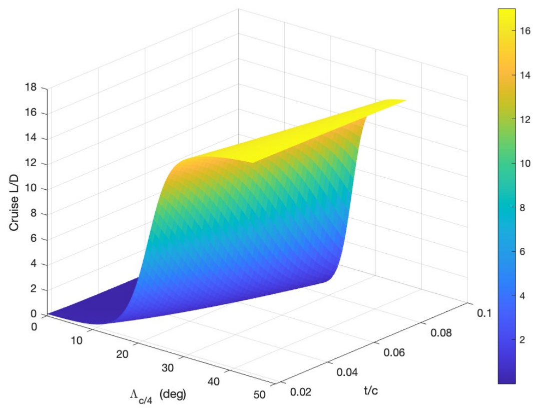


(b) Wingspan vs AR for various C_L

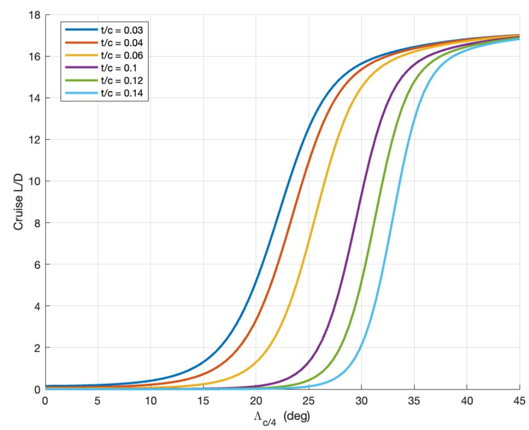
Figure 5: Relationship between Wing Area, Wingspan, Aspect Ratio, and Lift Coefficient

The resulting Figure 5a, shows a surface sloping downward with increasing AR because a longer span distributes the same load over more lifting line, while the color ramp steepens dramatically at higher C_L, showing how even small changes above the coefficient of lift would greatly increase the wetted area and therefore fuel burn. The data was then sliced through that surface at each C_L to obtain wingspan versus aspect-ratio lines in Figure 5b. These lines reveal a near-linear rule of thumb: every 0.5-point increase in AR adds roughly 0.8 m of span across the feasible region. This trade assisted in initial sizing and performance estimates by demonstrating ideal coefficient of lifts and accounting for the maximum span in the mission airports (KASE airport: $b_{max} = 29m$). Now it is important to find the ideal thickness-to-chord ratio and wing sweep.

Wing Sweep Trade Study An important aspect of reducing the compressible flow effects when flying at high Mach numbers is the supercritical airfoil, sweep of the wing and thickness-to-chord ratio. As shown in equation 7, wing sweep reduces the incident ΔC_{D_c} and $M_{cc,\Lambda}$. Therefore, a trade study was performed that compares $\frac{L}{D}|_{cruise}$ to changes in thickness to chord ratio (t / c) and wing sweep (Λ).



(a) Trade Surface: Cruise CL, t/c and wing sweep



(b) Cruise L/D vs Sweep (t/c bands)

Figure 6: Trade of the Wing Sweep of the HR One

The chosen combination is a wing sweep of 35 and a thickness to chord ratio of $t/c = 0.1$. The reason for this is due to the high cruise lift over drag value attained from this value. Furthermore, a thickness below 8 %pushes the spar caps and fuel volume below mission requirements, whereas a thickness above 12 % would force compressible wave drag to increase as indicated by equation [X]. Therefore the chosen ratio is $(t/c) = 0.10$. By combining these factors, the airfoil selected for this project is the **NASA-SC(2) 0610**:

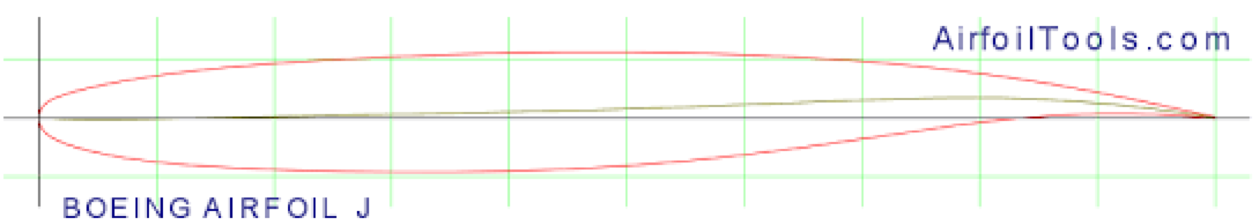
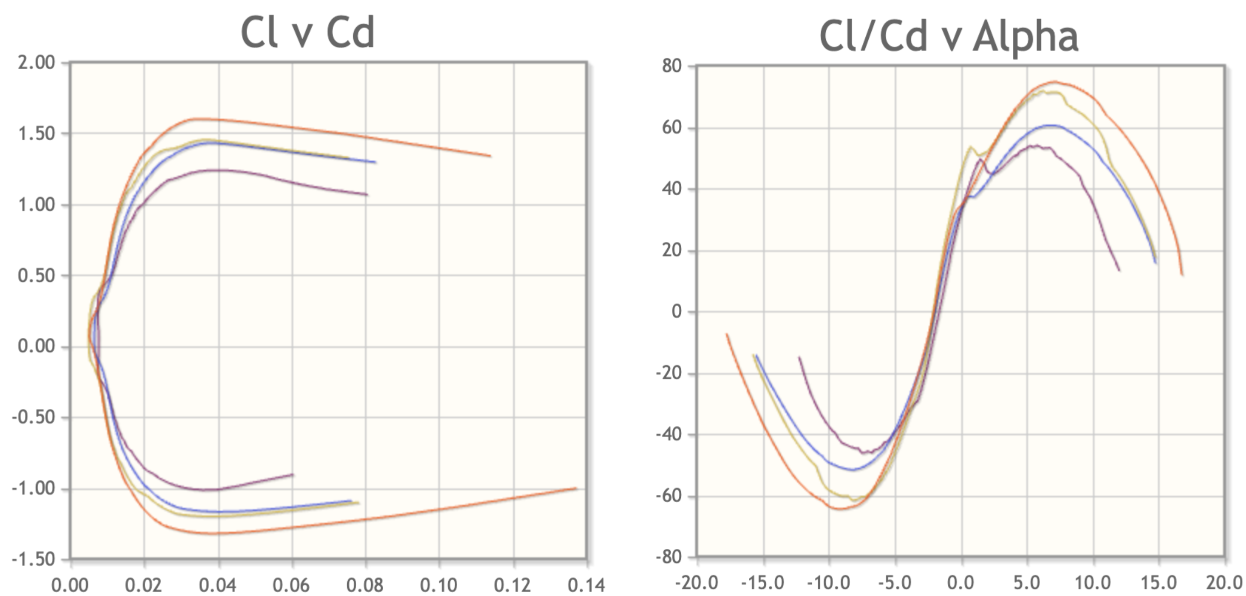


Figure 7: NASA-SC 0410

This airfoil offers a good balance of low drag at high speeds and structural convenience, making it well-suited for the long-range missions. Its moderate thickness ensures sufficient internal wing volume for structural elements and fuel storage, while its aerodynamic shape delays the onset of wave drag, helping reduce fuel consumption at high cruise Mach numbers.

As it is optimized for high-speed flight, the chosen airfoil has reduced low-speed performance compared to thicker or cambered sections. To address this, the aircraft incorporates advanced high-lift devices like flaps and slats. These enhancements significantly boost lift during take-off and landing phases, ensuring safe, comfortable, and efficient operation at all stages of flight.

4.2.1 Airfoil Performance



(a) Drag vs Velocity

(b) Drag vs Mach Number

Figure 8: Demonstration of the Drag Buildup for a Supercritical Airfoil for HR One

4.2.2 Drag Buildup

The bar chart below demonstrates the buildup of HR One's zero-lift parasite drag, C_{D_0}:

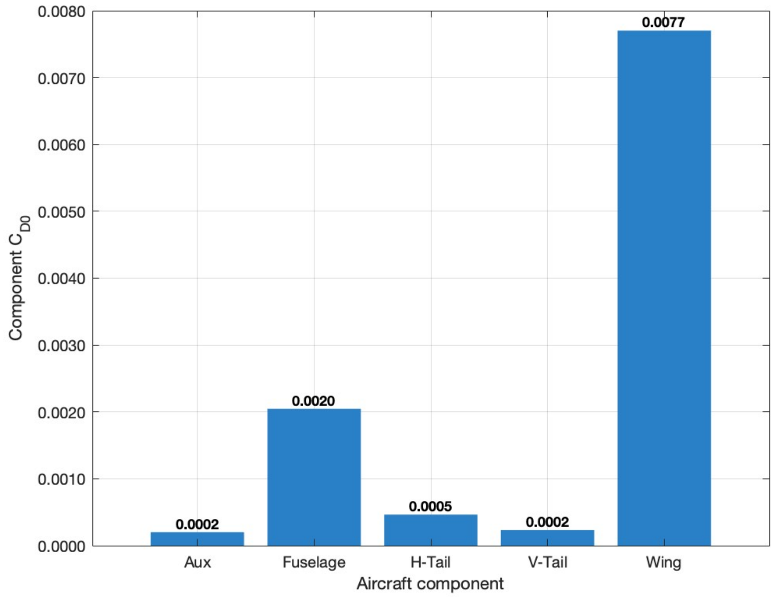
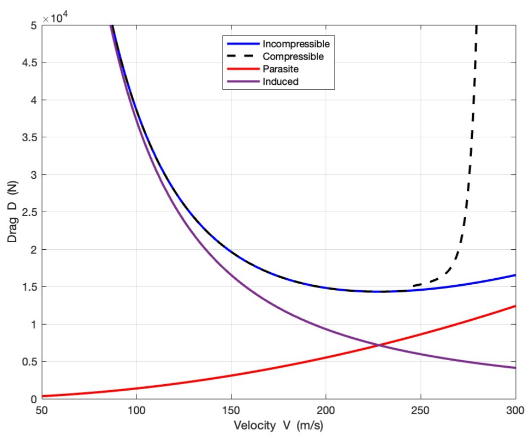
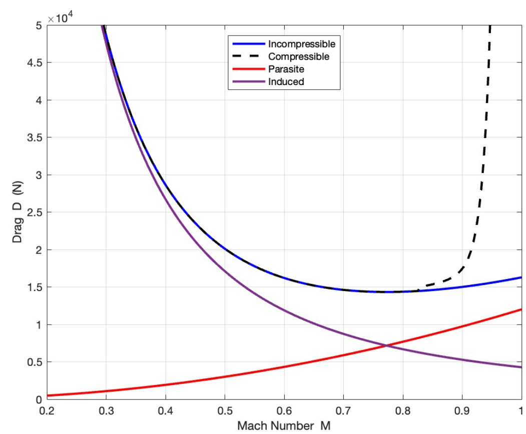


Figure 9: Component Breakdown of Parasitic Drag

As demonstrated the wing and fuselage account for most of the parasitic drag, while the empennage components and auxiliary approximation accounts for the rest to give a parasitic drag coefficient of $C_{D_0} = 0.0107$. It is important to note that the HR One will incorporate Hybrid Laminar Flow Control that will help maintain laminar flow throughout a portion of the wing. This technology has been tested previously and has been implemented in the tail of the Boeing 787 Dreamliner [?] [0] [0]. This impact was measured by calculating the laminar friction coefficient $C_{f,l} = \frac{1.328}{\sqrt{Re}}$ and accounting for a conservative 10% of the wing. This allows the parasitic coefficient to drop from 0.0130 to 0.0107.



(a) Drag vs Velocity



(b) Drag vs Mach Number

Figure 10: Demonstration of the Drag Buildup for a Supercritical Airfoil for HR One

Figure 10 demonstrates the drag vs speed including the compressible effects at cruise altitude of $43,000$ft (13100 m). Figure 10b demonstrates the mach number effect, which demonstrates the impact of supercritical airfoils. As the drag divergence Mach number is $M \approx 0.85$.

4.3 Propulsion System Description and Characterization

The HR One will be powered by two $\tilde{8}$,000 lbf-class high-bypass turbofan engines based on NASA's Hybrid Thermally Efficient Core (HyTEC), scaled from the 30,000 lbf demonstrators that are currently under development. This HyTEC program was developed in a partnership between NASA, GE Aerospace, and Pratt & Whitney to increase core efficiency through higher pressure ratios, improved thermal management, and hybrid-electric integration. Although the core remains in development, targeting all technologies at the Technology Readiness Level of 6 by 2028, this proposal assumed that its minimum performance targets detailed in Figure 8 are met and can be scaled down for use in the business jet thrust class.

Table 4: HyTEC Key Performance Parameters (KPPs)

Performance Parameter	**Full Success**	**Minimum Success**
Fuel burn reduction from core (vs 2020 SOA)	10%	5%
Engine bypass ratio (BPR)	>15	>12
Overall pressure ratio (OPR)	>50	>45
Durability (time between major refurbishment)	>5% over SOA	Meet 2020 SOA
Hybridization: power extraction at altitude (turbofan level)	20%	10%
HPC exit corrected flow	<3.0 lbm/s	<3.5 lbm/s
Combustor operability with SAF	100% SAF compatible	>80% SAF compatible
Hybridization: power extraction/insertion from core	10%	5%

To account for the minimum success of a 5% fuel burn reduction, the Specific Fuel Consumption (SFC) for a comparable engine by thrust class, the Rolls Royce Pearl 15, was decreased by 5%, resulting in an SFC value of 0.0610 N fuel/N Thrust hr.

4.3.1 Propulsion System Integration

Integrating this propulsion system will have a few key differences from similar engines in the business jet thrust class. The HyTEC-based engine is expected to achieve a bypass ratio of 12:1 and an overall pressure ratio above 45:1. To accommodate the increased flow, a larger fan diameter must be used therefore increasing the nacelle diameter, lightly increasing drag.

This integration will also add complexity in thermal management due to the smaller, hotter core with electric components integrated within the nacelle and wing root.

4.3.2 HyTEC Competitive and Long-Term Advantage

Despite adding integration complexity, the HyTEC system's advantages outweigh the added complexity. The hybrid architecture allows for power extraction and power insertion to support climb and descent assist, onboard systems, and other power demands. The high bypass and pressure ratios contribute to reduced block fuel consumption and noise, and the design aligns with the global push to reduce emissions and increase sustainability in aviation. The HyTEC core's high pressure ration and low exit flow show that it is more thermally efficient than comparable business jet engines, specifically this shows that it extracts more energy per unit of airflow making it a desirable choice.

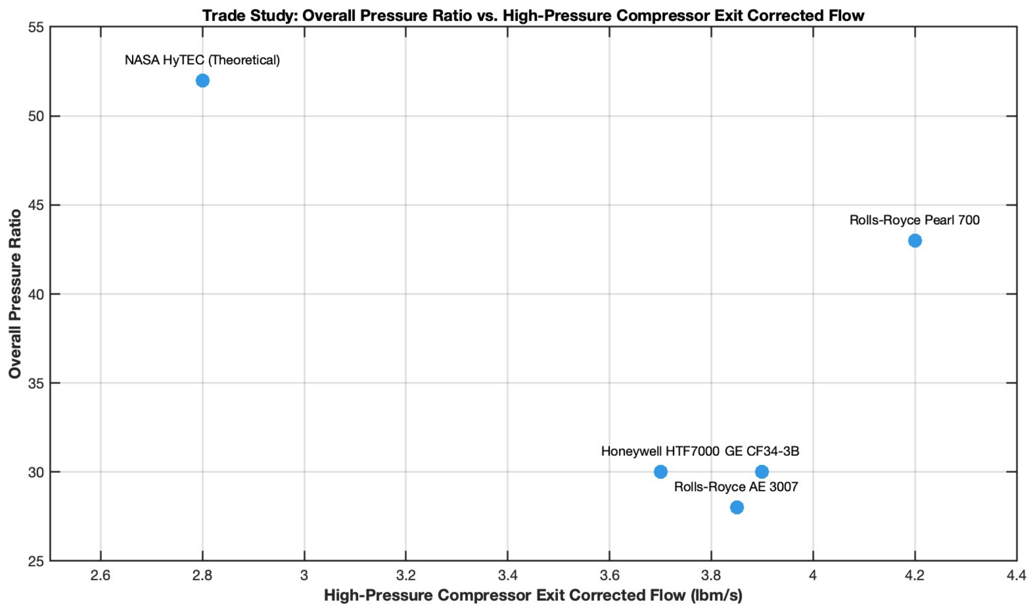


Figure 11: Overall Pressure Ratio vs High Pressure Compressor Exit Corrected Flow for Business Jet Engines

4.3.3 Risk Mitigation with Experimental Technologies

To mitigate the risk associated with relying on an in-development engine core, an alternate configuration has been considered. If HyTEC does not meet its minimum performance or readiness targets in time for integration, a fallback propulsion option such as the Rolls-Royce Pearl 15 could be implemented with minor changes to the nacelle and pylon geometry. This would remove the aircraft's competitive fuel burn and hybridization benefits, but would be a viable path to certification in the case that the technology does not adequately develop. The Pearl 15 has a similar thrust class, slightly higher SFC, and lighter weight, making it a reasonable fallback option to reduce overall program risk.

4.4 Aircraft Performance

4.4.1 Flight Envelope

An important factor to analyze the performance of the airplane is the Flight Envelope demonstrated in Figure 12 below.

