

COVER SHEET – COURSEWORK

TITLE: Preliminary Aerodynamic Design, Wing Optimisation, and
Emergency Parachute Sizing for a Four-Seat Electric General
Aviation Aircraft

MODULE: Aerodynamics 1 [ME4424]

ACADEMIC TERM: Autumn 2025/26

SUBMISSION DATE: 29/11/2025

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Abstract

The push for sustainable aviation has made electric aircraft a key area of development, but their design is constrained by the limited energy density of batteries. This makes aerodynamic efficiency paramount for achieving viable flight ranges. This report details the preliminary design of a wing and emergency parachute system for a new four-seat electric general aviation aircraft with a mass of 2,000 kg.

A systematic, computational approach was employed for the wing design. The analysis began by analysing a wide range of NACA 4-digit airfoils using XFOIL, leading to the selection of the NACA 3213 for its optimal balance of high lift-to-drag ratio and low pitching moment. This airfoil was then analysed in a 3D environment using XFLR5, where the wing planform was optimised to an aspect ratio of 10 and a taper ratio of 0.7. This configuration achieved a cruise lift-to-drag ratio (L/D) of 31.46. A plain flap system was also optimised, with a modest 5° flap deflection angle identified as optimal for both take-off and landing, successfully meeting the required lift coefficients while preserving aerodynamic efficiency. Using these results, the aircraft's theoretical range was calculated to be 308 km.

For safety, a two-stage emergency ballistic parachute system was designed. The analysis determined that a small, 4.4 m^2 reefed canopy is needed to decelerate the aircraft from cruise speed without exceeding $1g$, followed by a large 373 m^2 canopy to ensure a survivable vertical descent rate of 28 km/h. Finally, the designed wing was assessed for high-speed suitability, revealing a critical Mach number of 0.64, which confirms its unsuitability for high-subsonic flight without significant modification.

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Nomenclature

Symbol	Physical Property / Name	SI Unit
A	Area	m^2
A_{full}	Fully deployed parachute area	m^2
A_{reefed}	Reefed parachute area	m^2
AR	Aspect ratio	—
C_D	Drag coefficient (3-D)	—
$C_{D,0}$	Zero-lift drag coefficient	—
$C_{D,f}$	Skin-friction drag coefficient	—
$C_{D,i}$	Induced drag coefficient	—
$C_{D,para}$	Parachute drag coefficient	—
C_L	Lift coefficient (3-D)	—
C_m	Pitching-moment coefficient	—
C_p	Pressure coefficient	—
$C_{p,cr}$	Critical pressure coefficient	—
$C_{p,min}$	Minimum pressure coefficient	—
D	Drag force	N

Symbol	Physical Property / Name	SI Unit
E	Energy	J
L	Lift force	N
L/D	Lift-to-drag ratio	—
M	Mach number	—
M_{cr}	Critical Mach number	—
P	Static pressure	Pa
R	Specific gas constant for air	$\text{J}\cdot\text{kg}^{-1}\cdot\text{K}^{-1}$
Re	Reynolds number	—
S	Wing planform area	m^2
T	Absolute temperature	K
V	Velocity	m/s
$V_{approach}$	Landing approach velocity	m/s
V_{stall}	Stall speed	m/s
V_{TO}	Take-off velocity	m/s
$V_{vertical}$	Vertical descent velocity	m/s
a	Speed of sound	m/s
b	Wingspan	m
c	Chord length	m
c_r	Root chord	m
c_t	Tip chord	m
c_l	Airfoil-section lift coefficient	—
$c_{l,max}$	Maximum airfoil lift coefficient	—
e	Oswald efficiency factor	—
g	Gravitational acceleration	m/s^2
m	Mass	kg
$m_{aircraft}$	Aircraft mass	kg
$m_{battery}$	Battery mass	kg
α	Angle of attack	degrees or radians
γ	Ratio of specific heats (air)	—
δ	Flap deflection angle	degrees
η	Total powertrain efficiency	—
$\lambda = c_t/c_r$	Taper ratio	—
μ	Dynamic viscosity	$\text{Pa}\cdot\text{s}$
ρ	Air density	kg/m^3

1. Introduction

The aviation industry is seeking sustainable solutions to reduce its environmental impact. Electric aircraft offer a path to zero-emissions flight, but, are hindered by the low energy density of current batteries, which severely limits their range and payload.

This fundamental limitation makes aerodynamic efficiency the single most critical factor in electric aircraft design. Every optimisation translates directly into longer flight distances and more viable aircraft. The wing, as the primary source of both lift and drag, therefore becomes the central focus of the design process. Its performance dictates not only the aircraft's range but also its safety during take-off and landing.

This report details the preliminary aerodynamic design of a wing for a new four-seat, general-aviation electric aircraft with a take-off weight of 2,000 kg. Operating at a cruise speed of 300 km/h at 3,000 meters, the core objective is to maximise lift-to-drag ratio (L/D) to extend the aircraft's range, while ensuring the wing can also generate sufficient lift at low speeds for safe take-off and landing. Furthermore, recognizing that safety is paramount, this study also includes the design of an emergency whole-aircraft parachute system, a critical feature for occupant survival in the event of a catastrophic failure.

To achieve this, the project is divided into three integrated parts:

- Part A focuses on the wing itself. A structured, computational approach is employed using automated python scripts with XFOIL and XFLR5 to first select an optimal airfoil and then design the full 3D wing, including its planform geometry and high-lift flaps.
- Part B sizes the emergency parachute system, calculating the canopy area needed to decelerate the aircraft from cruise speed safely and ensure a survivable vertical descent.
- Part C steps back to evaluate the limitations of our low-speed wing design, analysing its suitability for high-speed flight and suggesting modifications that would be necessary for such a mission.

2. Objectives

This report aims to design a wing and emergency parachute system for a four-seat electric aircraft, with the focus on maximising cruise range through aerodynamic efficiency.

To achieve this, the following specific objectives were defined:

1. To Select and Justify an Optimal Airfoil:

- Identify a suitable NACA airfoil using a systematic, weighted selection process.
- The airfoil must exhibit a high lift-to-drag ratio (L/D) for cruise efficiency, a high maximum lift coefficient ($C_{l,max}$) for slow-speed operation, gentle stall characteristics, and a low pitching moment for stability.

2. To Determine the Optimal 3D Wing Planform:

- Using XFLR5, the ideal wing geometry must be found by evaluating the trade-offs of Aspect Ratio (AR) and Taper Ratio (λ).

3. To Design an Effective High-Lift Flap System:

- Configure a plain flap (defining its chord, span, and deflection angle) that provides the necessary lift coefficient (C_L) for specified take-off and landing speeds using the preliminary clean wing design.

4. To Calculate the Aircraft's Theoretical Range:

- Using the optimised wing's L/D ratio and given battery parameters, compute the aircraft's achievable range, neglecting fuselage drag.

5. To Size an Emergency Parachute System:

- Calculate the required parachute area for a two-stage deployment sequence. Stage one involves a deceleration phase with a limit of 1g, while stage two involves a vertical descent phase requiring a constant velocity of 28 km/h to ensure occupant safety.

6. To Analyse High-Speed Suitability:

- Determine the critical Mach number (M_{cr}) of the designed wing and assess its suitability for high-speed (Mach 0.8) flight.

3. Literature review

This review covers the key concepts and methods from existing literature that directly support the three parts of this project: wing design, parachute systems, and high-speed flight considerations.

3.1 Part A: Airfoil and wing optimisation

Selecting the right airfoil and wing planform is critical for aircraft performance. The standard approach involves two steps: first analysing a 2D airfoil, then modelling the full 3D wing.

For the initial airfoil analysis, tools like XFOIL are industry standards. XFOIL uses a combination of inviscid and viscous flow models to accurately predict how an airfoil will perform, providing essential data on lift, drag, and pitching moment (Drela, 1989). Once a promising airfoil is found, its performance is evaluated in three dimensions using software like XFLR5. These programs use Vortex Lattice Methods to predict how the wing will behave as a whole, calculating effects like induced drag and spanwise lift distribution (Anderson, 2017).

The literature highlights clear trade-offs in this process. A high Aspect Ratio wing reduces induced drag, which is the dominant source of drag for an efficient aircraft, but it also increases weight and flutter risk (Raymer, 2018). For an electric aircraft, where battery energy is limited, maximising the Lift-to-Drag ratio (L/D) is the single most important goal for extending range (Brelje & Martins, 2019). However, this must be balanced against stability. An airfoil with a severe pitching moment requires a larger, heavier tail, which can cancel out any aerodynamic gains (Scholz, 2021).

3.2 Part B — Emergency parachute systems

Whole-aircraft parachute systems, like those made by Ballistic Recovery Systems (BRS), are proven safety devices in light aviation. Their design is based on a simple but crucial principle of managed deceleration.

To be survivable, the system must deploy in two stages (Knacke, 1992). First, a small, reefed (restricted) parachute opens to slow the aircraft rapidly without exceeding a safe force of about 1g. Once the speed has dropped significantly, the parachute fully inflates. This second stage is sized to create a gentle, controlled descent of around 7-8 m/s, ensuring a survivable impact (NASA, 2011).

In practice, designers often validate their calculations by comparing the mass-to-area ratio of their proposed parachute with those of certified systems, such as the BRS unit on a Cessna 172. This provides a quick, real-world check that the design is feasible (Ballistic Recovery Systems Inc., n.d.).

3.3 Part C — High-Mach Flight

The wing designed for an efficient, low-speed electric aircraft would perform poorly at high speeds. As an aircraft approaches and then exceeds Mach 0.3, compressibility effects become significant. The Critical Mach Number (M_{cr}) is the point at which the airflow over some part of the wing first becomes supersonic, leading to shockwaves and a sudden, large increase in drag known as "drag divergence" (Anderson, 2019).

Thick, high-lift airfoils optimised for low speeds have a low M_{cr} , making them unsuitable for transonic flight. Literature points to three main solutions for efficient high-speed flight (Bertin & Cummings, 2014; Whitcomb, 1976):

1. Thinner Airfoils: Reduce airflow acceleration over the wing.
2. Wing Sweep: Effectively makes the wing "see" a slower airflow.
3. Supercritical Airfoils: Use a unique flat upper surface to delay and weaken shockwaves.

These modifications are fundamental to designing for Mach 0.8 and are entirely different from the priorities for a low-speed electric aircraft, highlighting the specialised nature of wing design for different flight regimes.

4. Surrounding theory

This section covers the fundamental principles guiding this aircraft design, from basic aerodynamics to high-speed flight considerations.