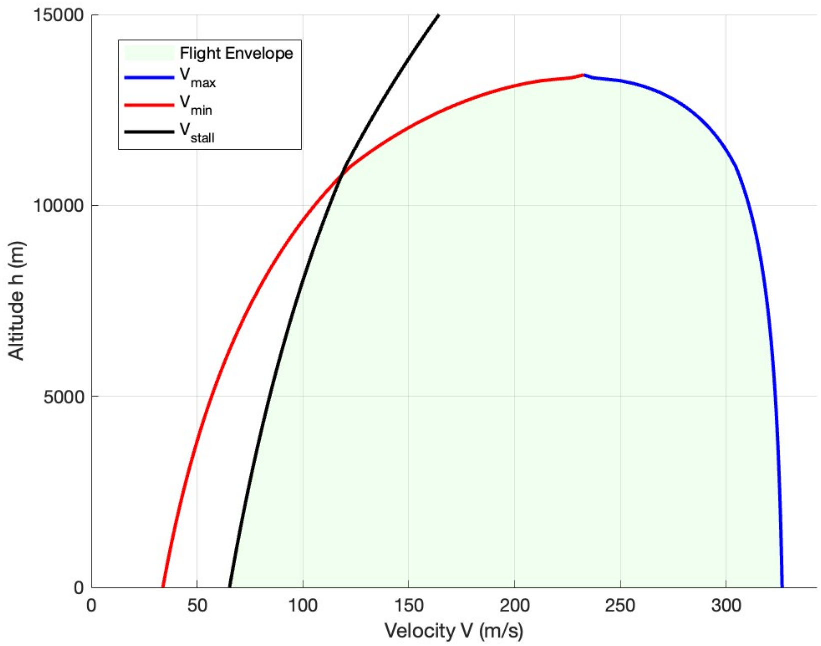
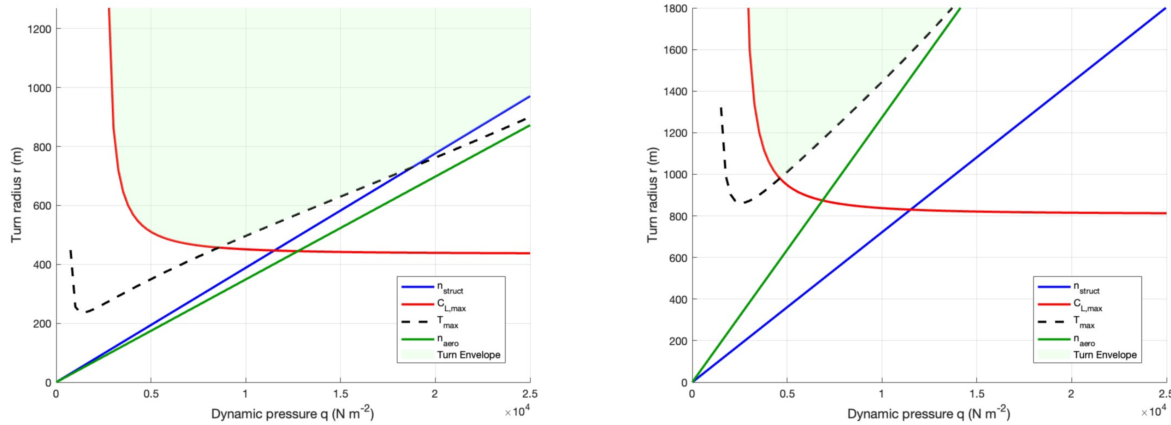


Figure 12: HR One Flight Envelope

Figure 12 demonstrates the absolute ceiling of 13,700m ($\approx$ 45000ft). Beyond that it demonstrates the wide operating range of the HR One At lower speeds, the stall speed is limiting as expected but due to the aerodynamic design it does not account for a large part of the envelope.

4.4.2 Turning Envelope

Based on the structural load limit found in Raymer, it was assumed that the structural limit for the HR One is $n_{struct} = 4.4$. This indicates that the airplane can handle a 3g load requirement. The following graph represents the dyanamic pressure and turn radius performance of the HR One at sea level and 6000m



(a) HR One Turning Envelope at Sea Level

(b) HR One Turning Envelope at 6000m

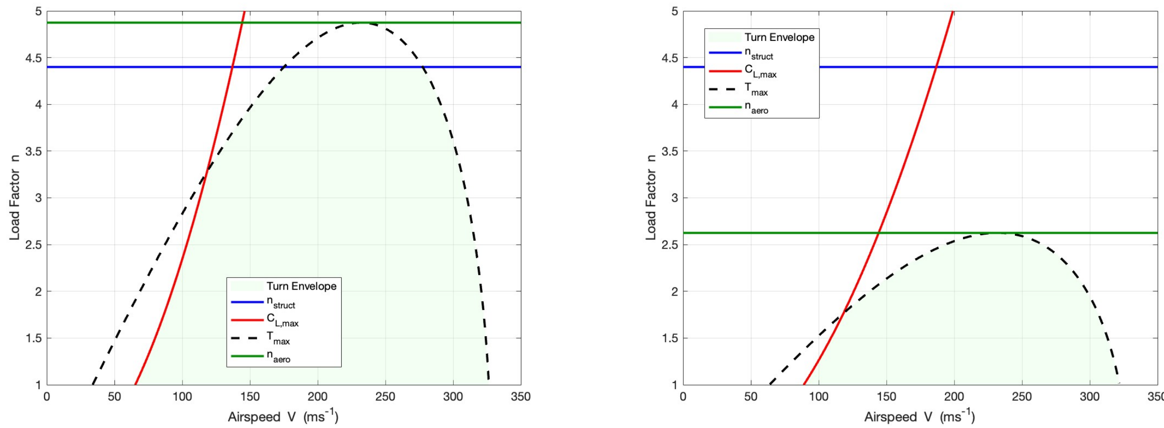
Figure 13: Demonstration of the r-q Turning Envelope at Various Heights for HR One

Figure 13a demonstrates that the limit set by the aerodynamics of the airplane does not effect the turning envelope and therefore only depends on $C_{L_{max}}$, T_{max} and partially n_{struct} at Sea Level. Figure 13b demonstrates that the structural and aerodynamic limits set by the geometry of the airplane does not effect the turning envelope and therefore only depends on $C_{L_{max}}$ and T_{max}. The following table demonstrates minimum turning values at the different altitudes:

Altitude h (m)	**Minimum Turn Radius r_{min} (m)**
0	457
6000	988

Table 5: Minimum Turning Radius of HR One at Sea Level and 6000m

It is important to note that passenger comfort and safety is HR One's utmost priority therefore, minimizing the turn radius significantly isn't the most important performance aspect of the plane.The following V-n diagram demonstrates the effect of velocity on the turning envelope.



(a) V-n Graph at Sea Level

(b) V-n Graph at 6000m

Figure 14: Demonstration of the V-n Turning Envelope at Various Heights for HR One

It is clear that the $C_{L_{max}}$ is the limiting factor at low velocities while at higher velocities the T_{max} is the limiting factor. At higher altitudes, the aerodynamic limit constrains the turning envelope. It is also important to note that the V-n graphs begin at $n = 1$ as this is the minimum value in which the plane can fly.

4.4.3 Takeoff and Landing Performance

The following tables describe the performances for this critical aspect of flight. All takeoff calculations are done at MTOW at wing-to-ground height of 2 meters which clears the engine diameter. The landing calculations are done at the minimum landing weight from the weight approximation.

Table 6: Take-off Performance Summary

Mission	Departure	Elev. (m)	Condition	Max. (m)	Calc. (m)
Design	SLS	0	Dry, ISA	1 829	1 760
Aspen	KVNY	244	15C	2 440	1 850
Napa	KAPC	11	24C	1 810	1 770

Take-off Performance

Table 7: Landing performance summary

Mission	Arrival	Elev. (m)	Condition	Max. (m)	Calc. (m)
Design	SSL	0	Dry, ISA	1 829	1 440
Aspen	KASE	2 389	$-7°$C, μ=0.16	2 440	2 120
Napa	MMMX	2 230	29°C	3 990	2 050

Landing Performance It is clear that the HROne can takeoff and land at each of the required airports.

4.4.4 Rate of Climb

The figure and table below demonstrates the rate of climb performance of the HROne.

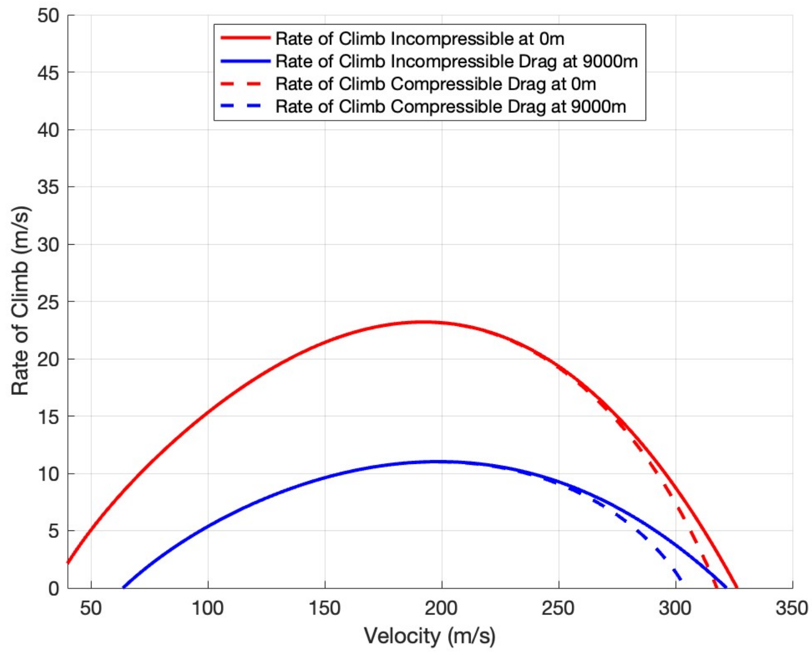


Figure 15: Rate of Climb at Sea Level and 6000m for HR One

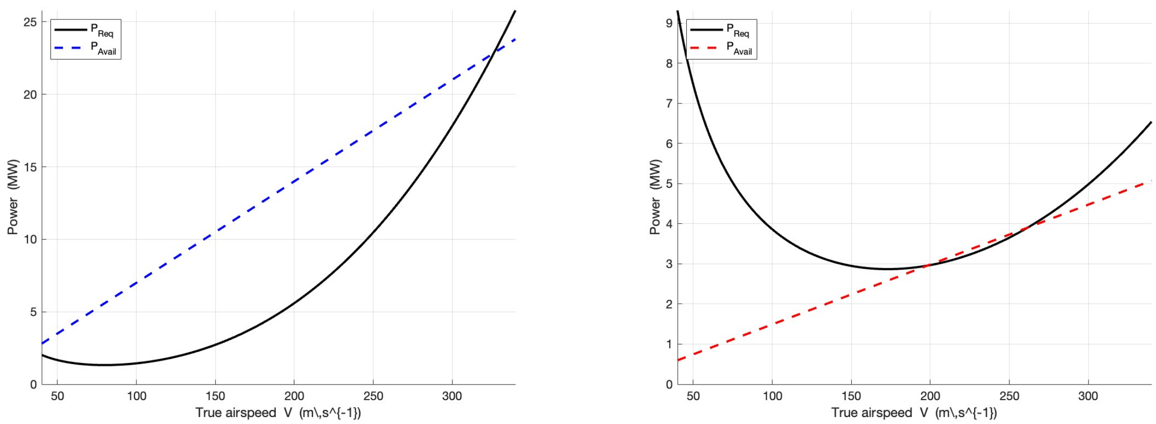
Altitude h (m)	**Maximum Rate of Climb (m/s)**
0	23.2
6000	11.0

Table 8: Maximum Rate of Climb of HR One at Sea Level and 6000m

Figure 15 demonstrates the rate of climb envelopes at sea level and 6000m accounting for both compressible and incompressible drag. The maximum values of $23.2m/s$ at sea level and 11.0m/s at 6000m. The broad shape of the RoC allows for pilots to climb to the cruise altitude and demonstrates climb efficiency due to the limited loss in maximum climb value.

4.4.5 Power Required and Power Available

The power graph demonstrates the HROne's engine capabilities.



(a) Power Required at Sea Level

(b) Power Required at Cruise Altitude

Figure 16: Demonstration of the Power Required and Available at Sea Level and Cruise

At sea level there is an abundance of power which connects to HROne's ability to climb at a high rate shown in Section 4.4.4. Furthermore, the cruise demonstrates minimal excess power due to the altitude effects and drag of the HROne. However, this value is at MTOW and early in cruise. As the HROne burns fuel, the power required will decrease and allow the pilots to reduce the throttle and reduce fuel consumption and emission.

4.5 Materials Selection and General Structural Design, Including Layout

This aircraft is built with composite materials to maximize strength and minimize weight. The wetted areas of the fuselage and wing will be made of carbon laminate materials for a smoother finish. The internal components such as wing ribs that withstand higher loads are comprised of carbon sandwich panels.

The composite ratios can be projected using the following figure .

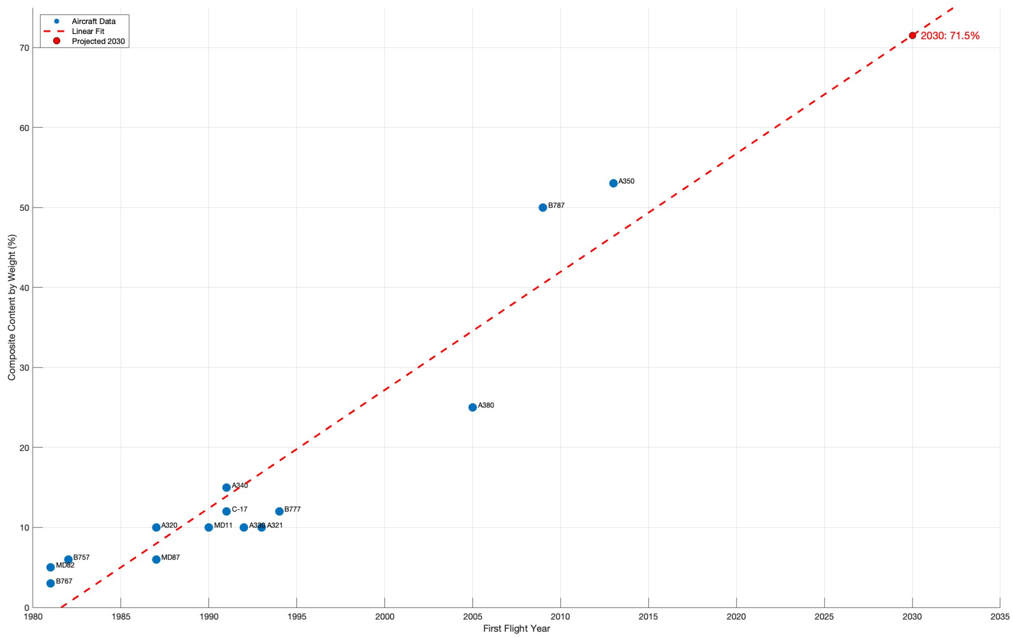


Figure 17: Composite ratios for historical aircraft [10]

Figure 16 projects the composite ratio for the HR One to be 71.5% at 2030.

4.6 Aircraft Weight Statement, Aircraft CG Envelope, and FWD/AFT CG Limits

4.6.1 Aircraft Mass Build Up

Using this same method, a mass breakdown of the following components was conducted: Airframe structure (wing, empennage, fuselage, landing gear), propulsion and both aircraft systems and control systems. Constants are derived from statistical weight estimates from similar aircraft, with unique parameters for the HR One defined in Table 9. The final mass build up supports the initial estimate for MTOW. The equations below were taken from Raymer, and ran in the appendix MATLAB script [4].

$$W_{\text{wing}} = 0.036\, S_w^{0.758}\, W_{fw}^{0.0035} \left(\frac{A}{\cos^2\Lambda}\right)^{0.6} q^{0.006} \lambda^{0.04} \left(\frac{100\, t/c}{\cos\Lambda}\right)^{-0.3} (N_z W_{dg})^{0.49} \tag{43}$$

$$W_{\text{HT}} = 0.016\, (N_z W_{dg})^{0.414}\, q_{\text{ht}}^{0.168} S_{\text{ht}}^{0.896} \left(\frac{100\, t/c}{\cos\Lambda_{\text{ht}}}\right)^{-0.12} \left(\frac{A_{\text{ht}}}{\cos^2\Lambda_{\text{ht}}}\right)^{0.043} \lambda_{\text{ht}}^{-0.02} \tag{44}$$

$$W_{\text{VT}} = 0.073 \left(1 + 0.2\frac{H_t}{H_v}\right) (N_z W_{dg})^{0.376}\, q_{\text{vt}}^{0.122} S_{\text{vt}}^{0.873} \left(\frac{100\, t/c}{\cos\Lambda_{\text{vt}}}\right)^{-0.49} \left(\frac{A_{\text{vt}}}{\cos^2\Lambda_{\text{vt}}}\right)^{0.357} \lambda_{\text{vt}}^{0.039} \tag{45}$$

$$W_{\mathrm{MLG}} = 0.095\,(N_l W_l)^{0.768}\left(\frac{L_m}{12}\right)^{0.409} \tag{46}$$

$$W_{\mathrm{NLG}} = 0.125\,(N_l W_l)^{0.566}\left(\frac{L_m}{12}\right)^{0.845} \tag{47}$$

$$W_{\mathrm{fuse}} = 0.052\,S_f^{1.086}\,(N_z W_{dg})^{0.177}\,L_t^{-0.051}(L/D)^{-0.072} q_{\mathrm{fuse}}^{0.241} + W_{\mathrm{press}} \tag{48}$$

$$W_{\mathrm{ENG}} = 2.575\left(\frac{W_{\mathrm{en}}}{\mathrm{lb}}\right)^{0.922} N_{\mathrm{en}} \tag{49}$$

$$W_{\mathrm{controls}} = 0.04\,W_e \tag{50}$$

$$W_{\mathrm{systems}} = 0.09\,W_e \tag{51}$$

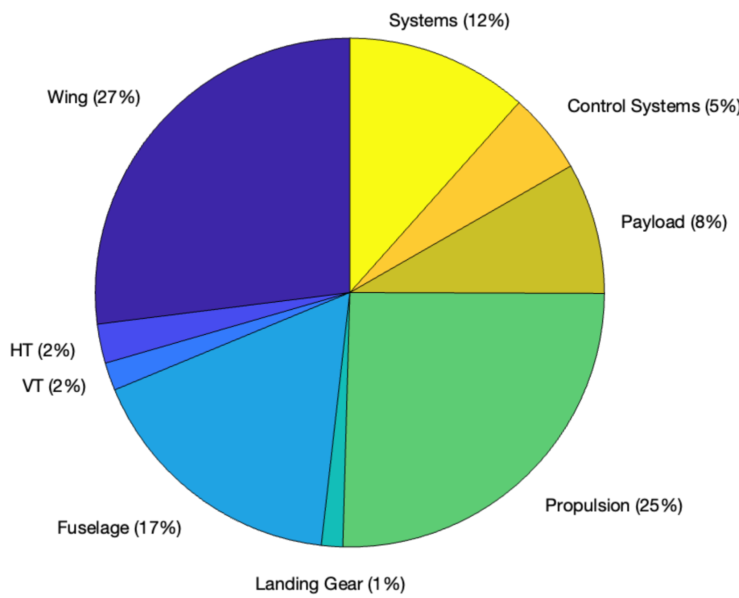


Figure 18: HR One Mass Breakdown at Max Payload

Table 9: Defined Design Parameters for Aircraft Mass Breakdown

Parameter	**Symbol and Value**
Empty weight	$W_e = 18998.1$ kg
Wing area	$S_w = 99.2$ m^2
Wing fuel weight	$W_{fw} = 18506$ kg
Wing aspect ratio	$A = 9.679$
Wing sweep angle	$\Lambda = 35°$
Dynamic pressure (wing)	$q = 1$ Pa
Wing taper ratio	$\lambda = 0.4165$
Thickness-to-chord ratio	$t/c = 0.1$
Ultimate load factor	$N_z = 4.4$
Design gross weight	$W_{dg} = 38745.86$ kg
Horizontal tail area	$S_{ht} = 19.84$ m^2
Horizontal tail AR	$A_{ht} = 7.75$
Horizontal tail sweep	$\Lambda_{ht} = 40°$
Horizontal tail taper ratio	$\lambda_{ht} = 0.4165$
Dynamic pressure (HT)	$q_{ht} = 9772.4$ Pa
Vertical tail area	$S_{vt} = 9.92$ m^2
Vertical tail AR	$A_{vt} = 3.87$
Vertical tail sweep	$\Lambda_{vt} = 10°$
Vertical tail taper ratio	$\lambda_{vt} = 0.4431$
HT height above fuselage centerline	$H_t = 1.375$ m
Fuselage height	$H_v = 2.75$ m
Dynamic pressure (VT)	$q_{vt} = 9772.4$ Pa
Fuselage wetted area	$S_f = 216.77$ m^2
Tail moment arm	$L_t = 12.005$ m
Lift-to-drag ratio	$L/D = 20$
Dynamic pressure (fuselage)	$q_{\text{fuse}} = 9772.4$ Pa
Pressurization weight	$W_{\text{press}} = 1$ kg
Number of landing gear struts	$N_l = 3$
Weight supported per strut	$W_l = 12915.29$ kg
Main gear length	$L_m = 1.5$ m
Single engine weight	$W_{en} = 1360$ kg
Number of engines	$N_{en} = 2$
Payload weight	$W_{\text{payload}} = 1231.3$ kg

4.6.2 Wing Moment Calculation

A static force calculation was performed on the wing for the maximum structural moment sustained in the wing. This maximum moment is at the root of the wing. The following diagram shows the forces on the wing, with the wing modeled as a simple beam. The weight of the wing is estimated using the methods outlined in Raymer's textbook [4].

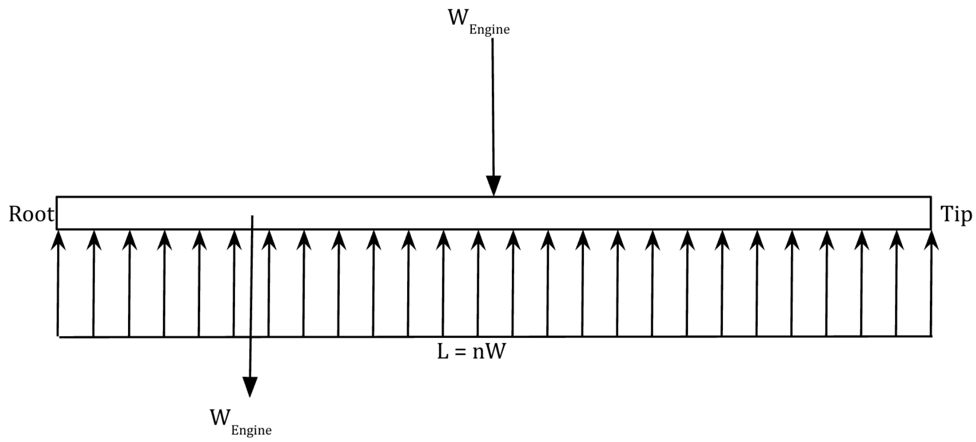


Figure 19: Wing force diagram

With the parameters of the aircraft and the MATLAB script in the Appendix A.16, the moment at the root of the wing was calculated to be the following: $M_{root} = 4902.5kNm$

4.6.3 FWD/AFT CG Limits and CG Envelope

To resolve the FWD and AFT CG limits, first state the non-negotiable values that are pre-determined from aerodynamics and stability/control:

- $C_{M_0} = 0.0359$
- $C_{L_{max}} = 1.4$
- $a_w = 5.08$
- $a_t = 3.95\frac{1}{rad} = 0.0689\frac{1}{degree}$
- $e = 0.90$
- $\eta_t = 0.90$
- $h_{ac} = 0.25$
- $\bar{c} = 3.26$
- $\tau = 0.50$

To obtain $\dfrac{\partial \varepsilon}{\partial \alpha}$, solve Eq. (23):

$$\frac{\partial \varepsilon}{\partial \alpha} = \frac{2(5.08)}{\pi(0.9)(9.68)} = 0.382$$

For a relatively small private jet, to define CG limits, as well as a target CG, a tail volume ratio needs to be selected. Typical values of V_t range from around 0.5 to 1.0 [8]; therefore, selecting a middle value of $V_t = 0.75$ is a good choice.

Now, the neutral point can be solved:

$$h_n = 0.25 + \frac{3.95}{5.08}(1 - 0.382)(0.9)(0.75) = 0.574$$

The dimensional neutral point, x_n is obtained by multiplying h_n by $\bar{c}$:

$$x_n = 0.574 \times 3.26 = 1.87 \text{ meters behind the LE of the wing}$$

To locate the FWD CG, first resolve $\frac{\partial C_{M_{\mathrm{cg}}}}{\partial C_L}$. A negative $\frac{\partial C_{M_{\mathrm{cg}}}}{\partial C_L}$ is absolutely necessary for stability. To find this slope, first solve for $\frac{\partial C_{M_{\mathrm{cg}}}}{\partial \delta_e}$:

$$\frac{\partial C_{M_{\mathrm{cg}}}}{\partial \delta_e} = -0.0689(0.9)(0.75)(0.50) = -0.0233$$

Now, at the most FWD CG location, HR One must maintain equilibrium. At equilibrium, and max elevator deflection of -15 degrees, solve for $\frac{\partial C_{M_{\mathrm{cg}}}}{\partial C_L}$:

$$0 = C_{M_{\mathrm{cg}}} = C_{M_o} + \frac{\partial C_{M_{\mathrm{cg}}}}{\partial C_L} \cdot C_{L_{max}} + \frac{\partial C_{M_{\mathrm{cg}}}}{\partial \delta_e} \cdot \delta_{e,max}$$

$$0 = 0.0359 + (-0.2)\frac{\partial C_{M_{\mathrm{cg}}}}{\partial C_L} + (-0.0233)(-15) \Rightarrow \frac{\partial C_{M_{\mathrm{cg}}}}{\partial C_L} \approx -0.275$$

Now, with $\frac{\partial C_{M_{\mathrm{cg}}}}{\partial C_L}$ solved, the FWD limit can be resolved.

$$h_{cg} - h_{ac} = \frac{\partial C_{M_{\mathrm{cg}}}}{\partial C_L} + \frac{a_t}{a_w}(1 - \frac{\partial \varepsilon}{\partial \alpha})\eta_t V_t.$$

$$h_{cg} - h_{ac} = -0.275 + \frac{3.95}{5.08}(1 - 0.382)(0.9)(0.75) = 0.0494$$

Multiply $h_{cg} - h_{ac}$ by $\bar{c}$ to achieve the dimensional FWD distance behind the LE of the wing:

$$x_{FWD} = 0.0494 \times 3.26 = 0.161 \text{ meters behind the LE of the wing}$$

Therefore, $x_{FWD} = 0.161$ meters behind the LE of the wing and $x_n = 1.87$ meters behind the LE of the wing.

The CG envelope can be simply visualized as 2 vertical lines, with the forward line denoting the FWD CG limit and the latter line denoting the neutral point, or the AFT CG limit, both as a percentage of the mean aerodynamic chord ($\bar{c}$).

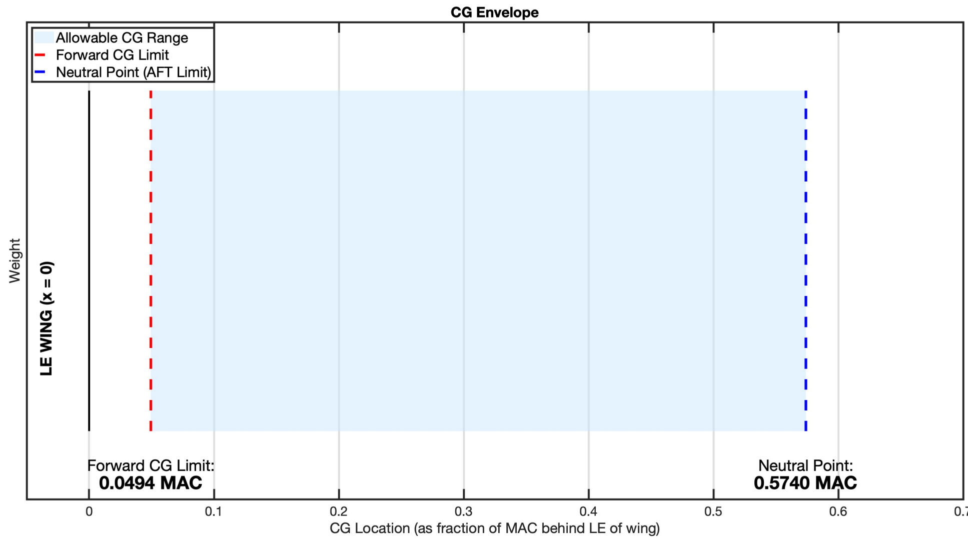


Figure 20: CG Envelope of the HR One

4.7 Stability and Control

Moment calculations on the center of gravity was done to ensure stability during cruise. The tail volume ratio was calculated first using equation .

$$V_t = \frac{S_t}{S_w} \times \frac{l_t}{c_w} = 0.5345 \tag{52}$$

Next, equation 41 was used to calculate the moment coefficient at the aerodynamic center.

$$C_{M_0} = C_{M_{ac}} + \alpha(\epsilon_0 + i_w - i_t)(V_t)(\eta_t) \tag{53}$$

Values at trimmed cruise ($C_{M_{cg}} = 0$) and other aircraft parameters were substituted.

$$C_{M_0} = C_{M_{ac}} + 0.1(1\,\text{deg} + 1.5\,\text{deg} - 1\,\text{deg})(0.5345)(0.9) \tag{54}$$

Equation 41 can be rearranged to find $C_{M_{ac}}$, and substituted into equation 43.

$$C_{M_{cg}} = C_{M_{ac}} + C_L(h_{cg} - h_{ac_w}) - CL_t\eta_t V_t \tag{55}$$

Substituting aircraft geometry parameters result in $C_{M_{cg}} = 0$, confirming a trimmed configuration.

4.8 Avionics and Instrumentation

Fly-by-wire systems are integrated into the HR One for safe and precise controls. Fly-by-wire systems hold advantages in lighter weight and ease of integration into the aircraft.

14 CFR Part 25 outlines the specific requirements for avionics systems to ensure safety and reliability. The HR One will house instrumentation specified in Table 10. In accordance with regulations, all avionics will be protected from lightning strikes and electromagnetic radiation.

Required Instruments
Airspeed Indicator
Altimeter
Magnetic Direction Indicator
Clock
Free Air Temperature Indicator
Gyroscopic Rate-of-Turn Indicator
Slip-Skid Indicator
Attitude Indicator (Pitch and Bank)
Heading Indicator (Directional Gyro)
Vertical Speed Indicator (VSI)

Table 10: Required Flight and Navigation Instruments under 14 CFR § 25.1303

Ice Protection Systems The HR One will comply with 14 CFR Part 25 icing requirements, including:

- **Airframe (§ 25.1419)**: Analysis and flight or lab testing per Appendix C; crew alerts for malfunctions; documented activation and cycling procedures.
- **Propeller (§ 25.929)**: Means to prevent or remove ice on all propellers; fluid-system compliance if using combustible-fluid de-icing.
- **SLD (§ 25.1420)**: Safe-operation demonstration in Appendix O conditions, with ice detection and prescribed exit procedures.

4.9 Geometry

Since the aircraft will be cruising at M > 0.3, compressibility effects must be considered. Specifically, these effects place heavy emphasis on AR and Λ. Additionally, the amount of lift necessary can be calculated by $L = W$, taking MTOW as W, then $S = \frac{L}{qC_{Lcruise}}$, where $C_{Lcruise}$ is 0.4, determined by the airfoil. This yielded a constraining value of $S_{min} = 99.2$ $[m^2]$. AR was traded from 6 to 11 [-] based off market research of existing private jets and limited at 11 to avoid structural concerns and excessive wing weight. Span values were calculated at each AR value, as $b = \sqrt{(AR * S)}$. The span is constrained by the width of the smallest gate of all intended airports, 28.96 [m] at KASE, multiplied by its potential to increase by utilizing folding wingtips: $b_{max} = 28.96 * 1.07$. Maximum payload weight was used here, and cruise altitude and velocity were held constant. The t/c ratio was a constant 0.1 multiplied by 0.6 to account for the supercritical airfoil.

AR and b were traded against a fuel burn metric to find the minimum weight of fuel burn per weight of payload per meter, which fits all constraints mentioned above. The Breguet Range equation for best range was used to calculate range to obtain fuel burn metric:

$$\text{Range} = V/SFC_j * (C_L/C_D) * ln(W_{TO}/W_1) \tag{56}$$

This fuel burn metric is calculated by equation 43,

$$\text{fuel burn metric} = \frac{W_{fuel}/W_{payload}}{Range} \tag{57}$$

Results can be seen in Figure 16 below. The best fuel burn metric was found to be 1.05×10^{-6} per meter at an AR of 9.68 [-] and b of 30.95 [m]. Fuel burn metric decreases with increasing aspect ratio because AR decreases induced drag. With CL and e held constant, C_{Di} $1/AR$. This is prominent until about an AR of 9, where one starts to see diminishing returns on Figure 16. As AR increases past here, fuel burn metric does not decrease as rapidly and therefore is not worth the structural penalty of having a higher aspect ratio and having to increase structural weight.

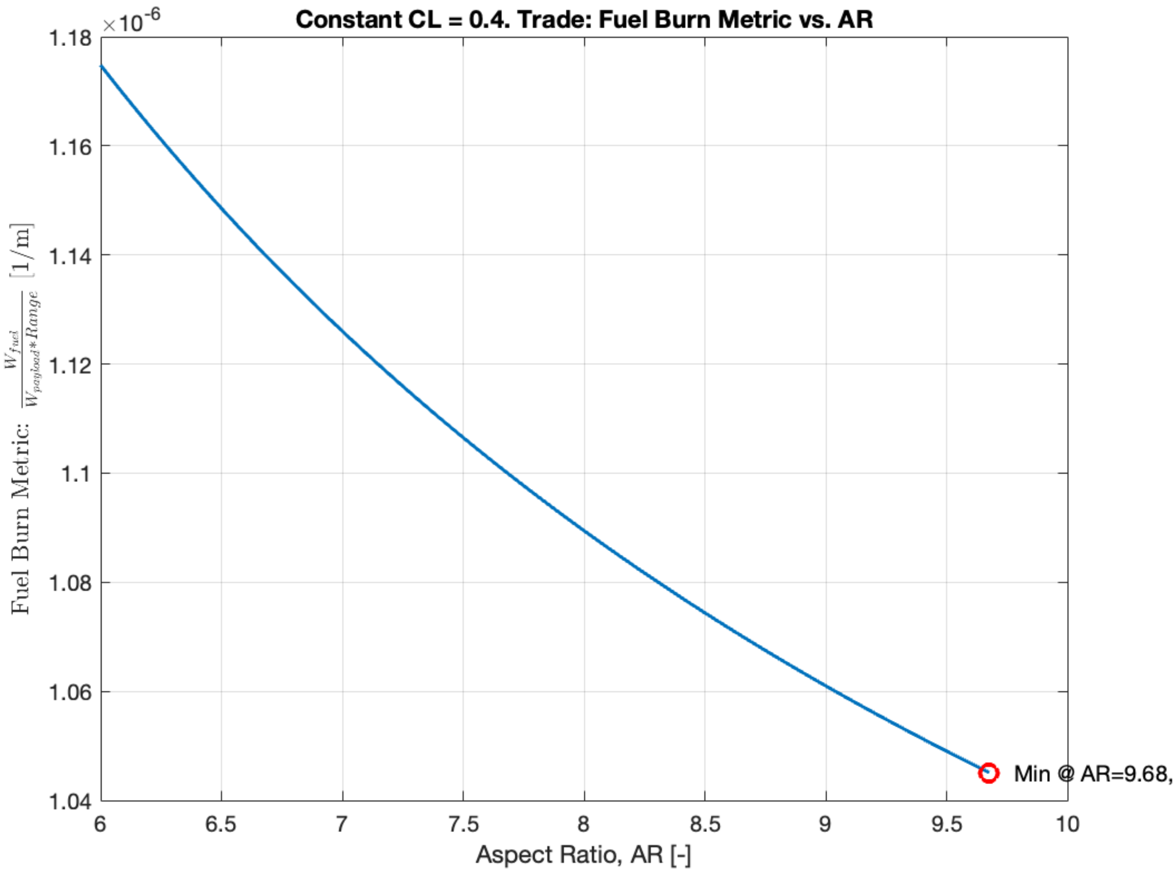


Figure 21: Fuel Burn Metric vs AR

4.10 CAD and Geometry

4.10.1 Three-Views + Isometric View: Scaled and Dimensioned

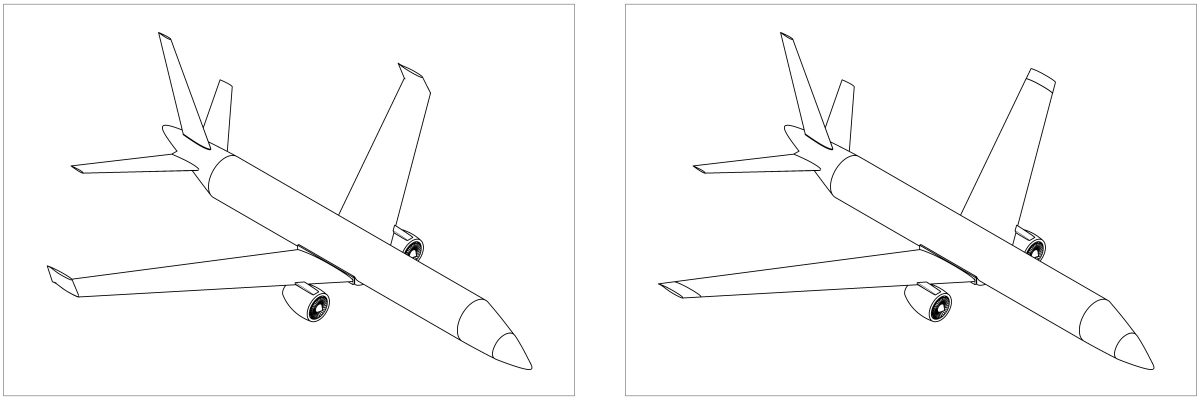


Figure 22: Isometric view of HR One. Unfolded wingtips (left) and unfolded wingtips (right).