4.1 Low-Speed Aerodynamics and Wing Design Theory

The wing design for this electric aircraft is based on incompressible flow theory, with optimisation for maximum aerodynamic efficiency being the primary objective.

4.1.1 Fundamental Forces, Coefficients, and Flight Regimes

The behaviour of an incompressible flow is governed by properties derived from ambient conditions. Air density (ρ) is given by the ideal gas law:

$$\rho = \frac{P}{RT} \quad [\text{Eqn. 1}]$$

where P is static pressure, R is the specific gas constant for air ($287 \text{ J}\cdot\text{kg}^{-1}\cdot\text{K}^{-1}$), and T is absolute temperature. The speed of sound (a) and Mach number (M) are defined as:

$$[\text{Eqn. 2}] \quad a = \sqrt{\gamma RT} \quad \text{and} \quad M = \frac{V}{a} \quad [\text{Eqn. 3}]$$

where γ is the ratio of specific heats (1.4 for air) and V is the flow velocity. The Reynolds number (Re), which characterises flow regime and viscous effects, is given by:

$$Re = \frac{\rho V c}{\mu} \quad [\text{Eqn. 4}]$$

where μ is dynamic viscosity and c is the characteristic chord length.

The fundamental aerodynamic forces of lift (L) and drag (D) are defined by:

$$\text{[Eqn. 5]} \quad L = \frac{1}{2} \rho V^2 S C_L \quad \text{and} \quad D = \frac{1}{2} \rho V^2 S C_D \quad \text{[Eqn. 6]}$$

where S is wing planform area, and C_L and C_D are the non-dimensional lift and drag coefficients, respectively (Anderson, 2016). For steady, level flight lift equals weight ($L = W$).

4.1.2 Take-off, Landing, and Regulatory Requirements

The required maximum lift coefficients for take-off and landing phases were determined using regulatory safety margins applied to the fundamental lift equation. For take-off, aviation regulations [FAR 25.107] specify $V_{TO} = 1.2 \times V_{Stall}$ [Eqn. 7], ensuring a 20% safety margin above stall speed (Federal Aviation Regulations, 2025a).

For landing approach, a more conservative requirement [FAR 25.125] mandates $V_{Approach} = 1.3 \times V_{Stall}$ [Eqn. 8], providing a 30% safety margin for emergency situations like aborted landings or “go-arounds” (Federal Aviation Regulations, 2025b). These regulatory margins directly dictate the minimum $C_{L,max}$ the wing-flap system must achieve in each configuration.

4.1.3 Airfoil Aerodynamics and Selection

Airfoil performance is characterised by the lift curve (C_L vs. α), drag polar (C_D vs. C_L), and pitching moment (C_m). The NACA 4-digit series, employed extensively in this study, defines an airfoil geometry through three parameters: maximum camber (percentage of chord), location of maximum camber (tenths of chord), and maximum thickness (percentage of chord) (Abbott & Von Doenhoff, 1959). A high maximum lift coefficient ($C_{L,max}$) is essential for meeting regulatory stall margins.

4.1.4 Wing Planform Geometry and Induced Drag

Extending from 2D airfoil to 3D wing theory introduces induced drag due to wingtip vortices. The total drag coefficient is expressed as:

$$C_D = C_{D,0} + C_{D,i} = C_{D,0} + \frac{C_L^2}{\pi A R e} \quad \text{[Eqn. 9]}$$

where $C_{D,0}$ is the zero-lift drag coefficient, $C_{D,i}$ is the induced drag coefficient, AR is the aspect ratio, and e is the Oswald efficiency factor (Raymer, 2018). The aspect ratio is defined as $AR = b^2/S$, where b is wingspan. This relationship demonstrates that high AR is the most effective means of reducing induced drag.

Planform geometry is determined through geometric relationships:

$$b = \sqrt{S \cdot AR} \quad \text{[Eqn. 10]}, \quad c_r = \frac{2S}{b(1 + \lambda)} \quad \text{[Eqn. 11]}, \quad c_t = \lambda \cdot c_r \quad \text{[Eqn. 12]}$$

where c_r is root chord, c_t is tip chord, and λ is taper ratio (c_t/c_r). The taper ratio influences the Oswald efficiency factor by affecting spanwise lift distribution.

4.1.5 High-Lift Devices

Plain flaps serve as simple high-lift devices that increase an airfoil's effective camber when deflected. This deflection generates a higher $C_{L,max}$, essential for meeting the demanding take-off and landing requirements established by regulatory margins (Sadraey, 2013).

4.2 Electric Aircraft Range

4.2.1 Electric Propulsion Range Equation

The range for an electric aircraft, from the Breguet range equation, is expressed as:

$$Range = \frac{\eta}{g} \cdot \frac{E}{m_{battery}} \cdot \frac{m_{battery}}{m_{aircraft}} \cdot \frac{L}{D} \quad [\text{Eqn. 13}]$$

where η is the total powertrain efficiency, $E/m_{battery}$ is the battery's specific energy (J/kg), g is gravitational acceleration, and L/D is the lift-to-drag ratio (Hepperle, 2012).

4.3 Compressible Flow and Critical Mach Number

As flow velocity increases to greater than Mach 0.3, compressibility effects become significant and must be accounted for in aerodynamic analysis. The pressure coefficient (C_p) is corrected for subsonic compressible flow using the Prandtl-Glauert rule, where $C_{p,0}$ is the incompressible pressure coefficient (Anderson, 2016). As Mach number increases, this correction predicts the strengthening of pressure peaks over the airfoil surface

$$C_p = \frac{C_{p,0}}{\sqrt{1 - M^2}} \quad [\text{Eqn. 14}]$$

The Critical Mach Number (M_{cr}) is defined as the freestream Mach number at which the local flow at some point on the airfoil first reaches sonic velocity (Mach 1). It is determined by finding the intersection between the airfoil's compressibility corrected C_p and the critical pressure coefficient ($C_{p,cr}$) required for local sonic flow, given by the isentropic relation:

$$C_{p,cr} = \frac{2}{\gamma M^2} \left[\left(\frac{2 + (\gamma - 1)M^2}{\gamma + 1} \right)^{\frac{\gamma}{\gamma - 1}} - 1 \right] \quad [\text{Eqn. 15}]$$

Beyond M_{cr} , shockwaves form on the airfoil surface, leading to a rapid increase in wave drag and potential control issues known as transonic drag rise (Bertin & Cummings, 2014).

5. Overall Methodology

This report employed a structured, sequential computational methodology to design and analyse the wing and emergency parachute system for a four-seat electric aircraft. The process was divided into three distinct parts (A, B, and C), each building upon the results of the previous one to ensure a coherent and integrated final design. The overall workflow is visualised in Figure 5.1 below.

Part A: Wing Design and Optimisation

The core of the project was the wing design, executed as a multi-stage optimisation process.

1. Foundation (Step A1): The aircraft's mission profile was established by calculating the reference cruise conditions: a Reynolds number of 6.06×10^6 , a Mach number of 0.25, and a target lift coefficient (C_L) of 0.34.
2. 2D Airfoil Selection (Steps A2 & A3): A parametric analysis of NACA 4-digit airfoils was conducted using XFOIL, automated via a custom Python script. A two-stage selection framework was then applied:
 - *Stage 1 (Mandatory Filters)*: Airfoils were filtered to ensure a minimum operational $C_L > 1.2$ and a practical pitching moment $|C_m| \leq 0.1$.
 - *Stage 2 (Weighted Optimisation)*: Qualified airfoils were ranked using a weighted decision matrix mainly prioritising L/D (40%), $C_{d,min}$ (20%), and C_m (20%).
3. 3D Wing Optimisation (Step A4): The 2D airfoil was analysed in a 3D environment using XFLR5's Vortex Lattice Method (VLM). The planform was optimised in two phases:
 - *Aspect Ratio (AR) Screening*: AR values from 7 to 10 were evaluated, with AR = 10 chosen for its superior L/D performance.
 - *Taper Ratio (λ) Refinement*: With AR fixed at 10, taper ratios from 0.65 to 0.80 were assessed. $\lambda = 0.7$ was selected for its optimal Oswald efficiency ($e = 0.929$) and favourable stall characteristics.
4. Flap Design (Step A5): A plain flap (25% chord, 50% span) was integrated. Six deflection angles (0° to 30°) were analysed. A Python automation script evaluated each configuration against a weighted multi-criteria matrix (lift achievement, L/D, operating angle), identifying 5° deflection as optimal for both take-off and landing.

Part B: Emergency Parachute Design

The parachute system was designed using Newton's laws applied to a two-stage deployment sequence:

1. Stage 1 (Reefed Deceleration): The parachute area required to decelerate the aircraft from cruise speed at a maximum of 1g was calculated using Newton's Second Law, equating drag force to the deceleration force.

2. *Stage 2 (Full Descent)*: The parachute area required to achieve a safe vertical descent velocity of 28 km/h at sea level was calculated from vertical force equilibrium (drag = weight). The final design was validated by comparing the calculated mass-to-area ratio with that of a certified system (Cessna 172 BRS), confirming its realism.

Part C: High-Mach Suitability Analysis

The low-speed wing's performance at high Mach numbers was assessed analytically.

1. The critical Mach number (M_{cr}) was determined by finding the intersection between the airfoil's Prandtl-Glauert corrected pressure coefficient and the critical pressure coefficient required for sonic flow.
2. This analysis provided a clear limit to the designed wing's operational envelope and informed recommendations for high-speed modifications, such as reduced thickness and wing sweep.

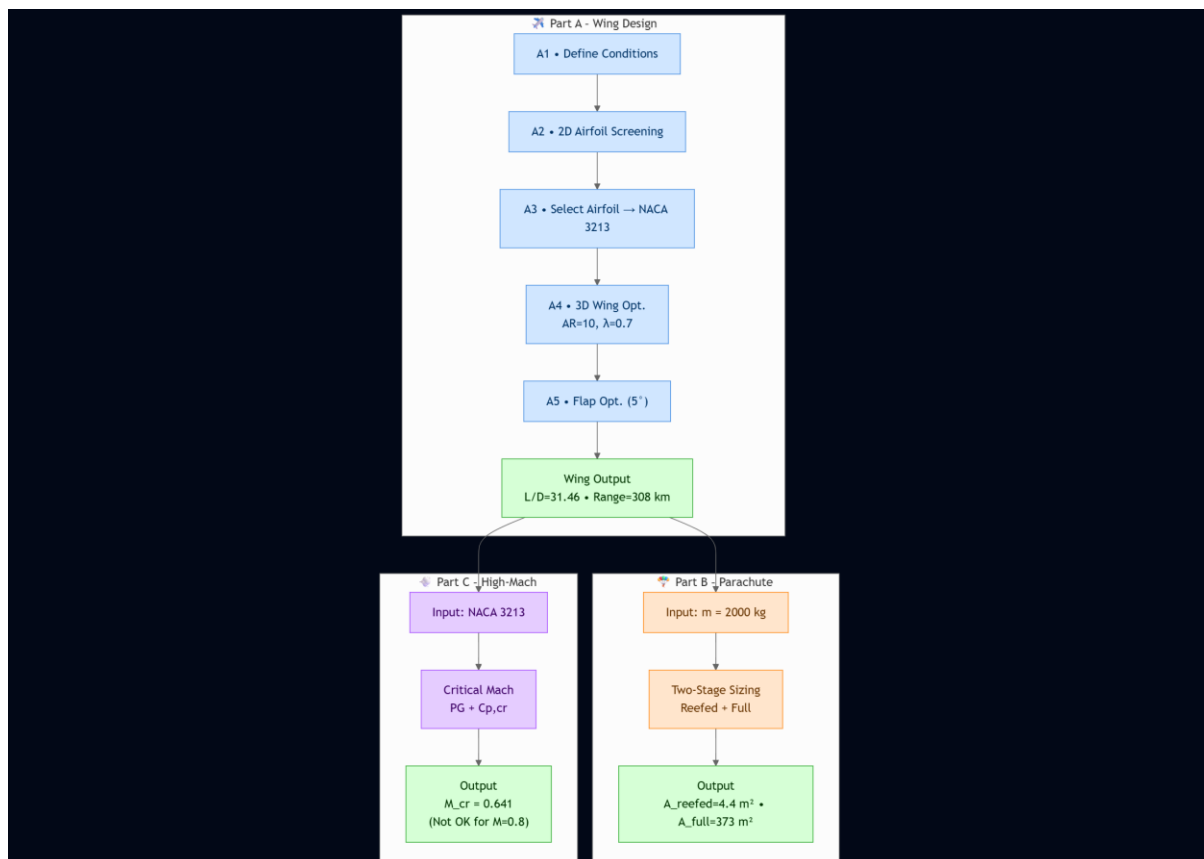


Figure 5.1: Overarching Methodology Workflow for Electric Aircraft Wing Design and Parachute Sizing

6. Part A: Wing Design and Optimisation

Part A: Methodology

The wing design process used a systematic computational approach to identify an optimal airfoil and flap configuration for the specified electric aircraft's mission profile. Aerodynamic analyses were conducted using XFOIL via a custom Python automation script adapted from previous work (Wijayaratne, 2025b). See appendix section A1 for full code.

Step A1: Computing Environmental and Geometric Parameters

The flight conditions for optimisation were established from given cruise conditions of 3,000 m altitude and 83.33 m/s (300 km/h) velocity. From that, atmospheric properties were derived using the International Standard Atmosphere tables (Figure 6.2), yielding: air density (ρ) of 0.91 kg/m³, dynamic viscosity (μ) of 1.86×10^{-5} Pa·s, and speed of sound (a) = 328.58 m/s. Additionally, based on the comparable light aircraft geometry of Cessna 172 Skyhawk as shown in figure 6.1, a representative mean aerodynamic chord of 1.49 m was chosen (Mousa et al., 2022). The corresponding non-dimensional parameters were calculated as:

- Eqn. 4: $Re = \frac{\rho V c}{\mu} = \frac{0.91 \times 83.33 \times 1.49}{1.86 \times 10^{-5}} = 6.06 \times 10^6$
- Eqn. 3: $M = \frac{V}{a} = \frac{83.33}{328.58} = 0.25$



Figure 6.1: Reference Aircraft Geometry - Cessna 172 Used for Mean Aerodynamic Chord. Source: (iStock, 2006)

ALTITUDE (feet)	TEMP (°C)	PRESSURE			PRESSURE RATIO $\sigma = P/P_0$	DENSITY $\rho = \rho_0 \sigma$	Speed of sound (ft/s)	ALTITUDE (meters)
		inHg	PSI	hPa				
40 000	-66.5	188	2.72	5.54	0.1881	0.2462	573	12 192
38 000	-66.5	197	2.88	5.81	0.1962	0.2582	573	11 887
36 000	-66.5	206	2.99	6.10	0.2038	0.2710	573	11 582
34 000	-66.5	217	3.14	6.40	0.2138	0.2844	573	11 278
32 000	-66.3	227	3.30	6.71	0.2263	0.2981	573	10 973
30 000	-64.3	238	3.46	7.04	0.2383	0.3099	576	10 668
28 000	-62.4	250	3.63	7.38	0.2487	0.3220	579	10 363
26 000	-60.4	262	3.80	7.74	0.2586	0.3345	581	10 058
24 000	-58.4	274	3.98	8.11	0.2709	0.3473	584	9 754
22 000	-56.4	287	4.17	8.49	0.2837	0.3605	586	9 449
20 000	-44.4	301	4.36	8.89	0.2970	0.3741	589	9 144
18 000	-42.5	315	4.57	9.30	0.3107	0.3881	591	8 839
16 000	-40.5	329	4.78	9.73	0.3250	0.4025	594	8 534
14 000	-38.5	344	4.99	10.17	0.3398	0.4173	597	8 230
12 000	-36.5	360	5.22	10.63	0.3552	0.4326	599	7 926
10 000	-34.5	376	5.45	11.10	0.3711	0.4481	602	7 622
8 000	-32.5	393	5.70	11.60	0.3876	0.4642	604	7 318
6 000	-30.6	410	5.95	12.11	0.4046	0.4806	607	7 014
4 000	-28.6	428	6.21	12.64	0.4223	0.4976	609	6 709
2 000	-26.6	446	6.47	13.18	0.4406	0.5150	611	6 404
0 000	-24.6	466	6.75	13.75	0.4595	0.5329	614	6 098
10 000	-22.6	485	7.04	14.34	0.4781	0.5511	616	5 793
8 000	-20.7	506	7.34	14.94	0.4964	0.5699	619	5 488
6 000	-18.7	527	7.65	15.57	0.5203	0.5892	621	5 182
4 000	-16.7	549	7.97	16.22	0.5420	0.6090	624	4 877
2 000	-14.7	572	8.29	16.89	0.5643	0.6292	626	4 572
0 000	-12.7	596	8.63	17.58	0.5875	0.6500	628	4 267
10 000	-10.8	619	8.99	18.29	0.6113	0.6713	631	3 962
8 000	-8.8	644	9.35	19.03	0.6360	0.6932	633	3 658
6 000	-6.8	670	9.72	19.79	0.6614	0.7156	636	3 353
4 000	-4.8	697	10.10	20.58	0.6877	0.7385	638	3 048
2 000	-2.8	724	10.51	21.38	0.7148	0.7620	640	2 743
0 000	-0.8	753	10.92	22.22	0.7429	0.7860	643	2 438
10 000	+1.1	782	11.34	23.09	0.7716	0.8106	645	2 134
8 000	+3.1	812	11.78	23.98	0.8014	0.8359	647	1 829
6 000	+5.1	843	12.23	24.90	0.8320	0.8617	650	1 524
4 000	+7.1	875	12.69	25.84	0.8637	0.8881	652	1 219
2 000	+9.1	908	13.17	26.82	0.8962	0.9151	654	914
0 000	+11.0	942	13.67	27.82	0.9298	0.9428	656	610
1 000	+13.0	977	14.17	28.86	0.9644	0.9711	659	305

Figure 6.2: ISA Tables. Source: (ISO, 1975)

Step A2: Parametric Airfoil Analysis

A systematic parametric analysis of NACA 4-digit airfoils was performed over a design space defined by camber (2–6% chord), camber location (20–60% chord), thickness (12–18% chord), and angle of attack (-4° to 18° in 2° increments). These bounds were selected based on established aerodynamic principles and prior findings reported by Wijayaratne (2025a), which reported that camber near 6% produces “*dramatic and highly beneficial*” improvement in aerodynamic efficiency. Furthermore, the report stated that camber locations with a “central location” around 20% to 60% generally improved airfoil efficiency. Finally, a thickness range

of 12% to 18% was chosen to ensure structural integrity by avoiding thin airfoils while mitigating large drag penalties associated with very thick sections. The final parameter sets that were entered into the sweeper are shown in Figure 6.3 below.

```
# ----- user-configurable -----
RE = 6064796
MACH = 0.2536
ALPHA_START = -4.0
ALPHA_END = 18.0
ALPHA_STEP = 2.0
ITER = 80
TIMEOUT = 120
NUM_WORKERS = min(4, cpu_count())

XF_PATH = shutil.which("xfoil") or shutil.which("xfoil.exe")

OUTDIR = Path("xfoil_comprehensive_outputs")
POLAR_DIR = OUTDIR / "polars"
RESULTS_CSV = OUTDIR / "airfoil_data.csv"
FAILED_FILE = OUTDIR / "failed_runs.txt"

# Minimum number of data points required (changed from requiring all 12)
MIN_DATA_POINTS = 10

# Target angles from your experimental data
TARGET_ANGLES = [-4.0, -2.0, 0.0, 2.0, 4.0, 6.0, 8.0, 10.0, 12.0, 14.0, 16.0, 18.0]

if XF_PATH is None:
    raise SystemExit("xfoil executable not found in PATH.")

M_RANGE = range(2, 7) # 2 to 6
P_RANGE = range(2, 7) # 2 to 6
TT_RANGE = range(12, 19) # 12 to 18 - This ensures structural viability while avoiding the severe drag penalties of very thick airfoils
# -----
```

Figure 6.3: Final Input Parameters for NACA Airfoil Analysis

Step A3: Ideal Airfoil Selection Framework

Following the parametric sweep, a custom python script implemented a structured two-stage selection process to find the optimal NACA airfoil (see Appendix A2 for complete code).

Stage 1: Evaluation against mandatory design requirements:

Table 1: Lift Coefficient required for each flight phase

Phase	Velocity [m/s]	Operating C_l	$C_{l,max}$ required	Stall margin
Take-off	46.67	0.818	1.177	1.2×
Cruise	83.33	0.345	-	-
Landing	40.28	1.097	<u>1.863</u>	1.3×

Calculations:

- C_l values were found using lift equation [Eqn. 5]
- Take-off: $\rho = 1.225 \text{ kg/m}^3$, $V_{stall} = 46.67/1.2 = 38.89 \text{ m/s}$ [Eqn. 7]
- Landing: $\rho = 1.225 \text{ kg/m}^3$, $V_{stall} = 40.28/1.3 = 30.98 \text{ m/s}$ [Eqn. 8]
- $C_{L,max}$ values were found by using lift equation on stall velocities.