Full Dimensioned Front, Left and Top Views

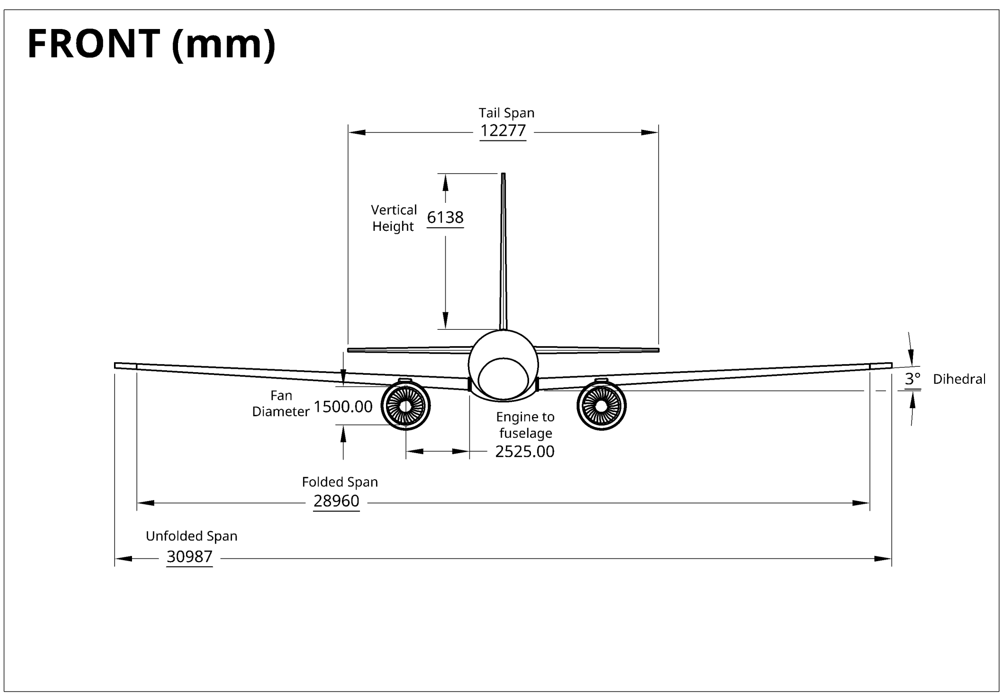


Figure 23: Fully dimensioned view of HR One from the front side.

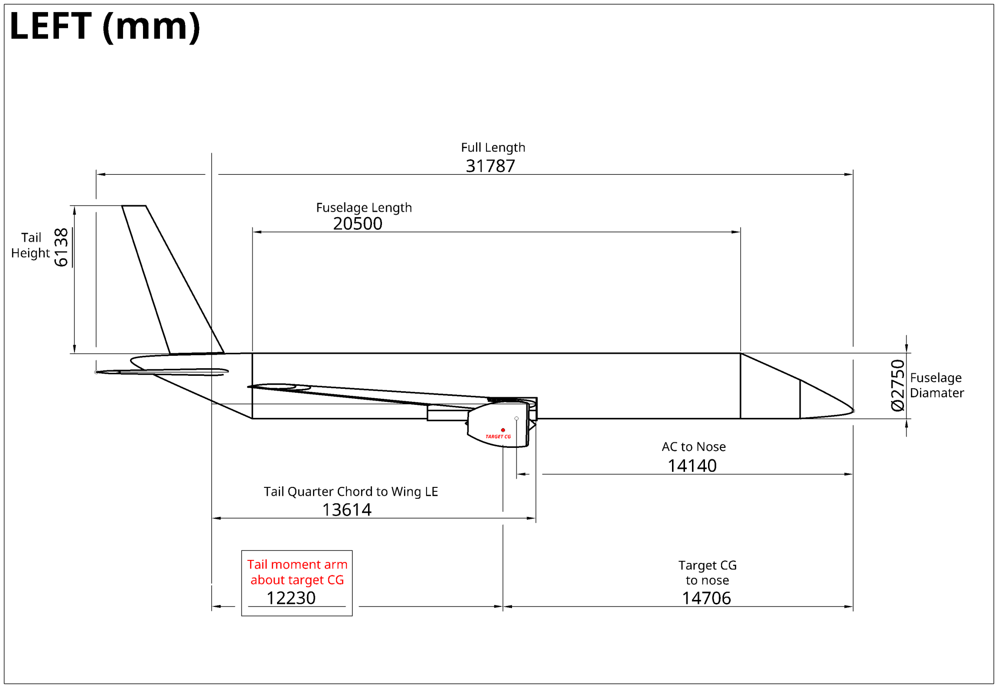


Figure 24: Fully dimensioned view of HR One from the left, including tail moment arm, ℓ_t.

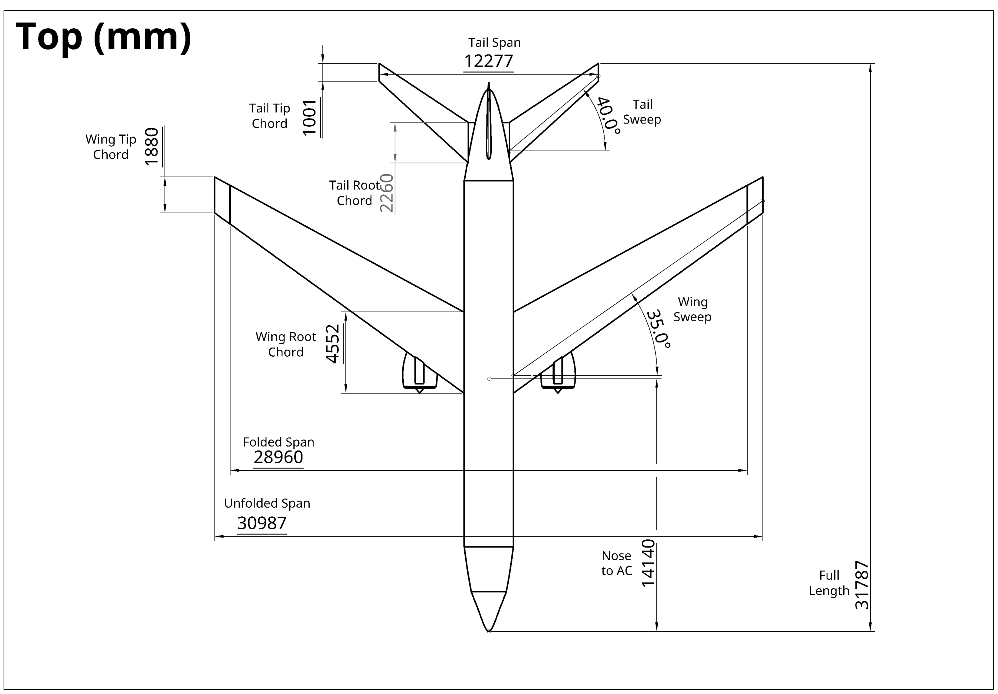


Figure 25: Fully dimensioned view of HR One from the top side.

4.10.2 Internal Diagrams, Centerline Diagram, and Internal Volume Requirements

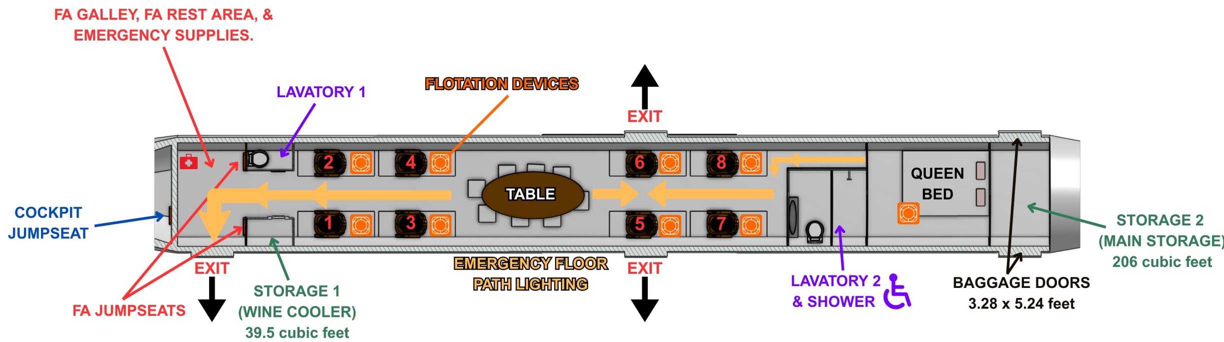


Figure 26: Cross-section showing layout of passenger accommodations (LOPA), including layout of baggage compartment and baggage doors

Notes:

HR One comfortably fits 8 individuals, with each lying fully flat to a length of 5.9 feet, including a width of 1.97 feet. The aisle has a width of 2.7 feet, with a minimum width of 2.2 feet beside the table (with table-seats stowed), and beside lavatory 2. Lavatory 2 is fully handicap accessible, including the shower. If desired, a foldout shower-seating rest is available for complementary installation. For reference, the shower has a width of 5.25 feet, a depth of 2.62 feet, and a minimum height of 6.01 feet (5.25 x 2.62 x 6.01 feet).

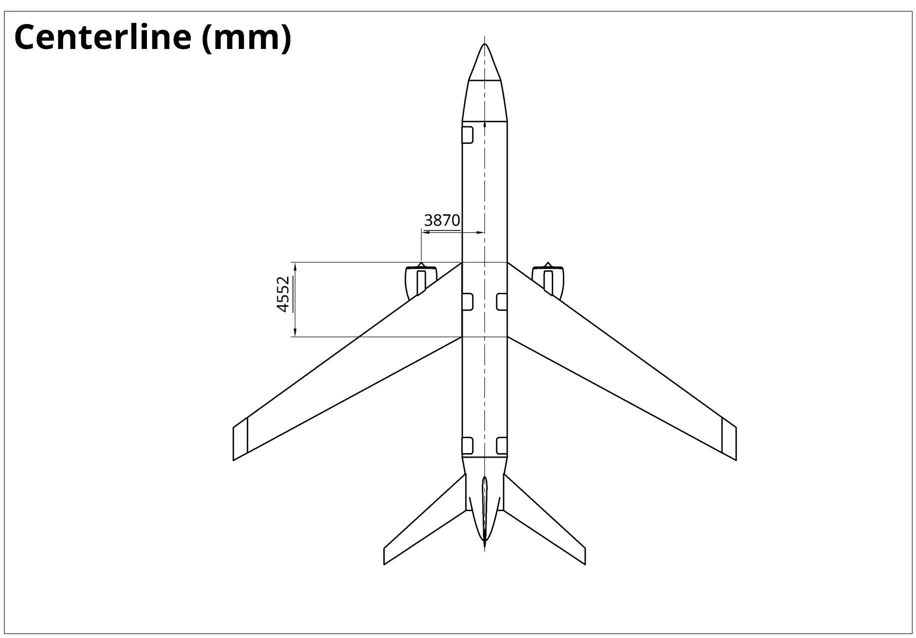


Figure 27: Fuselage centerline diagram of HR One.

4.11 Cost Estimate and Business Analysis

4.11.1 Non-Recurring Development Cost

The Non-Recurring Development Cost's main cost groups are engineering and design labor, FAA/EASA Certification, manufacturing/facilities, and flight test. Using the RAND cost estimation framework, data from the development of the Bombardier Global 7500 and Gulfstream G650, and the BLS Producer Price Index for the Aircraft Industry to scale money with industry inflation.

Table 11: Non-Recurring Development Cost Estimate (2025 USD)

Cost group	**Cost (B$)**	**Primary reference(s)**
Engineering & design labour	0.65	[12, 14, 13]
Certification (FAA + EASA)	0.35	[24, 16]
Tooling, jigs, facilities	0.40	[20, 14]
Flight-test fleet (4 aircraft + instr.)	0.30	[17, 18]
Contingency / program management (8 %)	0.14	[26]
Total NRC	**1.84**	—

The engineering and design block was derived from RAND's estimation of airframe cost then his assumption that airframe is 30% of total engineering and design labor. Certification costs were estimated from industry sources as listed. Tooling costs were calculated using RAND's cost relationship. The flight testing fleet costs were estimated based on the Gulfstream G650 and the Global 7500 testing fleets. Then a contingency overhead was added as a buffer.

This Non-Recurring Development Cost is within industry reason. Given the ultra-long range capabilities and new technologies being integrated into the HR One this is a competitive development cost.

4.11.2 Flyaway Cost

Table 12: Fly-away (recurring) cost per aircraft – 2025 USD

Weight group	**Mass [kg]**	**Cost [M$]**
Wing structure (CFRP)	3 906	3.14
Fuselage structure	2 458	1.98
Tail surfaces	579	0.46
Landing gear	145	0.06
Propulsion + hybrid kit (+25%)	3 614	30.00
Aircraft systems	1 735	0.84
Avionics / control	723	6.10
Manufacturing labor	–	14.90
Final assembly & QA	–	5.20
Contingency	–	7.50
Total fly away cost	–	**70.20**

This fly away cost breakdown was done by estimating the weight breakdown of the Empty Operating Weight, then using various cost relationships published in RAND, NASA engine cost relationships, and Roskam VI aircraft systems and control weight based cost estimates [20] [**?**]. Manufacturing costs, quality assurance, and contingency were all added to come to a total fly away cost of $70.2 Million. This flyaway cost is on par with the current industry standards for luxury business jets.

4.11.3 Direct Operating Cost

The main cost groups for DOC are Fuel, airframe and engine maintenance, crew, depreciation over time, insurance and general fees. These can be estimated using industry data, and the SFC.

Table 13: Direct operating cost per flight-hour (2024 USD)

Cost element	**USD / hr**	**Notes / basis**
Fuel (Jet-A, 4 150 lb hr^{-1} @ $0.85 lb^{-1})	3 530	Cruise SFC 0.610; 8 000 lbf ×2
Engine maintenance	800	MSP-Gold analogue
Airframe maintenance	300	Based on Global 7500 data
Crew salaries, benefits, training	250	Two pilots + cabin
Depreciation ($70 M over 20 yr, 500 h yr^{-1})	700	Straight-line, no salvage
Insurance, nav	180	landing fees
Typical large-cabin jet		
Total DOC / hr	**5 760**	

This direct operating cost is competitive due to the HR One's competitive SFC. The Global 7500 is reported to have about $7500 /hr direct operating cost.

4.11.4 Cost Saving Measures

To limit costs and keep this program cutting edge in the development process, digital twin technology will be utilized to decrease the learning curve and expedite production. The use of digital twins in the aerospace industry can reduce the iterations of prototypes needed which are very resource heavy parts of aircraft development as seen in the non-recurring cost figure. According to Blare Tech digital twins can reduce the number of failures in design by 50-90%. To further lower costs in development and certification, already certified subsystems such as landing gear and previously designed avionics are going to be used.

4.11.5 Cost Model Defense

The use of RAND's cost relationships and other industry online cost relationships instead of the Raymer DAPCA model is because the DAPCA model put the non-recurring cost with the flyaway cost in a way that breaking them apart would not make logical sense. The DAPCA model is also so conservative due to its basis on military aircraft that for general aviation, it can be more accurate if it is done then divided by 4, as stated by Raymer [**?**]. Due to this large variation in accuracy, the simpler more accessible cost relationships were used.

4.12 Life-cycle Emissions Analysis

4.12.1 Production Emissions

The production emissions of the HR One can be estimated by multiplying the empty weight of the aircraft with the emission factors of carbon dioxide, methane, and nitrous oxide[6]. These are then converted to their carbon dioxide equivalent to standardize units and understand the warming effects of all gasses emitted on the same scale.

Table 14: Estimated Aircraft Manufacturing Emissions by Gas

Gas	**Emission Factor (kg gas/kg aircraft)**	**Total Emissions (kg)**	**CO_2e Equivalent (kg)**
CO_2	200	2,632,000	2,632,000
CH_4	0.005	65.8	1,842
N_2O	0.002	26.32	6,977
Total	—	—	**2,640,819**

The production emissions for the HR One are in line with the estimation production emissions of recent business jet campaigns.

4.12.2 In-Service Emissions

The in-service emissions of each jet is given by:

$$\text{Lifetime Emissions (kg)} = SFC \times T_{SL} \times \Pi \times AFH \times L \times E \tag{58}$$

where SFC is the specific fuel consumption, T_{SL} is the thrust of both engines, Π is the throttle setting at cruise, AFH is the average annual flight hours, L is the operational

lifespan, and E is the emission factor.
Standard Jet-A fuel has a CO_2 emission factor of 3.16 kg $CO_2/kg-fuel$, and a N_2O emission factor of 0.000002 kg $N_2O/kf-fuel$[6]. Assuming an average operational lifespan of 27 years and an average annual flight time of 450 hours, both based on business jet trends [7].

$$CO_2 \text{ Emissions} = 0.610 \times 70\text{kN} \times 1 \times 450 \times 27 \times 3.16 = 1.64 \text{ M kg } CO_2 \tag{59}$$

$$N_2O \text{ Emissions} = 0.610 \times 70\text{kN} \times 1 \times 450 \times 27 \times 0.000002 = 1.04 \text{ kg } N_2O \tag{60}$$

Table 15: Estimated Lifetime Emissions of HR One

Emission Type	**Emissions (kg)**
Carbon Dioxide (CO_2)	1.64×10^6
Nitrous Oxide (N_2O)	1.04
N_2O in CO_2-equiv (GWP 273)	283
Total CO_2-Equivalent	$\mathbf{1.64 \times 10^6}$

The lifetime emissions of the HR One with these assumptions is better than the average business jet of a similar size and thrust because it is directly dependent on the specific fuel consumption. The HyTEC's fuel consumption reduction gives the HR One a competitive advantage by lowering life time emissions and aligning with aviation sustainability initiatives.

5 Conclusion

The HR One was designed to meet the challenge of building an ultra-long-range business jet that can cruise at Mach 0.92 and fly over 8,000 nautical miles nonstop. This was achieved through aerodynamic optimization, careful weight and mission analysis, and thoughtful systems integration. The aircraft meets all mission requirements—from high-altitude takeoffs in Aspen to long transoceanic legs—while staying within structural, regulatory, and environmental limits.

The Weight Estimation & Mission Performance work confirmed that the wing and propulsion package support a realistic MTOW, balancing empty weight, fuel, and payload across all three missions with margin to spare. The aerodynamic team selected a high-aspect-ratio, supercritical wing that delivers a lift-to-drag ratio of 18 at cruise, with drag-buildup studies validating performance across flight conditions. Engine selection focused on meeting regulations and requirements for thrust and fuel burn, ensuring power margins throughout the mission while minimizing fuel consumption. The cabin layout was designed to comfortably fit system volumes and integrate wing tanks without sacrificing passenger space or comfort. Material and emissions studies showed that using advanced composites and efficient engines could cut lifecycle CO_2 emissions by 10–15% compared to current competitors, all while staying cost-competitive.

Overall, HR One fills a unique spot in the market: fast, efficient, and capable of flying farther than most other business jets, without compromising on luxury or environmental performance. The design meets FAA Part 25 requirements and is flexible enough for a wide range of missions—from transcontinental business trips to high-elevation resort runs. Going forward, the team will focus on structural sizing, fatigue analysis, flight-control development, and test planning to move the aircraft closer to flight readiness.

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Results from the BLADE (BreakthroughLaminar Aircraft Demonstrator inEurope) Flight-Test Campaign

CleanSky2 HLFC Wing-Box: Design, Manufacturing and Ground Testing

Certification Flight Testing of Hybrid-Laminar-Flow-Control on the Boeing 787 Vertical Tail.

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A Matlab Appendix

A.1 Section 3.3

```
    #!/usr/bin/env python3
# -*- coding: utf-8 -*-
"""
Airplane sizing script (design-project version)

Author: diegobermejo
Created: 24-Apr-2025
"""

import math


# ----------------------------------------------------------------------
# 1. Mission-segment inputs  (all numbers are the SAME as your draft)
# ----------------------------------------------------------------------
R_NM = 644                           # range, nmi
R_FT = R_NM * 6076.12                 # convert to feet

TSFC_HR = 0.642 * 0.95                        # lb / lb / hr
C_CRUISE = TSFC_HR / 3600             # 1/s

V_FT_S  = 891           # cruise true air-speed, ft/s at mach 0.92
L_D     = 20                  # lift-to-drag ratio
ETA     = 0.866
E = 1800                    # loiter time 3600 = 1   hr ; 1800 = 30 min

# Discrete segment fractions (yours)
W1_W0 = 0.97                          # warm-up + take-off
W2_W1 = 0.985                         # climb
W7_W6 = 0.995                         # landing
loiter = math.exp((-E*C_CRUISE)/(L_D))             # 0.948   used previously (mistake)

# Cruise segment driven by range equation
W3_W2 = math.exp(-(R_FT * C_CRUISE) / (V_FT_S * L_D * ETA))

# Product to start of landing
W7_W0 = (W1_W0 *
         W2_W1 *
         W3_W2 *
         loiter *
         W7_W6)
```

```
# Fuel fraction (includes 3 % trapped fuel and reserve multiplier)
Wf_W0 = 1.06 * (1.0 - W7_W0)

# ----------------------------------------------------------------------
# 2. Empty-weight model (kept identical constants, just isolated)
# ----------------------------------------------------------------------
GAMMA   = 0.95      # composites weight reduction
k_e     = 1.02      # baseline constant (your value)
a_exp   = 0.06      # exponent  (your value)


def We_W0(W0: float) -> float:
    """Empty-weight fraction for a given guess of W0 (lb)."""
    return GAMMA * k_e * W0 ** (-a_exp)

# ----------------------------------------------------------------------
# 3. Newton?Raphson solver for take-off weight W0
# ----------------------------------------------------------------------
PAYLOAD = 2715    # lb  (your number)

def f(W0: float) -> float:
    """Root function  f(W0) = 0  ->  sizing equation."""
    return (1.0 - Wf_W0 - We_W0(W0)) * W0 - PAYLOAD

def df(W0: float) -> float:
    """Exact derivative of f(W0)."""
    return (1.0 - Wf_W0 - We_W0(W0)) + \
           W0 * (-GAMMA * k_e * (-a_exp) * W0 ** (-a_exp - 1.0))

def newton_W0(initial: float = 1.0e5,
              tol: float = 1.0e-6,
              max_iter: int = 50) -> float:
    """Newton?Raphson root finder for W0."""
    W0 = initial
    for i in range(1, max_iter + 1):
        W1 = W0 - f(W0) / df(W0)
        if abs(W1 - W0) < tol:
            print(f"Converged after {i:2d} iterations ?  W0 = {W1:,.1f} lb")
            return W1
        W0 = W1
    raise RuntimeError("Newton solver failed to converge.")

# ----------------------------------------------------------------------
# 4. Run the solve and print a quick breakdown
```

```
# ----------------------------------------------------------------------
if __name__ == "__main__":
    W0 = newton_W0(initial=100_000)          # you can change the seed

    We_frac = We_W0(W0)
    We      = We_frac * W0
    Wf      = Wf_W0 * W0
    Wp_w0 =   PAYLOAD / W0
    print(f"\n--- Mission summary with original parameters ---")
    print(f"W7/W0  = {W7_W0:.4f}")
    print(f"Wf/W0  = {Wf_W0:.4f}   ?  Wf = {Wf:,.0f} lb")
    print(f"We/W0  = {We_frac:.4f} ?  We = {We:,.0f} lb")
    print(f"Payload           = {PAYLOAD:,.0f} lb")
    print(f"Take-off weight   = {W0:,.0f} lb")
    print(f"Payload Fraction  =   {Wp_w0:.4f} lb")
    print(f"Fuel Fraction  =   {Wf_W0:.4f} lb")

    print(TSFC_HR)
```

A.2 Section 4.1.2 (1)

```
    %% FLIGHT & AIRCRAFT PARAMETERS
h1 = 0;       % Altitude #1 (m) -> Sea Level
h2 = 13716;    % Altitude #2 (m) -> 9000 m (45000ft -> m)
[Tsl, ~, ~, rhosl] = atmosisa(0);  % Density and T at sea level
[T,~,a,rho] = atmosisa(h2); % Density, speed of sound, and T at cruise alt
g = 9.81;
e = 0.9;


%traded variables
v = linspace(40, 350, 20); % Velocity range (m/s)
AR_vals = linspace(5,15,20);

[V,AR] = meshgrid(v,AR_vals);

% weights, all given by diego
WTO = 38745.86*g;
W_payload = 1231.5 * g; % Max pay weight, Napa economic mission
fuel_fraction = 0.4780;
W_empty = 18995.9949*g;
W_fuel = fuel_fraction .* WTO;
% WTO calc by adding weight components, consistent w diego.
WTO = W_empty + W_payload + W_fuel;
payload_fraction = (W_payload + WTO) / WTO;                % Payload fraction
% Weight at end of flight
W1         = WTO - W_fuel;

% Aerodynamic aircraft parameters
CD0        = 0.014;     % Parasitic Drag Coefficient
e          = 0.9;      % Oswald Efficiency Factor
C_L = sqrt((pi*e*AR*CD0)/3); % eq for CL at best range (JET)
S          = WTO./(0.5.*C_L.*rho.*V.^2);  % Wing Area (m^2)
CD = CD0 + C_L.^2 ./ (pi * e .* AR);
b = sqrt(AR.*S);
SFC        = (0.642/3600)*0.95; % Specific Fuel Consumption of selected engine


%% compute compressible drag effects
sweep  = deg2rad(35);
tc     = 0.10*0.6; % *0.6 to account for supercritical airfoil
Minf       = V ./ a; % local mach

% PREALLOCATE
```

```
Range   = nan(size(V));
S       = nan(size(V));
b       = nan(size(V));

S_min = 99.2;   % the 1-D trade result you want to enforce
b_min = 30.95;
% MAIN LOOP: compute CL, S, CDi, CDc, CD_total, Range
for i = 1:numel(AR_vals)
  for j = 1:numel(v)
    AR_ij = AR_vals(i);
    V_ij  = v(j);

    % lift coefficient for best-range (JET)
    CL_ij = sqrt(pi * e * AR_ij * CD0);

    % enforce b_min and S_min
    S_ij  = WTO / (0.5 * rho * V_ij^2 * CL_ij); % wing area from CL lifting WTO (max) at
    S_ij = max(S_ij, S_min); % take max of calculated S and S_min to ensure wing area is
    b_ij = sqrt(AR_ij * S_ij);

    % overwrite CL for best range/lowest drag (max L/D) with CL actual (bc of forced S v
    % need to stay consistent.)
    CL_ij = WTO/(0.5 * rho * V_ij^2 * S_ij); % will be lower than CL for best range

    % induced drag
    CDi = CL_ij^2 / (pi * e * AR_ij);

    % compressibility bump
    Mcc_0     = 0.87 - 0.175*CL_ij - 0.83*tc;
    m_exp     = 0.83 - 0.583*CL_ij + 0.111*CL_ij^2;
    Mcc_sw    = Mcc_0 / (cos(sweep)^m_exp);
    x_ratio   = (V_ij/a) / Mcc_sw;
    deltaCDc  = (3.97e-9*exp(12.7*x_ratio) + 1e-40*exp(81*x_ratio)) ...
                * (cos(sweep)^3);

    % total drag coefficient
    CD_total = CD0 + CDi + deltaCDc;
    Em = CL_ij/CD_total; % L/D calculated at each step

    % Breguet range
    %X(i,j) = (V_ij / SFC) * (CL_ij / CD_total) * log(WTO / W1);
    k = 1/(pi*e*AR_ij);
    q = .5*(V_ij^2)*rho;
    WD = sqrt(CD0/k)*q*S_ij;
    W0star = WTO/WD;
```

```
    X(i,j) = ((2*V_ij*Em)/SFC) * atan((W0star.*fuel_fraction) / (1 + (W0star^2)*(1-fuel_
    % store geometry
    S(i,j) = S_ij;
    b(i,j) = b_ij;
  end
end

%%
% Breguet range equation (jet)
%X = V ./ SFC .* (C_L ./ CD) .* log(WTO ./ W1);

% Given cruise conditions: AR = 9.41, V = 271 m/s
AR_query = 9.68;
V_query  = 271;

% interpolate range, span, and wing area
range_interp = interp2(V, AR, X, V_query, AR_query);
b_query      = interp2(V, AR, b, V_query, AR_query);
S_query      = interp2(V, AR, S, V_query, AR_query);

fprintf('\nRange at AR = %.2f, V = %.1f m/s: \nX = %.2f km\nX = %.2f nm\n', AR_query, V_
fprintf('Span  at AR = %.2f, V = %.1f m/s: \nb = %.3f m\n',  AR_query, V_query, b_query)
fprintf('Wing area at AR = %.2f, V = %.1f m/s: \nS = %.3f m²\n', AR_query, V_query, S_qu

figure;
contourf(AR, V, X, 50, 'LineColor', 'none');
colorbar;
xlabel('Aspect Ratio (AR) [-]');
ylabel('Velocity [m/s]');
title('Flight Range (X) [m] vs AR and Velocity');
% Plot red marker at the queried point
hold on;
plot(AR_query, V_query, 'ro', 'MarkerSize', 8, 'LineWidth', 2);
```

A.3 Section 4.1.2 (2)

```
clear; clc; close all;
% payload weight vs range
% fuel decreases as payload increases (MTOW stays constant) -> more
% payload shrinks range

%% FLIGHT & AIRCRAFT PARAMETERS
h1 = 0;       % Altitude #1 (m) -> Sea Level
h2 = 13716;    % Altitude #2 (m) -> 9000 m (45000ft -> m)
g = 9.81;

% from trades
AR = 9.68; %change
b = 30.987;
S = b^2/AR

CD0         = 0.014;      % Parasitic Drag Coefficient
e           = 0.9;       % Oswald Efficiency Factor

V = 271; % V cruise, M0.92

[~, ~, ~, rhosl] = atmosisa(0);  % Density at sea level
[~,a,~,rho] = atmosisa(h2);
%% Traded variables
%range = linspace(0, 14816*1000, 20); % range up to 8000nm (m)
W_payload = linspace(0,1231.5*g); % [N], up to max Wpay. passenger mission 1 = 961.7792k

%[RANGE,WPAY] = meshgrid(range, W_payload);

SFC         = (0.642/3600)*0.95; % Specific Fuel Consumption of selected engine

range_vals = zeros(size(W_payload));    % preallocate to be same size

%% CL
%C_L  = WTO /(0.5*rho*Vcr^2*S0)
CL = sqrt((pi*e*AR*CD0)/3) % eq for CL at best range [jet]. selectd airfoil is already d
%C_L = 0.4; % CL cruise

CD = CD0 + CL.^2 ./ (pi * e .* AR);


%% Loop through payload values and compute range
for i = 1:length(W_payload)
```

```
    % Fuel weight
    %fuel_fraction = 0.4780; % given at max payload
    WTO = 38745.86*g; % required to be constant
    W_empty = 18995.9949*g;
    W_fuel = WTO - W_empty - W_payload(i); % fuel calculated based on room from pay;oad
    fuel_fraction = W_fuel / WTO; % calculate fuel fraction just for fun
    %W_fuel = fuel_fraction .* WTO;
    %W_fuel = WTO - W_empty - W_payload(i);  % Remaining weight = fuel
    %WTO = W_empty + W_payload(i) + W_fuel;

    if W_fuel <= 0
        range_vals(i) = NaN;  % infeasible case (too much payload)
        continue;
    end
    % W1 = WTO - W_fuel;  % Final weight after fuel burn
    % bregut range eq for FAR requirements (const Vcr, hcr)

    % Induced drag
    CDi = CL^2 / (pi*e*AR);
    sweep = deg2rad(35);
    tc = 0.1*0.6;
    Minf = V/a;

    % Compressibility penalty
    Mcc_0      = 0.87 - 0.175*CL - 0.83*tc;
    m          = 0.83 - 0.583*CL + 0.111*CL^2;
    Mcc_sweep  = Mcc_0 / (cos(sweep)^m);
    x          = Minf / Mcc_sweep;
    deltaCDc   = (3.97e-9*exp(12.7*x) + 1e-40*exp(81*x)) * cos(sweep)^3;

    % Total CD
    CD_total = CD0 + CDi + deltaCDc;

    Em = CL/CD_total; % L/D calculated at each step
    k = 1/(pi*e*AR);
    q = .5*(V^2)*rho;
    WD = sqrt(CD0/k)*q*S;
    W0star = WTO/WD;
    range_vals(i) = ((2*V*Em)/SFC) * atan((W0star*fuel_fraction) / (1 + (W0star^2)*(1-fu

    %range_vals(i) = V / SFC * (C_L / CD) * log(WTO / W1); % brugeut jet range
end

%% (after you compute range_vals and before plotting)
% find min/max range indices
```

```
[range_min, idx_min] = min(range_vals);
[range_max, idx_max] = max(range_vals);

% corresponding payloads in kg
payload_min = W_payload(idx_min)/g;
payload_max = W_payload(idx_max)/g;

% print to command window
fprintf('Minimum range = %.2f km or = %.2f nm at payload = %.2f kg\n', range_min/1000, r
fprintf('Maximum range = %.2f km or = %.2f nm at payload = %.2f kg\n', range_max/1000, r

payload_kg = [572.7, 961.8]
for i = 1:numel(payload_kg)
    Rm = range_vals(i)/1000;
    RN = range_vals(i)*0.000539957;
    fprintf('   %4.0f         %7.2f      %7.2f\n', payload_kg(i), Rm, RN);
end

%% Plotting
figure;
plot(W_payload / g, range_vals / 1000, 'r', 'LineWidth', 3); % kg vs km
hold on
% mark and label the extrema
plot(payload_min, range_min/1000, 'bs', 'MarkerSize', 10, 'LineWidth', 2, 'DisplayName',
hold on
plot(payload_max, range_max/1000, 'gs', 'MarkerSize', 10, 'LineWidth', 2, 'DisplayName',

xlabel('Payload Weight (kg)');
ylabel('Range (km)');
title('Range vs Payload Weight');
legend('Range vs Payload Weight','Min. Range', 'Max. Range');
ax = gca;                   % Get current axes
ax.XAxis.Exponent = 0;      % Disable sci notation on x-axis
ax.YAxis.Exponent = 0;
grid on;
```