The maximum lift C_L requirement is roughly 1.9. Accounting for a flap contribution of $\Delta C_L \approx 0.7$ (Mitchell, 1950), the clean airfoil was required to achieve a $C_L > 1.2$ within operational AoA ($0^\circ \leq \alpha \leq 8^\circ$). This ensures accordance with Federal Aviation Regulations.

A pitching moment constraint of $|C_m| \leq 0.1$ was also applied due to initial analysis indicating the NACA 6612 as theoretically optimal due to its high L/D ratio (210.60) and $C_{l,max}$ (2.05), consistent with Wijayaratne (2025a). However, its pitching moment ($C_m = 0.1987$) was impractical, as the aircraft would require an oversized tail and incur significant trim drag. The constraint, therefore, ensured consistency with conventional stability design practices (Scholz, 2021) and accounted for pitching-moment effects not considered by Wijayaratne (2025a).

Stage 2: Weighted multi-criteria Optimisation:

Airfoils passing the initial filter were ranked using a weighted scoring algorithm (Figure 6.4) with weights to reflect the prioritised parameters for the electric aircraft's performance.

- *Maximum Lift-to-Drag Ratio (40%)*: Prioritised for its direct influence on aircraft range and energy efficiency, following [Eqn. 13].
- *Minimum Drag Coefficient (20%)*: Critical for cruise efficiency, representing the airfoil's parasitic drag characteristics independent of lift-induced effects.
- *Pitching Moment Coefficient (20%)*: Evaluated as the mean absolute value of C_m in the pre-stall region ($\alpha \leq 12^\circ$) with values near zero preferred to ensure aircraft stability.
- *Maximum Lift Coefficient (10%)*: Reduced weighting as Stage 1 filtering guaranteed adequate lift performance, but it remains relevant for stall behaviour and safety margins.
- *Stall Characteristics (10%)*: Quantified as $(C_{L@18^\circ}/C_{L,max})$, with higher values indicating more gentle, predictable and gradual stall progression.

The optimisation algorithm employed min-max normalization to score each parameter relative to the best performer, with weighted aggregation determining the final ranking. As shown by Figure 6.5, this algorithm identified the NACA 3213 airfoil (Figure 6.6) as the optimal airfoil, balancing exceptional cruise efficiency with practical stability requirements.

```

1 ideal_NACA_finder.py > ...
2 import pandas as pd
3 import numpy as np
4
5 # User-configurable scoring and filter parameters
6
7 # STAGE 1: HARD REQUIREMENTS (Filtering)
8 CL_FILTER_THRESHOLD = 0.1 # Maximum acceptable pitching moment (|Cm|)
9 CL_MIN_REQUIREMENT = 1.2 # Minimum CL required at reasonable AoA for landing/takeoff (0-8°)
10
11 # STAGE 2: OPTIMIZATION WEIGHTS (Updated priorities)
12 METRIC_WEIGHTS = {
13     'L/D': 40, # Efficiency (L/D ratio) - CRITICAL for electric range
14     'CL_max': 10, # Maximum lift coefficient - minimal weight since requirement filtered
15     'Cm': 20, # Pitching moment stability - increased importance
16     'Cd_min': 20, # Minimum drag - important for cruise
17     'Stall': 10 # Stall behavior - minimal since most airfoils meet basic safety
18 }
19
20 # Function to gather and compute aerodynamic metrics from the csv
21 > def analyze_airfoils(csv_file): ...
22
23 # Function to calculate weighted scores for airfoils
24 > def calculate_scores(airfoil_results): ...
25
26 # Function to print best of the ideal airfoils
27 > def main(): ...
28
29 if __name__ == "__main__":
30     main()

```

Figure 6.4: Structure of Main Functions in the Multi-Criteria Optimisation Algorithm

```

PS C:\Users\Ashen\OneDrive\Code\Projects\Python\Projects\Aero_report3> python ideal_NACA_finder.py
===== AIRFOIL SELECTION WITH TWO-STAGE FILTERING =====
Stage 1: Hard Requirements | Stage 2: Weighted Optimization
Analyzing 170 unique airfoils...
Total airfoils analyzed: 170
Rejected - CL < 0.1: 75
Rejected - CL_operational < 1.2: 32
Airfoils passing Stage 1 filter: 64

===== TOP AIRFOILS RANKED BY WEIGHTED METRIC =====

#1: 3213
Total Score: 89.98
Key Performance Metrics:
- L/D_max: 36.84 (value: 172.38)
- CL_max: 7.57 (value: 1.84)
- Max CL: 1.94 (Operational CL: 1.28)
- Cm: 18.76 (avg |Cm|: 0.4518)
- Cd_min: 35.49 (value: 0.4003)
- Stall: 9.71 (quality: 0.971)

#2: 3214
Total Score: 89.47
Key Performance Metrics:
- L/D_max: 36.87 (value: 172.42)
- CL_max: 7.56 (value: 1.84)
- Max CL: 1.94 (Operational CL: 1.28)
- Cm: 18.85 (avg |Cm|: 0.4518)
- Cd_min: 35.49 (value: 0.4003)
- Stall: 9.68 (quality: 0.968)

#3: 3212
Total Score: 89.46
Key Performance Metrics:
- L/D_max: 36.33 (value: 168.98)
- CL_max: 7.56 (value: 1.84)
- Max CL: 1.89 (Operational CL: 1.27)
- Cm: 18.58 (avg |Cm|: 0.4517)
- Cd_min: 37.26 (value: 0.4003)
- Stall: 9.68 (quality: 0.968)

Thank you for using the Ideal NACA Airfoil Finder! - Ashen Wijayarathne
PS C:\Users\Ashen\OneDrive\Code\Projects\Python\Projects\Aero_report3>

```

Figure 6.5: Optimization Algorithm Terminal Output – Top Performing Airfoils

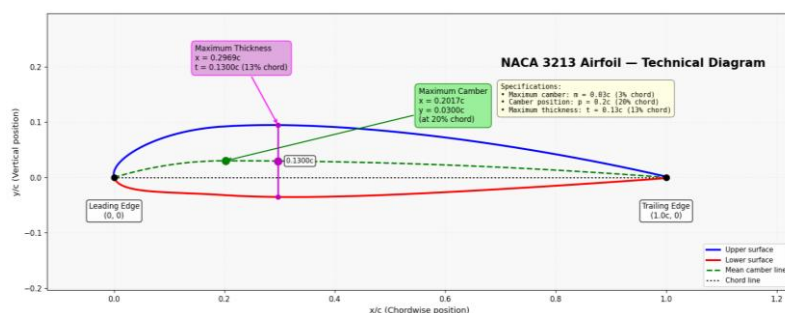


Figure 6.6: Detailed section geometry of NACA 3213 airfoil with annotated design parameters. Source: Author's own work.

Step A4: 3D Wing Optimisation using XFLR5

A two-phase method was used to optimise the wing planform. First, the aspect ratio (AR), the main driver of induced drag ($C_{Di} \propto 1/AR$) was optimised. Then, the taper ratio (λ) was adjusted for secondary refinement. This method captures AR's dominant aerodynamic effect before fine-tuning λ , improving overall performance and computational efficiency.

Phase 1: Aspect Ratio Screening

a) Geometric Definition and Computational Setup:

Four aspect ratios ($AR = 7, 8, 9, 10$) within the conventional range for light general aviation aircraft (Leishman, 2023) were evaluated, maintain a constant taper ratio of $\lambda = 0.75$ as an aerodynamically balanced baseline. Wing geometries calculated for each configuration using (Eqns. 10, 11 and 12) and systematically documented in Tables 2 – 5. Each wing configuration was implemented in XFLR5 using these calculated parameters. The XFLR5 wing definitions can be visualised in Figures 6.7 to 6.10.

b) XFLR5 Analysis Parameters:

- Analysis Method: Vortex Lattice Method (VLM2 - Ring Vortex)
- Flight conditions:
 - Cruise velocity: 83.33 m/s (300 km/h)
 - Altitude: 3,000 m ($\rho = 0.91 \text{ kg/m}^3$, $\mu = 1.86 \times 10^{-5} \text{ Pa}\cdot\text{s}$)
 - Angle of attack sweep: -3° to 12° in 0.5° increments

c) Aircraft Reference Parameters:

- Aircraft mass: 2000 kg
- Planform Area = 18 m^2

d) Evaluation Metrics:

The following metrics were extracted from the data for comparative analysis to find the optimal aspect ratio:

- Maximum lift-to-drag ratio $(L/D)_{\max}$
- L/D at cruise condition ($C_L = 0.34$) at $(0^\circ < \alpha < 4^\circ)$
- Cruise angle of attack
- Drag coefficient at cruise
- Maximum lift coefficient

This phase of the methodology isolates the dominant effect of aspect ratio on induced drag, which scales inversely with AR ($C_{Di} \propto 1/AR$). Testing multiple AR values with fixed λ enables clear identification of efficiency trends while maintaining computational efficiency.