A.4 Section 4.2 (1)

```
%% FLIGHT & AIRCRAFT PARAMETERS
h1 = 0;       % Altitude #1 (m) -> Sea Level
h2 = 13716;    % Altitude #2 (m) -> 9000 m (45000ft -> m)
[Tsl, ~, ~, rhosl] = atmosisa(0);  % Density and T at sea level
[T,~,a,rho] = atmosisa(h2); % Density, speed of sound, and T at cruise alt
g = 9.81;
e = 0.9;


%traded variables
v = linspace(40, 350, 20); % Velocity range (m/s)
AR_vals = linspace(5,15,20);

[V,AR] = meshgrid(v,AR_vals);

% weights, all given by diego
WTO = 38745.86*g;
W_payload = 1231.5 * g; % Max pay weight, Napa economic mission
fuel_fraction = 0.4780;
W_empty = 18995.9949*g;
W_fuel = fuel_fraction .* WTO;
% WTO calc by adding weight components, consistent w diego.
WTO = W_empty + W_payload + W_fuel;
payload_fraction = (W_payload + WTO) / WTO;                % Payload fraction
% Weight at end of flight
W1         = WTO - W_fuel;

% Aerodynamic aircraft parameters
CD0        = 0.014;     % Parasitic Drag Coefficient
e          = 0.9;      % Oswald Efficiency Factor
C_L = sqrt((pi*e*AR*CD0)/3); % eq for CL at best range (JET)
S          = WTO./(0.5.*C_L.*rho.*V.^2);  % Wing Area (m^2)
CD = CD0 + C_L.^2 ./ (pi * e .* AR);
b = sqrt(AR.*S);
SFC        = (0.642/3600)*0.95; % Specific Fuel Consumption of selected engine


%% compute compressible drag effects
sweep  = deg2rad(35);
tc     = 0.10*0.6; % *0.6 to account for supercritical airfoil
Minf       = V ./ a; % local mach

% PREALLOCATE
```

```
Range   = nan(size(V));
S       = nan(size(V));
b       = nan(size(V));

S_min = 99.2;   % the 1-D trade result you want to enforce
b_min = 30.95;
% MAIN LOOP: compute CL, S, CDi, CDc, CD_total, Range
for i = 1:numel(AR_vals)
  for j = 1:numel(v)
    AR_ij = AR_vals(i);
    V_ij  = v(j);

    % lift coefficient for best-range (JET)
    CL_ij = sqrt(pi * e * AR_ij * CD0);

    % enforce b_min and S_min
    S_ij  = WTO / (0.5 * rho * V_ij^2 * CL_ij); % wing area from CL lifting WTO (max) at
    S_ij = max(S_ij, S_min); % take max of calculated S and S_min to ensure wing area is
    b_ij = sqrt(AR_ij * S_ij);

    % overwrite CL for best range/lowest drag (max L/D) with CL actual (bc of forced S v
    % need to stay consistent.)
    CL_ij = WTO/(0.5 * rho * V_ij^2 * S_ij); % will be lower than CL for best range

    % induced drag
    CDi = CL_ij^2 / (pi * e * AR_ij);

    % compressibility bump
    Mcc_0     = 0.87 - 0.175*CL_ij - 0.83*tc;
    m_exp     = 0.83 - 0.583*CL_ij + 0.111*CL_ij^2;
    Mcc_sw    = Mcc_0 / (cos(sweep)^m_exp);
    x_ratio   = (V_ij/a) / Mcc_sw;
    deltaCDc  = (3.97e-9*exp(12.7*x_ratio) + 1e-40*exp(81*x_ratio)) ...
                * (cos(sweep)^3);

    % total drag coefficient
    CD_total = CD0 + CDi + deltaCDc;
    Em = CL_ij/CD_total; % L/D calculated at each step

    % Breguet range
    %X(i,j) = (V_ij / SFC) * (CL_ij / CD_total) * log(WTO / W1);
    k = 1/(pi*e*AR_ij);
    q = .5*(V_ij^2)*rho;
    WD = sqrt(CD0/k)*q*S_ij;
    W0star = WTO/WD;
```

```
    X(i,j) = ((2*V_ij*Em)/SFC) * atan((W0star.*fuel_fraction) / (1 + (W0star^2)*(1-fuel_
    % store geometry
    S(i,j) = S_ij;
    b(i,j) = b_ij;
  end
end

%%
% Breguet range equation (jet)
%X = V ./ SFC .* (C_L ./ CD) .* log(WTO ./ W1);

% Given cruise conditions: AR = 9.41, V = 271 m/s
AR_query = 9.68;
V_query  = 271;

% interpolate range, span, and wing area
range_interp = interp2(V, AR, X, V_query, AR_query);
b_query      = interp2(V, AR, b, V_query, AR_query);
S_query      = interp2(V, AR, S, V_query, AR_query);

fprintf('\nRange at AR = %.2f, V = %.1f m/s: \nX = %.2f km\nX = %.2f nm\n', AR_query, V_
fprintf('Span  at AR = %.2f, V = %.1f m/s: \nb = %.3f m\n',  AR_query, V_query, b_query)
fprintf('Wing area at AR = %.2f, V = %.1f m/s: \nS = %.3f m²\n', AR_query, V_query, S_qu

figure;
contourf(AR, V, X, 50, 'LineColor', 'none');
colorbar;
xlabel('Aspect Ratio (AR) [-]');
ylabel('Velocity [m/s]');
title('Flight Range (X) [m] vs AR and Velocity');
% Plot red marker at the queried point
hold on;
plot(AR_query, V_query, 'ro', 'MarkerSize', 8, 'LineWidth', 2);
```

A.5 Section 4.2 (2)

```
clear; clc; close all;
% payload weight vs range
% fuel decreases as payload increases (MTOW stays constant) -> more
% payload shrinks range

%% FLIGHT & AIRCRAFT PARAMETERS
h1 = 0;      % Altitude #1 (m) -> Sea Level
h2 = 13716;   % Altitude #2 (m) -> 9000 m (45000ft -> m)
g = 9.81;

% from trades
AR = 9.68; %change
b = 30.987;
S = b^2/AR

CD0        = 0.014;     % Parasitic Drag Coefficient
e          = 0.9;      % Oswald Efficiency Factor

V = 271; % V cruise, M0.92

[~, ~, ~, rhosl] = atmosisa(0);  % Density at sea level
[~,a,~,rho] = atmosisa(h2);
%% Traded variables
%range = linspace(0, 14816*1000, 20); % range up to 8000nm (m)
W_payload = linspace(0,1231.5*g); % [N], up to max Wpay. passenger mission 1 = 961.7792k

%[RANGE,WPAY] = meshgrid(range, W_payload);

SFC        = (0.642/3600)*0.95; % Specific Fuel Consumption of selected engine

range_vals = zeros(size(W_payload));    % preallocate to be same size

%% CL
%C_L  = WTO /(0.5*rho*Vcr^2*S0)
CL = sqrt((pi*e*AR*CD0)/3) % eq for CL at best range [jet]. selectd airfoil is already d
%C_L = 0.4; % CL cruise

CD = CD0 + CL.^2 ./ (pi * e .* AR);


%% Loop through payload values and compute range
for i = 1:length(W_payload)
```

```
    % Fuel weight
    %fuel_fraction = 0.4780; % given at max payload
    WTO = 38745.86*g; % required to be constant
    W_empty = 18995.9949*g;
    W_fuel = WTO - W_empty - W_payload(i); % fuel calculated based on room from pay;oad
    fuel_fraction = W_fuel / WTO; % calculate fuel fraction just for fun
    %W_fuel = fuel_fraction .* WTO;
    %W_fuel = WTO - W_empty - W_payload(i);  % Remaining weight = fuel
    %WTO = W_empty + W_payload(i) + W_fuel;

    if W_fuel <= 0
        range_vals(i) = NaN;  % infeasible case (too much payload)
        continue;
    end
    % W1 = WTO - W_fuel;  % Final weight after fuel burn
    % bregut range eq for FAR requirements (const Vcr, hcr)

    % Induced drag
    CDi = CL^2 / (pi*e*AR);
    sweep = deg2rad(35);
    tc = 0.1*0.6;
    Minf = V/a;

    % Compressibility penalty
    Mcc_0      = 0.87 - 0.175*CL - 0.83*tc;
    m          = 0.83 - 0.583*CL + 0.111*CL^2;
    Mcc_sweep  = Mcc_0 / (cos(sweep)^m);
    x          = Minf / Mcc_sweep;
    deltaCDc   = (3.97e-9*exp(12.7*x) + 1e-40*exp(81*x)) * cos(sweep)^3;

    % Total CD
    CD_total = CD0 + CDi + deltaCDc;

    Em = CL/CD_total; % L/D calculated at each step
    k = 1/(pi*e*AR);
    q = .5*(V^2)*rho;
    WD = sqrt(CD0/k)*q*S;
    W0star = WTO/WD;
    range_vals(i) = ((2*V*Em)/SFC) * atan((W0star*fuel_fraction) / (1 + (W0star^2)*(1-fu

    %range_vals(i) = V / SFC * (C_L / CD) * log(WTO / W1); % brugeut jet range
end

%% (after you compute range_vals and before plotting)
% find min/max range indices
```

```
[range_min, idx_min] = min(range_vals);
[range_max, idx_max] = max(range_vals);

% corresponding payloads in kg
payload_min = W_payload(idx_min)/g;
payload_max = W_payload(idx_max)/g;

% print to command window
fprintf('Minimum range = %.2f km or = %.2f nm at payload = %.2f kg\n', range_min/1000, r
fprintf('Maximum range = %.2f km or = %.2f nm at payload = %.2f kg\n', range_max/1000, r

payload_kg = [572.7, 961.8]
for i = 1:numel(payload_kg)
    Rm = range_vals(i)/1000;
    RN = range_vals(i)*0.000539957;
    fprintf('   %4.0f         %7.2f      %7.2f\n', payload_kg(i), Rm, RN);
end

%% Plotting
figure;
plot(W_payload / g, range_vals / 1000, 'r', 'LineWidth', 3); % kg vs km
hold on
% mark and label the extrema
plot(payload_min, range_min/1000, 'bs', 'MarkerSize', 10, 'LineWidth', 2, 'DisplayName',
hold on
plot(payload_max, range_max/1000, 'gs', 'MarkerSize', 10, 'LineWidth', 2, 'DisplayName',

xlabel('Payload Weight (kg)');
ylabel('Range (km)');
title('Range vs Payload Weight');
legend('Range vs Payload Weight','Min. Range', 'Max. Range');
ax = gca;                  % Get current axes
ax.XAxis.Exponent = 0;     % Disable sci notation on x-axis
ax.YAxis.Exponent = 0;
grid on;
```

A.6 Section 4.2.2

```
clear; clc; close all
[tCD0, CD0] = CD0build();
disp(tCD0); fprintf('\nTotal C_D0 = %.4f\n\n',CD0);

%  Flight parameters
h     = 13106.4;                   % cruise altitude (m)  43000ft
v     = linspace( 50, 300, 200);   % velocity sweep (m/s)
WTO   = 37019.73194*9.81;                  % weight (N)
CLmax = 1.40;
e     = 0.90;
b     = 30.987;                     % m
AR    = 9.68;
S     = b^2/AR;                    % m²
tc    = 0.10*0.6;
sweep = 35;                        % deg (_{c/4})
k     = 1/(pi*e*AR);
TSL   = 70000;                     % installed SLS thrust (N)


[Vstall, Dpara, Dind, Dinc, Dcomp, TA, CDc] = ...
        Drag(h,v,WTO,S,CD0,k,TSL,sweep,tc,CLmax);
% Drag vs velocity ---------------------------------------------------- %
figure('Name','Drag vs V');
plot(v,Dinc ,'b','LineWidth',2);  hold on; grid on
plot(v,Dcomp,'k--','LineWidth',2);
plot(v,Dpara,'r','LineWidth',2);
plot(v,Dind,'LineWidth',2);
legend('Incompressible','Compressible','Parasite', 'Induced', 'Location','best');
xlabel('Velocity  V  (m/s)'); ylabel('Drag  D  (N)');
title('HROne Drag vs Velocity');
ylim([0,5e4])

figure('Name','Drag vs Mach');
[T, ~, ~, ~] = atmosisa(h);
a = sqrt(1.4*287*T);% speed of sound at cruise altitude
M = v ./ a;                        % convert velocity to Mach number

plot(M, Dinc ,'b','LineWidth',2);  hold on; grid on
plot(M, Dcomp,'k--','LineWidth',2);
plot(M, Dpara,'r','LineWidth',2);
plot(M, Dind,'LineWidth',2);
legend('Incompressible','Compressible','Parasite', 'Induced', 'Location','best');
xlabel('Mach Number  M'); ylabel('Drag  D  (N)');
```

```
title('HROne Drag vs Mach Number');
ylim([0, 5e4])
xlim([0.2,1])



function [tbl,CD0tot] = CD0build()
% Zero-lift parasite drag build-up (hand-out \2Cf" simplification)
% ------------------------------------------------------------------------
Sref = 99.2;                   % m²  reference wing area
b    = 30.987;                 % m   span
cMAC = Sref/b;                 % m   mean aero. chord
lf   = 25.10125806;   df = 2.75;     % m   fuselage length & dia
t_c  = 0.10;                   % thickness-to-chord
rho  = 0.348;  Vcr = 270;      % ISA-43000ft
mu   = 1.46e-5;                % kg/(m·s)

%  Wetted areas
Sw_w  = (2*b*cMAC*(1-0.25*t_c))*0.85;     % wing     (Raymer Eq-12.2a)
Sw_ht = 0.2*b*0.5*cMAC;            % horiz-tail
Sw_vt = 0.5*Sw_ht;                 % vert-tail
Sw_f  = pi*df/2*(df+lf);           % fuselage  (prolate ellipsoid)

%  Skin-friction coefficients
Re_w  = rho*Vcr*cMAC/mu;
Re_f  = rho*Vcr*lf   /mu;
Cf_w = 0.1*(1.328/sqrt(Re_w)) + (1-0.1)*0.074/Re_w^0.2;
Cf_f  = 0.074/Re_f^0.2;


%     A_D = (C_D0)·S_wet  →  C_D0 = 2·C_f          (hand-out)
A = [ 2*Cf_w       *Sw_w ;      % Wing
      2*Cf_w       *Sw_ht;      % H-tail
      2*Cf_w       *Sw_vt;      % V-tail
      Cf_f         *Sw_f ;      % Fuselage uses full form-factor in note
      0.020 ];                  % Aux/excrescences

CD = A / Sref;


tbl = table(["Wing";"H-Tail";"V-Tail";"Fuselage";"Aux"],CD,...
            'VariableNames',{'Component','CD0_ind'});

CD0tot = sum(CD);
end
```

```matlab
figure('Name','C_{D0} build-up','Color','w');

barH = bar(categorical(tCD0.Component), tCD0.CD0_ind, ...
    'FaceColor',[0.16 0.50 0.78], ...
    'EdgeColor','none');

xlabel('Aircraft component','FontSize',11);
ylabel('Component C_{D0}','FontSize',11);
title({'HROne Parasite-Drag Breakdown'; ...
     sprintf('\\Sigma C_{D_{0}} = %.4f',CD0)}, ...
    'FontSize',12,'FontWeight','normal');
grid on;
ax = gca;
ax.YAxis.Exponent = 0;
ax.YRuler.TickLabelFormat = '%.4f';

x = barH.XEndPoints;

vals = barH.YData;

for ii = 1:numel(vals)
    text(x(ii), vals(ii), ...
        sprintf('%.4f', vals(ii)), ...
        'HorizontalAlignment', 'center', ...
        'VerticalAlignment', 'bottom', ...
        'FontSize', 9, 'FontWeight', 'bold');
end



function [Vstall, DP,Di, DI, DC, TA, CDc] = ...
          Drag(h,v,W,S,CD0,k,TSL,sweep,tc,CLmax)
% Incompressible & compressible drag vs velocity.
[~,~,~,rhosl] = atmosisa(0); [t,~,~,rho] = atmosisa(h);
q   = 0.5*rho*v.^2;  a = sqrt(1.4*287*t);  M = v./a;
CL  = W./(q*S);      CDi = k*CL.^2;       CDI = CD0 + CDi;
% Compressibility Correction
Mcc = 0.87-0.175*CL-0.83*tc;
m = 0.83-0.583*CL+0.111*CL.^2;
Mcrit = Mcc./(cosd(sweep).^m);
CDc = zeros(size(M));
idx = (CL<1.4) & (M>Mcrit);
x = M(idx)./Mcrit(idx);
```

```
CDc(idx) = (3.97e-9*exp(12.7*x)+1e-40*exp(81*x))*cosd(sweep)^3;
CDC = CDI + CDc;
DP = q.*S.*CD0;
Di = q.*S.*CDi;
DI  = q.*S.*CDI;
DC = q.*S.*CDC;
TA  = (rho/rhosl)*TSL;
Vstall = sqrt(2*W/(rho*S*CLmax));
end
```