Table 2: XFLR5 Inputs for AR = 7 and Fixed Taper = 0.75

Aspect Ratio (AR)	Taper Ratio (λ)
7	0.75
Wingspan (b)	11.22497216
Root Chord	1.832648516
Tip Chord	1.374486387

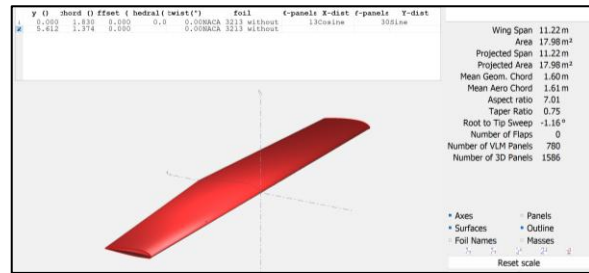


Figure 6.7: XFLR5 Wing Definition for NACA 3213 with AR = 7 and Taper = 0.75

Table 3: XFLR5 Inputs for AR = 8 and Fixed Taper = 0.75

Aspect Ratio (AR)	Taper Ratio (λ)
8	0.75
Wingspan (b)	12
Root Chord	1.714285714
Tip Chord	1.285714286

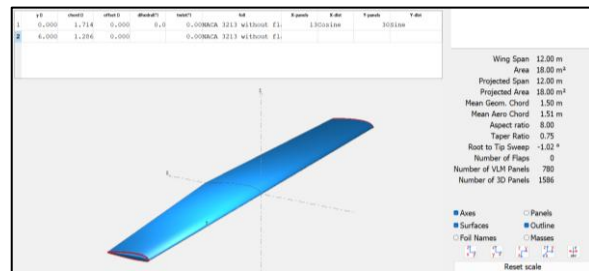


Figure 6.8: XFLR5 Wing Definition for NACA 3213 with AR = 8 and Taper = 0.75

Table 4: XFLR5 Inputs for AR = 9 and Fixed Taper = 0.75

Aspect Ratio (AR)	Taper Ratio (λ)
9	0.75
Wingspan (b)	12.72792206
Root Chord	1.616244071
Tip Chord	1.212183053

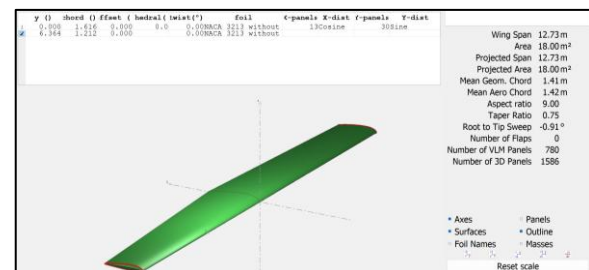


Figure 6.9: XFLR5 Wing Definition for NACA 3213 with AR = 9 and Taper = 0.75

Table 5: XFLR5 Inputs for AR = 10 and Fixed Taper = 0.75

Aspect Ratio (AR)	Taper Ratio (λ)
10	0.75
Wingspan (b)	13.41640786
Root Chord	1.533303756
Tip Chord	1.149977817

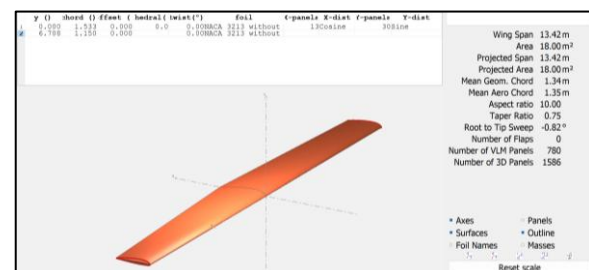


Figure 6.10: XFLR5 Wing Definition for NACA 3213 with AR = 10 and Taper = 0.75

Phase 2: Taper Ratio Refinement

a) Geometric Definition and Computational Setup:

Based on the Phase 1 analysis, the aspect ratio of $AR = 10$ was chosen as the optimal configuration (details discussed further in the Results section). Four taper ratios were then evaluated across the established optimal range for conventional light aircraft: $\lambda = 0.65, 0.70, 0.75$, and 0.80 (Özgen, 2015). The geometric parameters for each configuration were systematically documented in Tables 6-9. Each wing configuration was implemented in XFLR5 using these calculated parameters. The XFLR5 wing definitions can be visualised in Figures 6.11 to 6.14.

b) XFLR5 and Aircraft Reference Parameters:

Identical computational parameters and flight conditions from Phase 1 were rigorously maintained throughout Phase 2 to ensure consistent comparison across all configurations. This consistency allowed direct correlation of performance variations due to taper ratio effects.

c) Enhanced Evaluation Framework:

Beyond the Phase 1 metrics, Phase two included additional analyses of :

- Visual analysis of spanwise lift distribution curves against theoretical elliptical distribution
- Oswald efficiency factor (e) calculated using the relationship in Eqn. 9
- Induced and profile drag component analysis.
- Stall progression through high-angle-of-attack analysis (up to 12°)

Theoretical Rationale for analysis method:

Taper ratio primarily controls the Oswald efficiency factor in the induced drag equation $[C_{Di} = C_L^2 / (\pi \cdot AR \cdot e)]$. The selected range (0.65 - 0.80) represents the established optimal zone for achieving near-elliptical lift distribution while maintaining favourable stall behaviour and structural efficiency. This range balances aerodynamic performance with practical manufacturing constraints typical of light aircraft production.

Table 6: XFLR5 Inputs for Fixed
AR = 10 and Taper = 0.65

Aspect Ratio (AR)	Taper Ratio (λ)
10	0.65
Wingspan (b)	13.41640786
Root Chord	1.626231256
Tip Chord	1.057050317

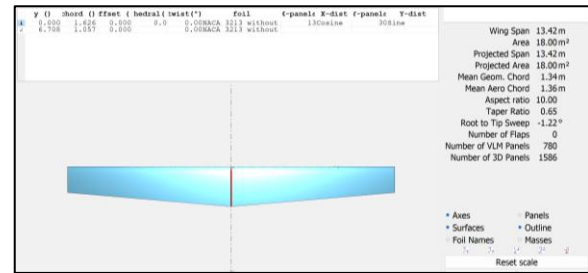


Figure 6.11: XFLR5 Wing Definition for NACA 3213 with AR = 10 and Taper = 0.65

Table 7: XFLR5 Inputs for Fixed
AR = 10 and Taper = 0.7

Aspect Ratio (AR)	Taper Ratio (λ)
10	0.7
Wingspan (b)	13.41640786
Root Chord	1.578400925
Tip Chord	1.104880648

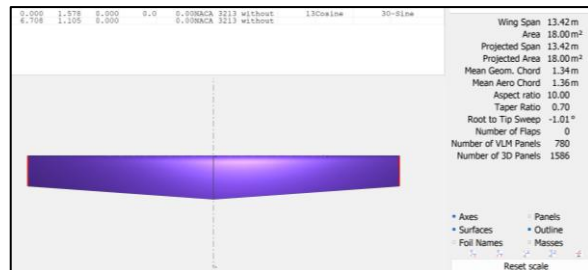


Figure 6.12: XFLR5 Wing Definition for NACA 3213 with AR = 10 and Taper = 0.7

Table 8: XFLR5 Inputs for Fixed
AR = 10 and Taper = 0.75

Aspect Ratio (AR)	Taper Ratio (λ)
10	0.75
Wingspan (b)	13.41640786
Root Chord	1.533303756
Tip Chord	1.149977817

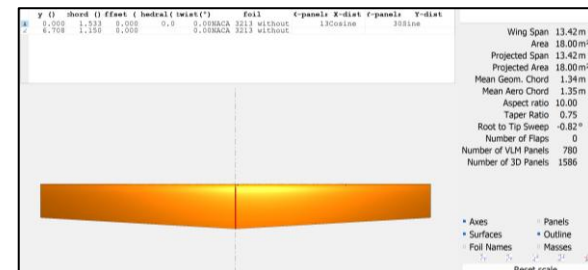


Figure 6.13: XFLR5 Wing Definition for NACA 3213 with AR = 10 and Taper = 0.75

Table 9: XFLR5 Inputs for Fixed
AR = 10 and Taper = 0.8

Aspect Ratio (AR)	Taper Ratio (λ)
10	0.8
Wingspan (b)	13.41640786
Root Chord	1.490711985
Tip Chord	1.192569588

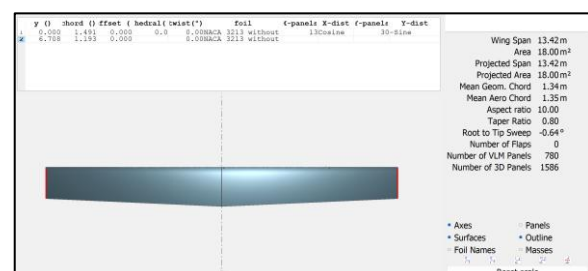


Figure 6.14: XFLR5 Wing Definition for NACA 3213 with AR = 10 and Taper = 0.8

Step A5: 3D Flap Configuration Optimisation using XFLR5

Following the selection of the NACA 3213 airfoil with an aspect ratio of 10 and taper ratio of 0.7, a systematic computational approach was employed to determine the optimal plain flap configuration for both take-off and landing phases.

Design Requirements and Constraints:

The flap configuration analysis was constrained by both regulatory requirements and operational practicality. From the design requirements established in Step A3, the following lift coefficient thresholds were imposed:

1. Take-off: $C_L \geq 0.818$ @ 46.67 m/s, 1.2× stall margin (Federal Aviation Regulations, 2025a).
2. Landing: $C_L \geq 1.097$ @ 40.28 m/s, 1.3× stall margin (Federal Aviation Regulations, 2025b).
3. Operational envelope: Preferred α range 0 - 8° for visibility and control

Analysis Method:

Six plain flap configurations (0° to 30° in 5° steps) with 25% chord and 50% span coverage were analysed in XFLR5. Two example airfoil definitions are shown in Figure 6.15 and 6.16. Complete polars were generated for each configuration across the AoA range from -9° to +18° in 0.5° increments at $\rho = 1.225 \text{ kg/m}^3$. The analysis used XFLR5's ring vortex lattice method with 30 panels per wing section, to capture the 3-D flow field and induced drag effects. Each flap configuration was assessed using a weighted composite as shown in Table 10 below:

Table 10: Multi-Criteria Weighting Framework for Flap Configuration Evaluation

Metric	Weight	Rationale
C_L Achievement	35%	Primary safety requirement; bonuses for 5-15% margins
Aerodynamic Efficiency	25%	Direct impact on battery endurance and range; normalized against $(L/D)_{max} = 35$
Operating Angle	20%	Critical for pilot visibility and control; strong penalties above 8° α
Stall Margin	15%	Regulatory compliance; bonuses for adequate but not excessive margins
Mechanical Simplicity	5%	Reliability and structural considerations; lower deflections preferred

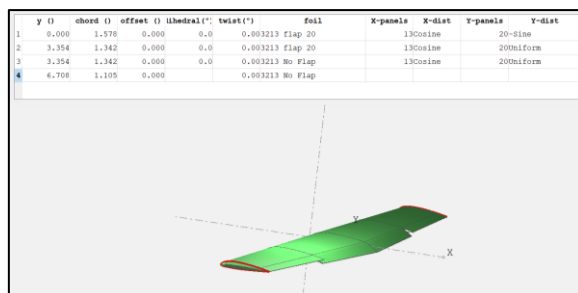


Figure 6.15: Wing definition on XFLR5 for NACA 3213 at deflection angle = 20°

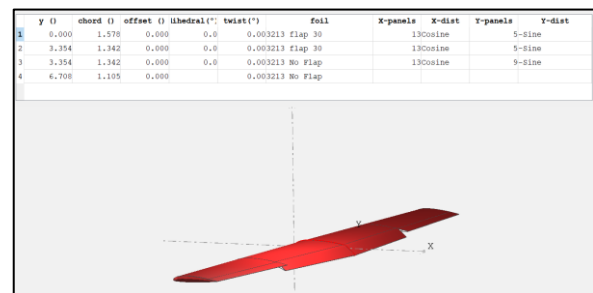


Figure 6.16: Wing definition on XFLR5 for NACA 3213 at deflection angle = 30°

Due to the large volume of data generated by XFLR5 (14 complete polar datasets comprising over 450 individual operating points), a custom Python-based analysis framework was developed to automate the evaluation process. The analyser employed a two-stage optimization approach:

For each flap configuration, the algorithm determined the optimal angle of attack through a hierarchical filtering process illustrated by the *find_optimal_operating_point()* function in Figure 6.17 which consists of stages:

- This approach recognized that different flap deflections naturally favour different operating angles, avoiding the artificial constraint of fixed-angle comparison that would disadvantage certain configurations.