A.7 Section 4.3

```
clear; clc; close all;
% Flight Parameters
hval = linspace(0, 15000, 200); % altitudes from sea level to ~19 km
[~, ~, ~, rhoSL] = atmosisa(0);
% Aircraft Parameters
WTO = 37019.73194*9.81; % Weight N
AR = 9.68; % Wing Area m^2
CLmax = 1.4; % Max Lift Coefficient
CD0 = 0.0107; % Parasitic Drag Coefficient
e = 0.90; % Oswald Efficiency Factor
b = 30.987; % Wingspan m
S = b^2 / AR; % Aspect Ratio
k = 1/(pi*e*AR);
T_SL = 70000; % Sea-level Thrust N
Vmax = zeros(size(hval));
Vmin = zeros(size(hval));
Vstall = zeros(size(hval));
for i = 1:length(hval)
 h = hval(i);

 [~, ~, ~, rho] = atmosisa(h);
% Density ratio
 sigma = rho / rhoSL;
% Thrust available
 T_h = sigma * T_SL;
 sqrtTerm = sqrt( 1 - (4*k*CD0) / ( (T_h / WTO) ^ 2 ) );
 q_max = (T_h)/(2*CD0*S) * (1 + sqrtTerm);
 q_min = (T_h)/(2*CD0*S) * (1 - sqrtTerm);
 % Velocity
 Vmax(i) = sqrt( 2*q_max / rho );
 Vmin(i) = sqrt( 2*q_min / rho );
% Stall speed
 Vstall(i) = sqrt( (2*WTO) / (rho*S*CLmax) );
end

crossover_velocity = 232.998;
idx = find(Vmin > crossover_velocity, 1, 'first');
idx1 = find(Vmax < crossover_velocity, 1, 'first');
if ~isempty(idx)
    Vmin(idx:end) = NaN;
end
if ~isempty(idx1)
    Vmax(idx:end) = NaN;
```

```
end

% Plot + Fill
figure; hold on; grid on
Vlow = max(Vstall, Vmin);
valid = Vlow <= Vmax & ~isnan(Vlow) & ~isnan(Vmax);
h_ok = hval(valid);
Vlow_ok = Vlow(valid);
Vmax_ok = Vmax(valid);
xPoly = [Vlow_ok , fliplr(Vmax_ok)];
yPoly = [h_ok , fliplr(h_ok)];
fill(xPoly, yPoly, [0.80 1.00 0.80], ...
'EdgeColor','none', 'FaceAlpha',0.30, 'DisplayName','Flight Envelope');
plot(Vmax, hval,'b','LineWidth',2,'DisplayName','V_{max}');
plot(Vmin, hval,'r','LineWidth',2,'DisplayName','V_{min}');
plot(Vstall, hval,'k','LineWidth',2,'DisplayName','V_{stall}');
xlabel('Velocity V (m/s)');
ylabel('Altitude h (m)');
title('HROne Flight Envelope');
legend('Location','best');
xlim([0 max(Vmax)*1.05]);
ylim([0 max(hval)]);
set(gca,'FontSize',11);
```

A.8 Section 4.3.2

```
    % Trade Study: Overall Pressure Ratio vs. High-Pressure Compressor Exit Flow
clear; clc; close all;

engines = {
    'Rolls-Royce Pearl 700',    18750, 4000, 0.638, 43, 4.20;
    'Honeywell HTF7000',         7300, 1665, 0.623, 30, 3.70;
    'NASA HyTEC (Theoretical)',  8000, 1980, 0.610, 52, 2.80;
    'GE CF34-3B',                9000, 1580, 0.640, 30, 3.90;
    'Rolls-Royce AE 3007',       8800, 1640, 0.640, 28, 3.85
};

% Extract data
n = size(engines, 1);
opr = zeros(n, 1);
hpc_flow = zeros(n, 1);
labels = cell(n, 1);

for i = 1:n
    labels{i} = engines{i,1};
    opr(i) = engines{i,5};
    hpc_flow(i) = engines{i,6};
end

% Plot formatting
figure('Color', 'w');
scatter(hpc_flow, opr, 120, 'filled', 'MarkerFaceColor', [0.2 0.6 0.9]);
xlabel('High-Pressure Compressor Exit Corrected Flow (lbm/s)', 'FontSize', 12, 'FontWeig
ylabel('Overall Pressure Ratio', 'FontSize', 12, 'FontWeight', 'bold');
title('Trade Study: Overall Pressure Ratio vs. High-Pressure Compressor Exit Corrected F
    'FontSize', 14, 'FontWeight', 'bold');
grid on;
xlim([2.5 4.4]);
ylim([25 55]);


for i = 1:n
    text(hpc_flow(i), opr(i) + 1.2, labels{i}, ...
        'FontSize', 10, 'HorizontalAlignment', 'center');
end


set(gca, 'FontSize', 11, 'LineWidth', 1.2);
box on;
```

A.9 Section 4.4.2 (1)

```
clear; clc; close all;
% Flight and Aircraft Parameters
h1 = 0;
h2 = 6000;
g = 9.81;
q = linspace(0,25e3);
[~,~,~,rhosl] = atmosisa(0);
WTO = 37019.73194*9.81; % N
CLmax = 1.4;
CD0 = 0.0107;
e = 0.90;
b = 30.987; % m
AR = 9.68;
S = b^2/AR; % m²
sweep = 35; % deg
tc = 0.10;
TSL = 70000; % N
k = 1/(pi*e*AR);
nstruct = 4.4;
[struct, lift, thrust, ~, raero] = TurningEnvelope( ...
 WTO,S,CLmax,CD0,k,TSL,nstruct,q,h1,rhosl,g);
figure('Name','HROne Turning Envelope at Sea Level'); grid on; hold on

% Keep the original plot order as in your code
plot(q, struct ,'b' ,'LineWidth',2,'DisplayName','n_{struct}');
plot(q, lift ,'r' ,'LineWidth',2,'DisplayName','C_{L,max}');
plot(q, thrust ,'k--','LineWidth',2,'DisplayName','T_{max}');
plot(q, raero ,'Color',[0 0.6 0],'LineWidth',2, 'DisplayName','n_{aero}');

% Fix the turn envelope shaded area (points 2 and 4)
Rlow = max([struct ; lift ; thrust ; raero],[],1,'omitnan');
q_stall = WTO /( S * CLmax ); % q at n = 1 (level-flight stall)
valid = ~isnan(Rlow) & (q >= q_stall);
q_ok = q(valid);
Rlow_ok = Rlow(valid);

max_y = 1800; % Using the ylim max value
fill([q_ok, fliplr(q_ok)], [Rlow_ok, max_y*ones(size(Rlow_ok))], [0.80 1.00 0.80], ...
    'EdgeColor','none','FaceAlpha',0.30, 'DisplayName','Turn Envelope');

xlabel('Dynamic pressure q (N m^{-2})','FontSize',12);
ylabel('Turn radius r (m)','FontSize',12);
title('HROne Turning Envelope at Sea Level','FontSize',14);
```

```
legend('Location','best');
xlim([0 25e3]); ylim([0 1270]);

Rlow = max([struct; lift; thrust; raero], [], 1, 'omitnan');
% Mask out below-stall region
q_stall = WTO/(S*CLmax);
valid = (q >= q_stall) & ~isnan(Rlow);
R_env = Rlow(valid);
q_env = q(valid);
% Global minimum
[rmin, idx] = min(R_env);
q_at_min = q_env(idx);
fprintf('Minimum turn radius r_{min} = %.1f m at q = %.1f N/m^2\n', ...
 rmin, q_at_min);

function [struct, lift, thrust, w, raero] = TurningEnvelope( ...
 WTO,S,CLmax,CD0,k,TSL,nstruct,q,h,rhosl,g)
 [~,~,~,rho] = atmosisa(h);
% structural limit
 struct = 2*q ./ (rho*g*sqrt(nstruct^2 - 1));
 struct(~isreal(struct)) = NaN;
% aerodynamic (stall) limit
 qsafe = (CLmax*q) > (WTO/S);
 lift = NaN(size(q));
 lift(qsafe) = 2*q(qsafe)*(WTO/S) ./ (rho*g*sqrt((CLmax*q(qsafe)).^2 - (WTO/S)^2));
% aerodynamic limit
 sigma = rho / rhosl;
 TAW = sigma * TSL / WTO;
 Em = 1 / (2*sqrt(k*CD0));
 naero = Em * TAW;
 raero = 2*q ./ (rho*g*sqrt(naero^2 - 1));
 raero(~isreal(raero)) = NaN;
% thrust-limited
 TA = (rho/rhosl)*TSL;
 n = sqrt( (TA - CD0*q*S) .* (q*S) /(k*WTO^2) );
 thrust = NaN(size(q));
 valid_thrust = (n.^2 > 1);
 thrust(valid_thrust) = 2*q(valid_thrust) ./ (rho*g*sqrt(n(valid_thrust).^2 - 1));
% Instantaneous turn rate (q)
 w = sqrt(n.^2 - 1).*g ./ sqrt(2*q./rho);
end
```

A.10 Section 4.4.2 (2)

```
clear; clc; close all

% Aircraft and Flight Parameters
h       = 0;
g       = 9.81;                      % m/s²

WTO     = 37019.73194*g;          % N
b       = 30.987;                      % m
AR      = 9.68;
S       = b^2/AR;                    % m²
CLmax   = 1.4;
CD0     = 0.0107;
e       = 0.90;
k       = 1/(pi*e*AR);
TSL     = 70000;
sweep   = 35;                        % deg
tc      = 0.10;                      % t/c
nstruct = 4.4;

[~,~,~,rho]  = atmosisa(h);          % density
[~,~,~,rho_c]= atmosisa(13106.4);    % density at ceiling
sigma        = rho ./ (1.225);

V = linspace(0,350,1200);            % m/s   (change range if desired)
q = 0.5*rho.*V.^2;                   % dynamic pressure

n_stall = (q*S*CLmax) ./ WTO;        % eqn:  L = n W

n_struct = nstruct.*ones(size(V));   % horizontal line

TA      = sigma*TSL;                 % thrust available at altitude
n_th    = sqrt( max(0,(TA - CD0*q*S).*(q*S) ./ (k*WTO^2)) ); % Eqn from hand-out
n_th( n_th<1 ) = NaN;                % ignore sub-level-flight values

Em     = 1/(2*sqrt(k*CD0));
n_aero = Em * (sigma*TSL/WTO);
n_aero = n_aero .* ones(size(V));

n_upper = min([n_stall; n_struct; n_th], [], 1,'includenan');

n_lower = zeros(size(V));

valid = n_upper >= n_lower;
```

```
xPoly = [V(valid)          fliplr(V(valid))];
yPoly = [n_upper(valid)    fliplr(n_lower(valid))];

% Vn Plot
figure('Name','HR-1  V{n Diagram','Color','w'); hold on; grid on
fill(xPoly,yPoly,[0.85 1.0 0.85],...
     'EdgeColor','none','FaceAlpha',0.32,...
     'DisplayName','Turn Envelope');

plot(V,n_struct ,'b' ,'LineWidth',2,'DisplayName','n_{struct}');
plot(V,n_stall  ,'r' ,'LineWidth',2,'DisplayName','C_{L,max}');
plot(V,n_th     ,'k--','LineWidth',2,'DisplayName','T_{max}');
plot(V,n_aero   ,'Color',[0 0.55 0],'LineWidth',2,...
     'DisplayName','n_{aero}');


xlabel('Airspeed  V  (ms^{-1})');
ylabel('Load Factor  n');
title('HROne V{n Graph at 6000m','FontSize',14,'Interpreter','latex');
legend('Location','best');
axis([0 350 1 5]); box on
set(gca,'FontSize',11)
```

A.11 Section 4.4.3 (1)

```
clear; clc; close all;

% Flight Parameters
h1 = 11;      % Airport Altitude (m)
[~, ~, ~, rho] = atmosisa(h1);  % Density
[~, ~, ~, rhosl] = atmosisa(0);  % Density
g = 9.81; %Gravity

% Aircraft
WTO        = 37019.73194*g;     % Weight (N)
CLmax = 1.4*0.8; %Max Coefficient of Lift
CL = 0.4; %Lift Coefficient
CD0        = 0.0107;     % Parasitic Drag Coefficient
e          = 0.9;      % Oswald Efficiency Factor
b          = 30.987;      % Wingspan (m)
h = 2; %Height above ground
AR = 9.68;
S = b^2 / AR;   % Aspect Ratio
mu = 0.03; % Ground-Roll Friction
TSL = 70000*1.05; % Thrust
sigma = rho/rhosl; % Density Ratio
T = sigma*TSL; %Thrust at Altitude
phi = ((16*(h/b))^2)/(1+(16*(h/b))^2); % Ground Effect
k = 1/(pi*e*AR); % Induced drag factor

VLo = 1.2*((2*WTO)/(rho*S*CLmax))^0.5;
q = 0.5*rho*(0.7*VLo)^2;

L = CLmax*q*S;
Dp = CD0*q*S;
Di = phi*k*CLmax^2*q*S;
Dphi = Dp+Di;

dlo = (1.44*WTO^2/(rho*g*S*CLmax*(T-Dphi+(mu*(WTO-L)))))
```

A.12 Section 4.4.3 (2)

```
clear; clc; close all;

% Flight Parameters
h1 = 0;      % Airport Altitude (m)
[~, ~, ~, rho] = atmosisa(h1);  % Density
[~, ~, ~, rhosl] = atmosisa(0);  % Density
g = 9.81; %Gravity

% Aircraft
WL        = 20112.97214*g;     % Weight (N)
S          = 99.2;  % Wing Area (m^2)
CLmax = 0.8*1.4; %Max Coefficient of Lift
CL = 0.4; %Lift Coefficient
CD0        = 0.0107;     % Parasitic Drag Coefficient
e          = 0.9;      % Oswald Efficiency Factor
b          = 30.987;      % Wingspan (m)
h = 3.75; %Height above ground
AR         = b^2 / S;   % Aspect Ratio
mur = 0.16; % Ground-Roll Friction
TSL = 70000*1.05; % Thrust
sigma = rho/rhosl; % Density Ratio
Trev = 0.3*sigma*TSL; %Thrust at Altitude
phi = ((16*(h/b))^2)/(1+(16*(h/b))^2); % Ground Effect
k = 1/(pi*e*AR); % Induced drag factor

VT = 1.3*((2*WL)/(rho*S*CLmax))^0.5;
q = 0.5*rho*(0.7*VT)^2;

L = CLmax*q*S;
Dp = CD0*q*S;
Di = phi*k*CLmax^2*q*S;
Dphi = Dp+Di;

dg = (1.69*WL^2/(rho*g*S*CLmax*(Trev+Dphi+(mur*(WL-L)))))
```

A.13 Section 4.4.4

```
clear; clc; close all;

% Flight Parameters
h1 = 0;      %Altitude (m)
h2 = 6000; %Altitude (m)
[~, ~, ~, rho] = atmosisa(h1);  % Density
[~, ~, ~, rhosl] = atmosisa(0);  % Density
g = 9.81; %Gravity
v = linspace(0, 400, 200); %Velocity

% Aircraft
WTO         = 37019.73194*g;     % Weight (N)
S           = 99.2;  % Wing Area (m^2)
CLmax       = 1.4; %Max Coefficient of Lift
CL          = 0.4; %Lift Coefficient
CD0         = 0.0107;     % Parasitic Drag Coefficient
e           = 0.9;      % Oswald Efficiency Factor
b           = 30.987;      % Wingspan (m)
h           = 2; %Height above ground
AR          = b^2 / S;   % Aspect Ratio
L           = 35; % Wing Sweep Angle (°)
tc          = 0.1*0.6; %Thickness to Chord Ratio
TSL         = 70000; % Thrust
sigma       = rho/rhosl; % Density Ratio
T           = sigma*TSL; %Thrust at Altitude
phi         = ((16*(h/b))^2)/(1+(16*(h/b))^2); % Ground Effect
k           = 1/(pi*e*AR); % Induced drag factor

[TA0, RoCi0, RoCc0, M0] = RoC(h1, v, WTO, S, CD0, k, TSL, L, tc, CLmax);
[TA9, RoCi9, RoCc9, M9] = RoC(h2, v, WTO, S, CD0, k, TSL, L, tc, CLmax);

figure('Name','Rate of Climb vs. Velocity');
hold on; grid on;
plot(v, RoCi0, 'r',  'LineWidth',2, 'DisplayName','Rate of Climb Incompressible at 0m');
plot(v, RoCi9, 'b',  'LineWidth',2, 'DisplayName','Rate of Climb Incompressible Drag at
plot(v, RoCc0, '--r',  'LineWidth',2, 'DisplayName','Rate of Climb Compressible Drag at
plot(v, RoCc9, '--b',  'LineWidth',2, 'DisplayName','Rate of Climb Compressible Drag at
legend('Location', 'best');
xlabel('Velocity (m/s)','FontSize',12);
ylabel('Rate of Climb (m/s)','FontSize',12);
xlim([40 350])
ylim([0 50])
title('Rate of Climb vs Velocity of HROne','FontSize',14);
```

```
set(gca,'FontSize',12);
hold off;


RoCmaxi0 = max(RoCi0);
RoCmaxc0 = max(RoCc0);
RoCmaxi9 = max(RoCi9);
RoCmaxc9 = max(RoCc9);

fprintf('  Incompressible RoCmax at 0m = %.2f m/s\n', RoCmaxi0);
fprintf('  Compressible RoCmax at 0m = %.2f m/s\n', RoCmaxc0);
fprintf('  Incompressible RoCmax at 9000m = %.2f m/s\n', RoCmaxi9);
    fprintf('  Compressible RoCmax at 9000m = %.2f m/s\n', RoCmaxc9);

function [TA, RoCi, RoCc, M] = RoC(h, v, WTO, S, CD0, k, TSL, L, tc, CLmax)
    %Flight Parameters
    [~, ~, ~, rhosl]   = atmosisa(0);  % Density at sea level
    [t, ~, ~, rho]   = atmosisa(h);  % Density at sea level

    q   = 0.5 * rho * v.^2; %Dynamic Pressure

    a = sqrt(1.4*287*t); %Speed of Sound m/s
    M = v/a; %Mach Number at height h

    %Airplane Parameters
    Vstall = sqrt((2*WTO)/(rho*S*CLmax)); %Stall Speed

    CL  = WTO ./ (q * S); %Coefficient of Lift
    CDi = k * CL.^2; %Induced Drag Coefficient
    CDI  = CD0 + CDi; %Incompressible Drag Coefficient

    Mcc = 0.87 - (0.175*CL) - (0.83*tc);
    m = 0.83 - (0.583*CL) + (0.111*CL.^2);
    Mccl = (Mcc)./(cosd(L).^m);
    CDc = zeros(size(M));
    valid = (CL < 1.4);
    x = M(valid) ./ Mccl(valid);
    CDc(valid) = ((3.97e-9)*exp(12.7*x) + (1e-40)*exp(81*x)) * (cosd(L)^3); %Coefficient

    CDC = CD0 + CDi + CDc; %Compressible Drag Coefficient

    DI  = q .* S .* CDI; %Incompressible Drag
    DC = q .* S .* CDC; %Compressible Drag

    sigma = rho / rhosl; %Density Ratio
```

```
    TA = sigma * TSL; %Thrust available at altitude h

    RoCi = (v.*(TA-DI))/WTO; %Rate of Climb Incompressible
    RoCc = (v.*(TA-DC))/WTO; %Rate of Climb Compressible
end
```