Each configuration's optimal operating point was evaluated against the five weighted criteria described in Table 10 above. The composite scores were then computed and ranked automatically, with the highest-scoring configuration identified as the recommended design for each flight phase. This is demonstrated by the logic shown in Figure 6.18

The complete analysis pipeline, from XFLR5 data ingestion through final configuration recommendation was executed in under 5 seconds, enabling rapid iteration during the design process and ensuring that all configurations were evaluated under identical criteria.

Figure 6.17: Logic breakdown of stage one of the flap optimiser code

```

104         # Cl achievement scoring (35%)
105         if meets_cl_requirement: ...
106     else:
107         scores['cl_requirement'] = max(0, min(120, scores['cl_requirement']))
108
109     # Efficiency scoring (25%)
110     max_realistic_LD = 35
111     scores['efficiency'] = 100 * min(optimal_op['LD'] / max_realistic_LD, 1.2)
112
113     # Alpha quality scoring (20%)
114     if optimal_op['in_preferred_range']: ...
115     elif optimal_op['alpha'] < req['preferred_alpha_min']: ...
116     else: ...
117
118     # Stall margin scoring (15%)
119     if safe_stall_margin: ...
120     else:
121         scores['stall_margin'] = min(120, scores['stall_margin'])
122
123     # Simplicity scoring (5%)
124     scores['simplicity'] = 100 * (1 - flap_angle / 40)
125
126     # Calculate total weighted score
127     weights = {
128
129         total_score = sum(scores[k] * weights[k] for k in scores)
130
131     return f

```

Figure 6.18: Logic illustration of the weighted matrix being applied to the different XFLR5 outputs