A.14 Section 4.4.5

```
clear;  clc;  close all

% Aircraft and Flight Parameters
W     = 37019.73194*9.81;           % weight (N)
b     = 30.987;                      % span  (m)
AR    = 9.68;                       % aspect ratio
S     = b^2/AR;                     % wing area (m²)
CD0   = 0.0107;                     % zero-lift drag
e     = 0.90;                       % Oswald efficiency
k     = 1/(pi*e*AR);                % induced-drag factor
TSL   = 70000;                       % thrust (N)

V = linspace(40,340,400);           % true airspeed (m/s)
powerReq = @(rho) ...
    (0.5*rho.*V.^2).*S .* (CD0 + k*(W./(0.5*rho.*V.^2.*S)).^2) .* V;

% Sea Level Plot
[~,~,~,rhoSL] = atmosisa(0);
P_R_SL = powerReq(rhoSL);
P_A_SL = TSL .* V;

figure('Name','Power vs Velocity • Sea Level','Color','w'); hold on; grid on
plot(V, P_R_SL/1e6 ,'k' ,'LineWidth',2,  'DisplayName','P_{Req}');
plot(V, P_A_SL/1e6 ,'b--','LineWidth',2, 'DisplayName','P_{Avail}');
xlabel('True airspeed  V  (m\,s^{-1})');
ylabel('Power  (MW)');
legend('Location','northwest'); axis([min(V) max(V) 0 inf]);
set(gca,'FontSize',11);

% Cruise Plot
hCruise = 13106.4;
[~,~,~,rhoCR] = atmosisa(hCruise);
sigma  = rhoCR / rhoSL;

P_R_CR = powerReq(rhoCR);
P_A_CR = (TSL*sigma) .* V;

figure('Name','Power vs Velocity • 43000ft','Color','w'); hold on; grid on
plot(V, P_R_CR/1e6 ,'k' ,'LineWidth',2,  'DisplayName','P_{Req}');
plot(V, P_A_CR/1e6 ,'r--','LineWidth',2, 'DisplayName','P_{Avail}');
xlabel('True airspeed  V  (m\,s^{-1})');
ylabel('Power  (MW)');
legend('Location','northwest'); axis([min(V) max(V) 0 inf]);
```

```
set(gca,'FontSize',11);
```

A.15 Section 4.6

```
clear; clc; close all;

%% Unit conversion
psf2Pa = 47.8803;           % 1 psf = 47.8803 Pascal
ft2m2 = 0.092903;           % 1 ft^2 = 0.092903 m^2
ft2m = 0.3048;              % 1 ft = 0.3048 m
lb2kg = 0.45359237;         % 1 lb = 0.45359237 kg

%% Placeholder parameters (user must assign these)
W_e = 18998.1;
S_w   = 99.2;    % wing area [m^2]
W_fw  = 18506;    % wing fuel weight [kg]
A     = 9.679376704;    % aspect ratio [-]
Lambda = 35;   % wing sweep angle [deg]
q     = 1;    % dynamic pressure [Pa]
lambda = 0.4165232294;   % taper ratio [-]
t_c   = 0.1;    % thickness-to-chord ratio [-]
N_z   = 4.4;    % ultimate load factor [-]
W_dg  = 38745.86;    % design gross weight [kg]

% Horizontal tail parameters
S_ht = 19.84;     % horizontal tail area [m^2]
A_ht = 7.75;     % horizontal tail AR [-]
Lambda_ht = 40; % horizontal tail sweep [deg]
lambda_ht = 0.4165232294; % horizontal tail taper ratio [-]
q_ht = 9772.4;     % dynamic pressure at HT [Pa]

% Vertical tail parameters
S_vt = 9.92;     % vertical tail area [m^2]
A_vt = 3.87;     % vertical tail AR [-]
Lambda_vt = 10; % vertical tail sweep [deg]
lambda_vt = 0.4431079721; % vertical tail taper ratio [-]
H_t = 2.75/2;      % HT position above fuselage centerline [m]
H_v = 2.75;      % fuselage height [m]
q_vt = 9772.4;     % dynamic pressure at VT [Pa]

% Fuselage parameters
S_f = 216.77;      % fuselage wetted area [m^2]
L_t = 12.005;      % tail moment arm [m]
LD_ratio = 20; % lift-to-drag ratio [-]
q_fuse = 9772.4;   % dynamic pressure at fuselage [Pa]
```

```
W_press = 1;  % pressurization structure weight [kg]

% Landing gear parameters
N_l = 3;      % number of landing gear struts [-]
W_l = W_dg/3;      % weight supported per strut [kg]
L_m = 1.5;      % length of main gear [m]

% Engine / propulsion
W_en = 1360;     % single engine weight [kg]
N_en = 2;     % number of engines [-]

% Payload
W_payload = 1231.3; % payload weight [kg]

%% Weight Equations (results in kg)

% Wing weight (Raymer Eq. 15.46)
W_wing_kg = 0.036 * (S_w / ft2m2)^0.758 * (W_fw / lb2kg)^0.0035 * ...
            ((A / cosd(Lambda)^2)^0.6) * ...
            (q / psf2Pa)^0.006 * lambda^0.04 * ...
            ((100 * t_c / cosd(Lambda))^-0.3) * ...
            ((N_z * W_dg / lb2kg)^0.49) * lb2kg;

% Horizontal tail weight (Eq. 15.47)
W_ht_kg = 0.016 * (N_z * W_dg / lb2kg)^0.414 * (q_ht / psf2Pa)^0.168 * ...
          (S_ht / ft2m2)^0.896 * ((100 * t_c / cosd(Lambda_ht))^-0.12) * ...
          ((A_ht / cosd(Lambda_ht)^2)^0.043) * lambda_ht^-0.02 * lb2kg;

% Vertical tail weight (Eq. 15.48)
W_vt_kg = 0.073 * (1 + 0.2 * (H_t / H_v)) * (N_z * W_dg / lb2kg)^0.376 * ...
          (q_vt / psf2Pa)^0.122 * (S_vt / ft2m2)^0.873 * ...
          ((100 * t_c / cosd(Lambda_vt))^-0.49) * ...
          ((A_vt / cosd(Lambda_vt)^2)^0.357) * lambda_vt^0.039 * lb2kg;

% Fuselage weight (Eq. 15.49)
W_fuse_kg = (0.052 * (S_f / ft2m2)^1.086 * (N_z * W_dg / lb2kg)^0.177 * ...
             (L_t / ft2m)^-0.051 * LD_ratio^-0.072 * ...
             (q_fuse / psf2Pa)^0.241 + (W_press / lb2kg)) * lb2kg;

% Main landing gear (Eq. 15.50)
W_mlg_kg = 1 *(0.095 * ((N_l * W_l / lb2kg)^0.768) * ((L_m / 12 / ft2m)^0.409) * lb2kg);

% Nose landing gear (Eq. 15.51)
W_nlg_kg = 1* (0.125 * ((N_l * W_l / lb2kg)^0.566) * ((L_m / 12 / ft2m)^0.845) * lb2kg);
```

```
% Installed engine weight (Eq. 15.52)
W_engines_kg = 2.575 * (W_en / lb2kg)^0.922 * N_en * lb2kg;

%% Control systems (placeholder)
W_controls_kg = 0.04 * W_e; % control systems (avionics, instruments, etc.) [kg]

%% Systems (placeholder)
W_systems_kg = 0.09 * W_e; % systems (hydraulics, electrical, air, etc.) [kg]

%% Total Weights (in kg)
allWeights_kg = [W_wing_kg, W_ht_kg, W_vt_kg, W_fuse_kg, ...
                 W_mlg_kg + W_nlg_kg, W_engines_kg, ...
                 W_payload, W_controls_kg, W_systems_kg];




%% Display breakdown
componentLabels = {'Wing', 'HT', 'VT', 'Fuselage', 'Landing Gear', 'Propulsion', ...
                   'Payload', 'Control Systems', 'Systems'};
% Filter out zero values
nonzeroIdx = allWeights_kg > 1e-2;
nonzeroWeights = allWeights_kg(nonzeroIdx);
nonzeroLabels = componentLabels(nonzeroIdx);

% Create pie chart
figure;
p = pie(nonzeroWeights);  % returns handles

% Find text handles (percentage labels)
percentHandles = findobj(p, 'Type', 'text');

% Attach label + percentage
for i = 1:length(percentHandles)
    percentStr = percentHandles(i).String;        % e.g. '12.3%'
    labelStr = nonzeroLabels{i};                  % correct label for this slice
    percentHandles(i).String = sprintf('%s (%s)', labelStr, percentStr);
end
```

A.16 Section 4.6.1

```
clear; clc; close all;

b = 30.987; %Wingspan [m]

d_fuselage = 2.75; %Fuselage Diameter [m]
MTOW = 38745*9.8; %Maximum takeoff weight, [N]
w_engine = 1360*9.8; %engine weight [N]
l_engine = 2.49; %distance from wing root to engine [m]


%1000 ;
n_max = 2.5; %maximum load factor, specified in FAR requirements. [-]
FOS = 1.5; %Factor of safety, [-]
n = n_max*FOS;
%w_wing = 0.0051*((MTOW*n)^0.552) *(S^0.649)*(AR^0.5)*((t_c)^-0.4)*((1+taper)^0.1) *(cos

w_wing = 25523/2; %[N];
w_wing_total = w_wing;




L = n*MTOW/2; %lift generated by each wing during max load factor [N]

l_beam = (b-d_fuselage)/2;

L_distribution = L/l_beam; %lift distribution on each wing [N/m]

fun = @(x) L_distribution*x;

M_L  = integral(fun,0,l_beam);

M_engine = -w_engine*l_engine; %[Nm]
M_wing = -w_wing_total*l_beam/2; %[Nm]

M_root = M_wing + M_engine + M_L; %[Nm]
```

A.17 Section 4.6.2

```
clc
clear

% Given values
```

```
h_n = 0.574;                % Neutral point (as fraction of MAC)
h_cg_minus_hac = 0.0494;   % Forward CG limit (h_cg - h_ac)

% Create the plot with specific size
figure('Position', [100, 100, 1200, 800]);
hold on;

% Set arbitrary weight range for the vertical lines (since no specific weights given)
yLimits = [0 1];

% Fill the region between the two lines
x = [h_cg_minus_hac, h_n, h_n, h_cg_minus_hac];
y = [0, 0, 1, 1];
fill(x, y, [0.8 0.9 1], 'EdgeColor', 'none', 'FaceAlpha', 0.5);

% Plot vertical lines for forward CG limit and neutral point with dashed lines
plot([h_cg_minus_hac h_cg_minus_hac], yLimits, 'r--', 'LineWidth', 4);
plot([h_n h_n], yLimits, 'b--', 'LineWidth', 4);

% Plot vertical line at x = 0 to show LE WING position
plot([0 0], yLimits, 'k-', 'LineWidth', 3);

% Add labels and title
xlabel('CG Location (as fraction of MAC behind LE of wing)', 'FontSize', 24);
ylabel('Weight', 'FontSize', 24);
title('CG Envelope', 'FontSize', 32, 'FontWeight', 'bold');

% Add legend
legend('Allowable CG Range', 'Forward CG Limit', 'Neutral Point (AFT Limit)', 'Location'

% Remove y-axis ticks since no specific weight values are provided
set(gca, 'YTick', []);

% Adjust axes for better appearance
ax = gca;
ax.FontSize = 20;
ax.LineWidth = 3;

% Create more space for text labels by adjusting y-limits
ylim([-0.2 1.2]);

% Add text labels for the values with background boxes for clarity
% Forward CG limit label
text(h_cg_minus_hac, -0.1, 'Forward CG Limit:', 'FontSize', 24, 'HorizontalAlignment', '
text(h_cg_minus_hac, -0.15, [num2str(h_cg_minus_hac, '%.4f') ' MAC'], 'FontSize', 28, 'H
```

```
% Neutral point label
text(h_n, -0.1, 'Neutral Point:', 'FontSize', 24, 'HorizontalAlignment', 'center');
text(h_n, -0.15, [num2str(h_n, '%.4f') ' MAC'], 'FontSize', 28, 'HorizontalAlignment', '

% Label origin as LE WING with arrow and clear positioning
text(-0.035, 0.5, 'LE WING (x = 0)', 'FontSize', 24, 'FontWeight', 'bold', 'HorizontalAl

% Add grid and adjust the appearance
grid on;
box on;
xlim([-0.05 0.7]);

% Add some padding around the plot
set(gca, 'Position', [0.15, 0.2, 0.75, 0.7]);

hold off;

% Print the values for reference
fprintf('CG Envelope Limits:\n');
fprintf('Forward CG Limit: %.4f MAC\n', h_cg_minus_hac);
fprintf('Neutral Point: %.4f MAC\n', h_n);
```

A.18 Section 4.9

```
    % ----------------------------------------------------------------------
% CONSTANT-CL TRADE STUDY CL = 0.4
% ----------------------------------------------------------------------
clear; clc; close all;

%% 1) FLIGHT & ATMOSPHERIC Characteristics
h    = 43000/3.281;           % m
Vcr  = 271;                   % m/s (Mach 0.92)
g    = 9.81;
[T,~,~,rho] = atmosisa(h);    % ISA density at altitude
gamma = 1.4;  R = 287;
a     = sqrt(gamma*R*T);      % speed of sound

%% 2) WEIGHTS & PROPULSION
W_payload = 1231.5 * g; % Max pay weight, Napa economic mission
WTO = 38745.86*g;            % N
fuel_fraction  =   0.4780 ;
W_empty = 18995.9949*g;
W_fuel         = fuel_fraction * WTO;
WTO = W_empty + W_payload + W_fuel;
W1             = WTO - W_fuel;
SFC            = (0.642/3600)*0.95;         % 1/s

%% 3) DRAG MODEL CONSTANTS
CD0    = 0.014;
e      = 0.9;
sweep  = deg2rad(35);
tc     = 0.10*0.6; % *0.6 to account for supercritical airfoil

%% 4) FIXED CRUISE LIFT COEFFICIENT → wing area
%S_req  = WTO / (0.5*rho*Vcr^2 * CL0);  % [m^2], constant wing planform
S0 = 99.02;
CL0  = WTO /(0.5*rho*Vcr^2*S0)

%% 5) DESIGN SWEEP: AR only, derive b = sqrt(AR·S0)
AR_vals    = linspace(6, 15, 400);
b_vals     = sqrt(AR_vals * S0);
Minf       = Vcr / a;

% Preallocate
Range              = nan(size(AR_vals));
fuel_burn_metric   = nan(size(AR_vals));
```

```
for i = 1:numel(AR_vals)
    AR = AR_vals(i);
    b  = b_vals(i);
    S  = S0;
    CL = CL0;

    % Induced drag
    CDi = CL^2 / (pi*e*AR);

    % Compressibility penalty
    Mcc_0      = 0.87 - 0.175*CL - 0.83*tc;
    m          = 0.83 - 0.583*CL + 0.111*CL^2;
    Mcc_sweep  = Mcc_0 / (cos(sweep)^m);
    x          = Minf / Mcc_sweep;
    deltaCDc   = (3.97e-9*exp(12.7*x) + 1e-40*exp(81*x)) * cos(sweep)^3;

    % Total CD
    CD_total = CD0 + CDi + deltaCDc;

    % Breguet range
    Range(i) = (Vcr / SFC) * (CL / CD_total) * log(WTO / W1);

    % Fuel-burn metric
    fuel_burn_metric(i) = (W_fuel / W_payload) / Range(i);
end

%% 6) CONSTRAINTS
b_max    = 28.96*1.07; % m
AR_limit = 11; % was 15
mask     = (b_vals > b_max) | (AR_vals > AR_limit);
fuel_burn_metric(mask) = NaN;

%% 7) FIND & REPORT OPTIMUM
[min_val, idx] = min(fuel_burn_metric);
best_AR = AR_vals(idx);
best_b  = b_vals(idx);

fprintf('Best fuel-burn metric [1/m] : %.5e\n', min_val);
fprintf('→ AR = %.2f,   b = %.2f m  (S = %.1f m^2)\n', ...
        best_AR, best_b, S0);

%% 8) PLOT
figure;
plot(AR_vals, fuel_burn_metric, 'LineWidth',1.5);
hold on;
```

```
plot(best_AR, min_val, 'ro','MarkerSize',8,'LineWidth',2);
xlabel('Aspect Ratio, AR [-]');
ylabel('Fuel Burn Metric: $\frac{W_{fuel}}{W_{payload}*Range}$ [1/m]', 'Interpreter','la
title('Constant CL = 0.4. Trade: Fuel Burn Metric vs. AR');
grid on;
text(best_AR+0.1, min_val, ...
     sprintf('Min @ AR=%.2f, b=%.2f m',best_AR,best_b));
```