A logic breakdown of the code different functions used in the code is available in figure 6.19 illustrating the steps carried out to determine the best airfoil for step A5, as described above. The descriptions of what the code does is shown in the comment code in green. The full code is accessible via appendix A4.

```

Flap_optimiser.py > ...
1 import os
2 import pandas as pd
3 import numpy as np
4 from pathlib import Path
5 import matplotlib.pyplot as plt
6
7 class ImprovedFlapAnalyzer:
8     # Function to initialize the analyzer with data folder and requirements
9 > 1 def __init__(self, data_folder='Total_Data'):...
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42 > 2 def parse_xflr5_csv(self, filepath):...
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80 > 3 def extract_file_info(self, filename):...
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111 > 4 def find_optimal_operating_point(self, df, req, phase):...
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154 > 5 def analyze_configuration(self, df, flap_angle, phase):...
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239 > 6 def process_all_files(self):...
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293 > 7 def generate_summary(self):...
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329 > 8 def create_individual_plots(self, df):...
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405 > 9 def run(self):...
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Figure 6.19: Logic Architecture of the Python-Based Flap Configuration Optimisation Framework for Step A5

Part A: Results

Results - Step A3: Ideal Airfoil Selection

The two-stage airfoil selection method produced clear, objective results. Without this structure, the alternatives shown in Table 11 might have been chosen despite each failing at least one a key design requirement. In Stage 1, the initial set of 171 airfoils were filtered by removing those exceeding the pitching-moment limit ($|C_m| > 0.1$) and those below the minimum operational lift coefficient ($C_{l,operational} < 1.2$). This eliminated 75 airfoils for excessive pitching moment and 32 for insufficient lift, leaving 64 candidates that met the essential performance and safety criteria. The weighted optimisation then identified NACA 3123 as the optimal choice as it balances high aerodynamic efficiency with the practical stability and lift requirements.

Table 11: Performance Comparison of Top Airfoil Candidates from Weighted Optimisation

Metric	2212 (No CL filter applied)	6612 (No Cm filter applied)	3213 (Both filters applied)
$(L/D)_{max}$	149.12	210.60	172.30
$C_{l,max}$	1.80	2.05	1.84
Operational $C_{l,max}$	1.16	1.58	1.28
Avg. $ C_m $	0.0335	0.1987	0.0512
$C_{d,min}$	0.00544	0.00435	0.00586
Stall Quality	0.974	1.000 (no stall)	0.971
Status	Failed Lift Requirement	Failed Stability Requirement	Selected

Results - Step A4: 3D Wing Optimisation

Phase 1: Aspect Ratio Screening Results

The systematic evaluation of four aspect ratio configurations with a fixed taper ratio ($\lambda = 0.75$) revealed clear performance trends directly attributable to induced drag reduction. The performance metrics extracted from XFLR5 analyses are summarised in Table 12.

Table 12: Key Performance metrics for AR configurations ($7 < AR < 10$) at fixed λ

Aspect Ratio (AR)	$(L/D)_{max}$	$(L/D)_{cruise}$	Improvement Cruise L/D vs Baseline	AoA_{cruise} (°)	CD_{cruise}	Cm_{cruise}
7	28.28	27.96	Baseline	1.69	0.01216	-0.1423
8	30.28	29.36	5%	1.55	0.01158	-0.1428
9	32.38	30.5	9%	1.44	0.01115	-0.1433
10	34.3	31.45	12%	1.35	0.01081	-0.1436

The data demonstrates that increasing aspect ratio yields consistent improvements in aerodynamic efficiency. The AR = 10 configuration provided superior performance across all evaluated metrics, achieving a 12% improvement in cruise L/D compared to the AR=7 baseline. This enhancement was achieved while maintaining practical cruise angles of attack between 1-2°, with the AR=10 wing requiring only 1.35° AoA to generate the required cruise lift coefficient.

Comparative aerodynamic polar plots for all four configurations are presented in Figures 6.20 through 6.23. These visualizations confirm the superior performance of AR = 10 across multiple parameters, including the highest lift coefficients across the angle of attack range, lowest drag coefficients, and highest lift-to-drag ratios at the target cruise condition of $CL = 0.34$.

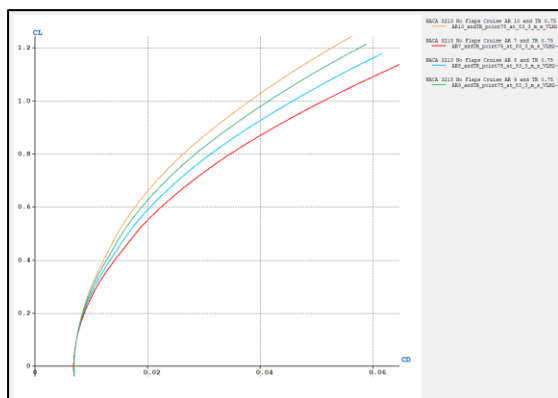


Figure 6.20: C_L vs C_D for NACA 3312 for $\alpha = -4$ to 15 , $\Delta\alpha = 0.5$, $M = 0.25$

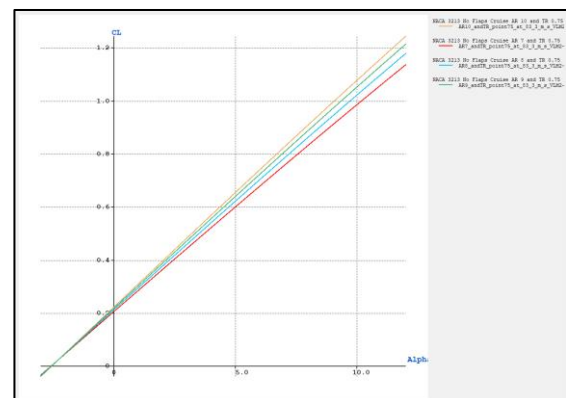


Figure 6.21: C_L vs α for NACA 3312 for $\alpha = -4$ to 15 , $\Delta\alpha = 0.5$, $M = 0.25$

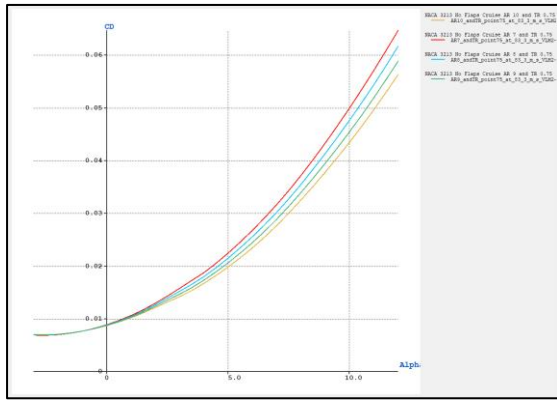


Figure 6.23: C_D vs α for NACA 3312 for $\alpha = -4$ to 15 , $\Delta\alpha = 0.5$, $M = 0.25$

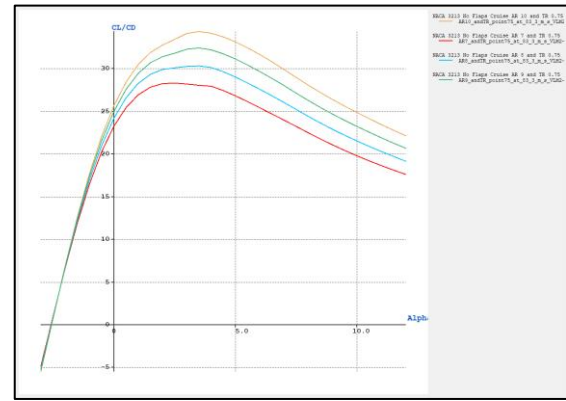


Figure 6.22: C_L vs C_D for NACA 3312 for $\alpha = -4$ to 15 , $\Delta\alpha = 0.5$, $M = 0.25$

Phase 2: Taper Ratio Refinement Results

Following the selection of $AR = 10$, four taper ratios were evaluated to optimize the wing's lift distribution and Oswald efficiency. The performance metrics for these configurations are presented in Table 13.

Table 13: Key Performance metrics for λ configurations ($0.65 < \lambda < 0.8$) at fixed AR

Taper Ratio (λ)	(L/D) max	L/D @ Cruise	Induced Drag Coefficient	Oswald efficiency (e)	$C_{D_{cruise}}$	$C_{m_{cruise}}$
0.65	34.41	31.53	0.00369	0.922	0.01078	-0.1438
0.7	34.32	31.46	0.00371	0.929	0.01081	-0.1436
0.75	34.3	31.45	0.00371	0.923	0.01081	-0.1436
0.8	34.22	31.4	0.00373	0.924	0.01083	-0.1434

The results indicate minimal variation in primary performance metrics across the taper ratio range, with differences of less than 0.5% in cruise L/D. However, the $\lambda = 0.7$ configuration demonstrated the highest Oswald efficiency factor ($e = 0.929$), indicating a more optimal lift distribution that minimizes induced drag.

Figure 6.24 shows nearly identical performance across all taper ratios in standard polar plots. However, analysis of spanwise lift distributions show key differences between taper ratio configurations. Figures 6.25 and its detailed view 6.26, reveal that the $\lambda = 0.65$ configuration exhibited a more pronounced lift reduction at the wing root. This characteristic became more evident at stall, as shown in Figure 6.27 where the $\lambda = 0.7$ configuration maintained a more favourable distribution pattern with better stall progression characteristics.

Based on this comprehensive analysis, the $\lambda = 0.7$ configuration was selected as optimal, providing the best balance of high Oswald efficiency, excellent cruise performance, and favourable stall characteristics.

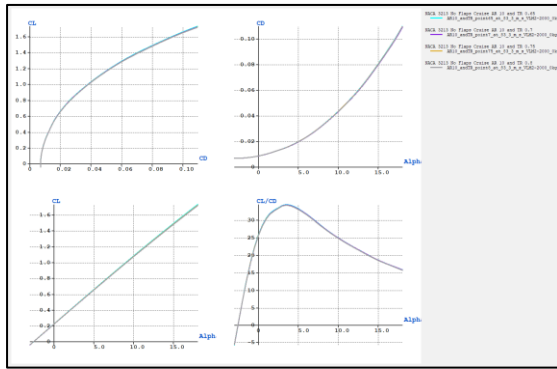


Figure 6.24: Comparative Aerodynamic Polars for Taper Ratio Variations for NACA 3312 for $\alpha = -4$ to 15 , $\Delta\alpha = 0.5$, $M = 0.25$

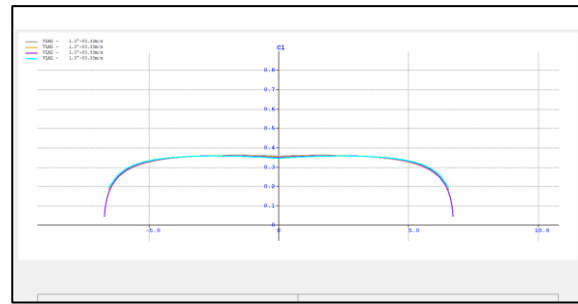


Figure 6.25: Spanwise Lift Distribution at Cruise Condition for Taper Ratio Variations for attack of 1.69 (cruise from table 12)

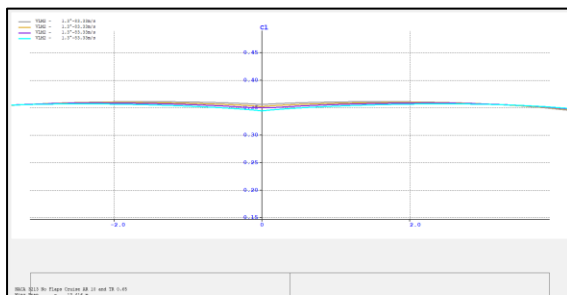


Figure 6.26: Detailed View of Spanwise Lift Distribution at Cruise Condition

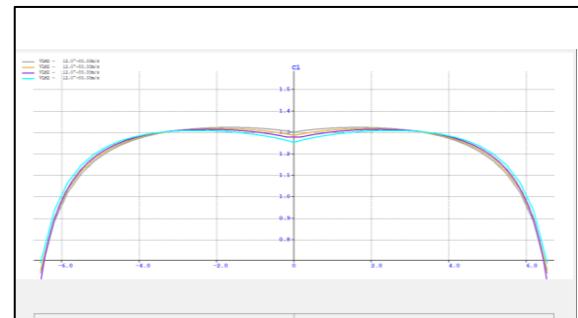


Figure 6.27: Spanwise Lift Distribution at Near-Stall Condition $AoA = 17.5$ for Taper Ratio Evaluation

Results - Step A5: 3D Flap Configuration Optimisation

The automated computational analysis of six flap configurations yielded definitive recommendations for both take-off and landing phases, with the optimisation algorithm successfully balancing the competing requirements of lift generation, aerodynamic efficiency, and operational practicality. The results from the Python optimisation framework are systematically presented in Tables 14 and 15.

Table 14: Complete Flap Configuration Analysis for Landing Phase (Required C_L : 1.097)

Flap	Operating α	In Preferred α Range	Achieved C_L	C_D at Operating α	L/D	Total Score
0° Deflection	9.0°	No ($\approx$)	1.105	0.0477	23.2	78.6
5° Deflection	7.5°	Yes	1.131	0.0494	22.9	92.4
10° Deflection	6.5°	Yes	1.115	0.0490	22.8	91.6
15° Deflection	5.5°	Yes	1.105	0.0512	21.6	90.2
20° Deflection	5.0°	Yes	1.123	0.0573	19.6	89.6
25° Deflection	4.5°	Yes	1.139	0.0643	17.7	89.1
30° Deflection	3.5°	Yes	1.098	0.0684	16.1	85.3

Table 15: Complete Flap Configuration Analysis for Take-off Phase (Required C_L : 0.818)

Flap	Operating α	In Preferred α Range	Achieved C_L	C_D at Operating α	L/D	Total Score
0° Deflection	6.5°	Yes	0.861	0.0308	28.0	97.7
5° Deflection	4.5°	Yes	0.853	0.0307	27.8	99.3
10° Deflection	3.5°	Yes	0.847	0.0315	26.9	98.1
15° Deflection	3.0°	Yes	0.888	0.0369	24.1	96.3
20° Deflection	3.0°	Yes	0.951	0.0457	20.8	95.4
25° Deflection	3.0°	Yes	1.012	0.0555	18.2	91.6
30° Deflection	3.0°	Yes	1.058	0.0657	16.1	88.5

The composite scoring results from the weighted multi-criteria evaluation are summarised in Table 16, revealing the optimal configurations for each flight phase.

Table 16: Optimal Flap Configuration Analysis Results

Flight Phase	Optimal Flap Deflection	Total Score	Operating α	Achieved C_L	L/D	C_L Margin
Take-off	5°	99.3/100	4.5°	0.853	27.8	4.3%
Landing	5°	92.4/100	7.5°	1.131	22.9	3.1%

For the take-off phase, the 5° flap configuration achieved near-perfect scoring (99.3/100), comfortably exceeding the required C_L of 0.818 with a 4.3% margin while maintaining excellent aerodynamic efficiency ($L/D = 27.8$) at a practical operating angle of attack of 4.5°. This configuration significantly outperformed both the clean wing (0° flaps, score: 97.7) and higher deflection angles, which suffered from progressively deteriorating L/D ratios despite generating higher lift coefficients.

In the landing phase, the same 5° flap deflection emerged again as optimal with a score of 92.4, successfully meeting the more demanding C_L requirement of 1.097 with a 3.1% safety margin. The analysis revealed that while higher flap deflections (20°- 30°) generated greater lift coefficients (up to $C_L = 1.139$ at 25°), this came at a severe penalty to aerodynamic efficiency, with L/D ratios degrading by approximately 30% compared to the 5° configuration.

Part A: Discussion

Discussion - Step A3: Ideal Airfoil Selection

The data in Table 11 shows the key trade-offs involved in airfoil selection. The NACA 6612 is the theoretical peak performer with raw aerodynamic performance, boasting the highest $(L/D)_{max}$ (210.60) and $C_{l,max}$ (2.05). However, it was rejected due to its high pitching moment ($C_m = 0.1987$). This characteristic would necessitate a larger, heavier horizontal stabilizer and incur significant, constant trim drag, effectively negating its L/D advantage and increasing the aircraft's structural weight which would negatively impact the electric aircraft's range.

Conversely, the NACA 2212 presents excellent stability with the lowest pitching moment ($C_m = 0.0335$). However, its lower $(L/D)_{max}$ (149.12) and its operational C_l (1.16), which only marginally meets the minimum requirement, make it a less effective candidate for maximizing the critical range parameter of the electric aircraft.

The NACA 3213 successfully navigates this compromise. It retains a high $(L/D)_{max}$ (172.30), which directly translates to extended range according to the Breguet-derived equation. Simultaneously, its mild pitching moment ($C_m = 0.0512$) ensures manageable trim demands and contributes to overall aircraft stability without requiring excessive tail sizing. Furthermore, it provides a robust operational Cl (1.28), creating a solid foundation for flap integration to meet take-off and landing requirements, coupled with gentle stall characteristics for enhanced safety. This balanced performance profile makes the NACA 3213 the most suitable choice for the efficiency and stability needs of a general-aviation electric aircraft.

Discussion - Step A4: 3D Wing Optimisation

The results of the 3D wing optimisation demonstrate the profound impact of planform geometry on aerodynamic performance, validating the theoretical relationship between aspect ratio, induced drag, and overall efficiency.

Aspect Ratio Selection: The Efficiency Compromise

The clear improvement in both maximum and cruise L/D ratios with increasing aspect ratio, as evidenced in Table 12, directly corroborates the fundamental aerodynamic principle that induced drag scales inversely with AR ($C_{Di} \propto 1/AR$). The 12% improvement in cruise L/D achieved by the AR = 10 wing over the AR = 7 baseline allows a large increase in the aircraft's range, a critical performance parameter for an electric aircraft with limited energy storage. Furthermore, the corresponding reduction in cruise AoA from 1.69° to 1.35° indicates a more aerodynamically clean wing operating closer to its design point, reducing profile drag and improving overall efficiency.

However, despite this apparent clear improvement on pure aerodynamic efficiency with increases in AR values, practical constraints should also be considered. While an even higher AR might offer further drag reduction, it would incur penalties in structural weight and stiffness, increasing susceptibility to flutter and complicating the aircraft's structural design.

The selection of $AR = 10$ therefore represents an optimal engineering compromise, maximising aerodynamic performance within the conventional bounds for light general aviation aircraft, balancing efficiency with structural practicality and safety.

Taper Ratio Refinement: The Subtle Art of Load Distribution

The Phase 2 results revealed a more nuanced relationship between taper ratio and performance. The variation in the performance metrics recorded in the Polar graphs were miniscule ($< 0.5\%$ in cruise L/D across the tested range). This indicates that for a well-chosen AR , the primary aerodynamic benefit has already been captured. However, the critical differentiator emerged in the analysis of the Oswald efficiency factor (e) and the spanwise lift distribution.

The selection of $\lambda = 0.7$, with its peak Oswald efficiency factor (e) of 0.929, signifies a lift distribution that most closely approximates the ideal elliptical form, thereby minimising induced drag for a given lift coefficient.

In addition to e , the lift distribution analysis (Figures 6.25 - 6.27) provided the decisive evidence for choosing an optimal taper ratio. The $\lambda = 0.65$ configuration's pronounced lift reduction at the wing root is a characteristic of highly tapered wings. In contrast, the $\lambda = 0.7$ configuration demonstrated a more robust and progressive stall pattern, ensuring safer low-speed handling characteristics.

Aircraft Performance and Range Calculation

The ultimate objective of the aerodynamic optimisation was to maximise the range of the electric aircraft. Using the derived lift-to-drag ratio ($L/D = 31.46$) from Table 13 which used the optimal wing configuration of a NACA 3213 with $AR = 10$ and $\lambda = 0.7$ at cruise conditions, the theoretical range was calculated using the Breguet-derived range equation for electric aircraft [Eqn. 13].

Substituting the given and optimised parameters given from (Frawley, 2025):

- Energy per mass = 600,000 J/kg
- Efficiency, $\eta = 0.80$
- Gravitational acceleration, $g = 9.81 \text{ m/s}^2$
- Lift-to-Drag ratio, $L/D = 31.46$
- Mass of battery, $m_b = 400 \text{ kg}$
- Mass of aircraft, $m_a = 2000 \text{ kg}$

$$\text{Range} = \left(\frac{600,000 \times 0.80}{9.81} \right) \times 31.46 \times \left(\frac{400}{2000} \right) = 308 \text{ km}$$

Therefore, the designed wing enables a calculated aircraft range of 308 kilometres under the specified cruise conditions, neglecting fuselage drag.

Discussion - Step A5: 3D Flap Configuration Optimisation

The results of the flap optimisation reveal a critical design paradigm: beyond a certain point, the increased lift via flap deflection yields diminishing returns and becomes worse for overall aircraft performance. The selection of a modest 5° flap deflection for both take-off and landing phases represents the optimal compromise, maximising the benefit-to-cost ratio of deploying high-lift devices.

The generated performance trends, illustrated in Figures 6.28 through 6.33 below, provide crucial insights into the flap behaviour. Figure 6.28 demonstrates a near-linear degradation in L/D with increasing flap angle, decreasing from approximately 23.2 at 0° deflection to 16.1 at 30° deflection. This is caused by increase in C_D that comes with the increase in flap deflection angle. Similarly, Figure 6.31 reveals the same trend for take-off configurations, though with higher baseline efficiency values.

Analysis of operating angles, shown in Figure 6.30, proved particularly revealing for landing configurations. The clean wing (0° flaps) required an operating α of 9.0° to meet the C_L requirement, falling outside the preferred $3 - 8^\circ$ range for pilot visibility. All deflected flap configurations successfully operated within this preferred envelope, with the optimal 5° configuration utilising 7.5° α . For take-off, as shown in Figure 6.33, all configurations operated comfortably within the preferred α range of $3 - 8^\circ$.

The computational analysis framework was instrumental in navigating these complex trade-offs. By automating the evaluation of over 450 operating points against the weighted multi-criteria matrix, the algorithm eliminated subjective bias and ensured a systematic, repeatable optimisation process. Its ability to rapidly identify the optimal operating point for each configuration factoring in the simultaneous constraints of C_L requirement, α range, and L/D efficiency was crucial in identifying the 5° deflection as a superior solution that a less rigorous, manual analysis might have overlooked.

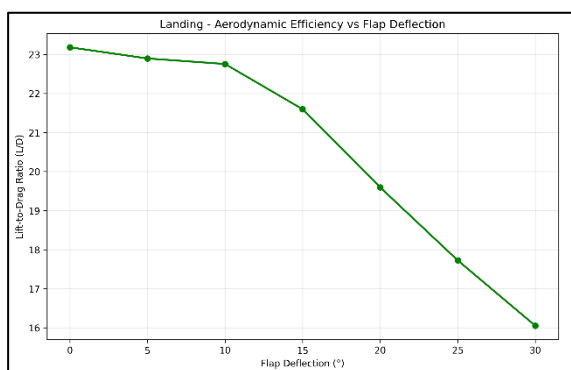


Figure 6.29: Aerodynamic Efficiency (L/D) as a Function of Flap Deflection for Landing Configurations.

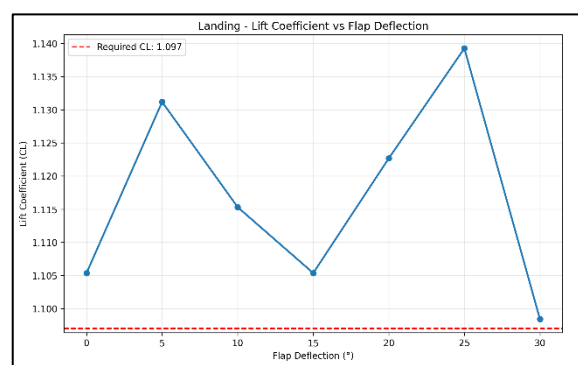


Figure 6.28: Achieved Lift Coefficient (C_L) vs. Flap Deflection for Landing Phase against Required C_L Threshold.

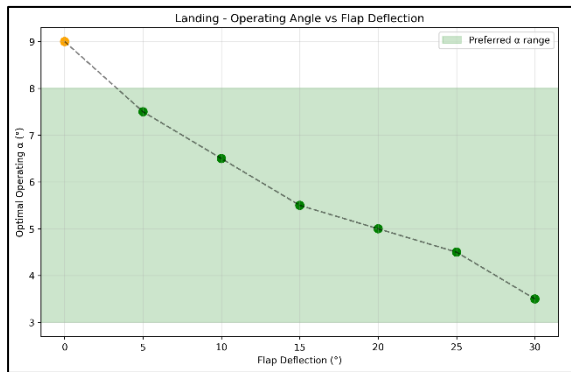


Figure 6.31: Optimal Operating Angle of Attack (α) for Landing Phase across Flap Deflection Angles

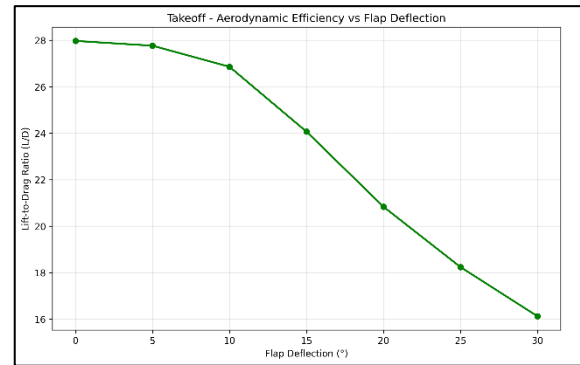


Figure 6.30: Aerodynamic Efficiency (L/D) as a Function of Flap Deflection for Take-off Configurations.

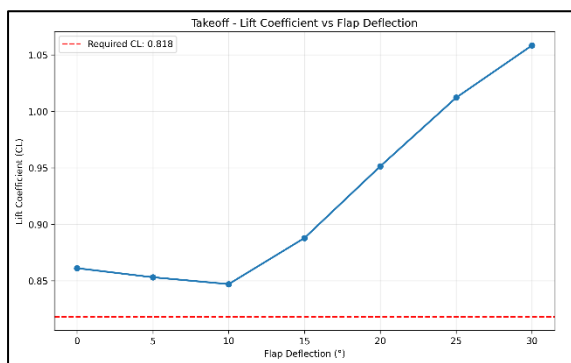


Figure 6.32: Achieved Lift Coefficient (C_L) vs. Flap Deflection for Take-off Phase against Required C_L Threshold.

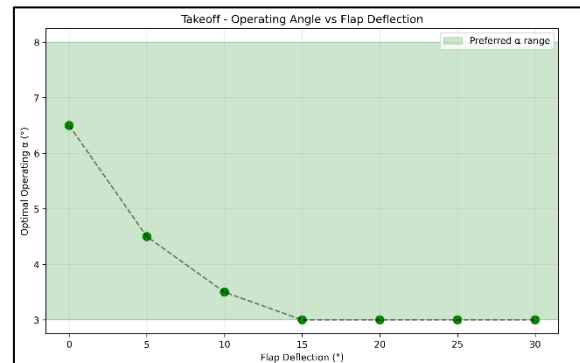


Figure 6.33: Optimal Operating Angle of Attack (α) for Take-off Phase across Flap Deflection Angles.

7. Part B: Emergency Parachute Design

Part B: Methodology

The objective of this section is to design a two-stage emergency parachute system for the electric aircraft designed in Part A, ensuring occupant safety by limiting deceleration forces and achieving a safe landing velocity. The design is based on two key operational phases derived from the problem statement (Frawley, 2025):

- *Stage 1 - Horizontal Deceleration:* The initial deployment must rapidly reduce horizontal velocity from cruise speed without exceeding a deceleration of 1g (9.81 m/s²), a force tolerable for the airframe and passengers.
- *Stage 2 - Vertical Descent:* After the horizontal velocity is dissipated, the fully deployed parachute must limit the vertical descent rate to 28 km/hr (7.78 m/s) at sea level, ensuring a survivable impact.

A reefing system is assumed, where the parachute initially deploys to a constrained area for Stage 1, then fully expands for Stage 2 once horizontal velocity is reduced.

- Stage 1 (Reefed): $A_{\text{reefed}} = \frac{2Ma}{\rho V_i^2 C_{D_{\text{para}}}}$, from Newton's Second Law
- Stage 2 (Full): $A_{\text{full}} = \frac{2Mg}{\rho V_{\text{vert}}^2 C_{D_{\text{para}}}}$, from vertical force equilibrium

Key assumptions for analysis

- The parachute drag coefficient, $C_{D_{\text{para}}}$, is 1.42, based on standard values for hemispherical canopies (NASA, 2011)
- Initial deployment occurs at the cruise altitude of 3,000 m, with ρ of 0.91 kg/m³.
- The vertical descent equilibrium is calculated at sea level, with an air density (ρ) of 1.225 kg/m³, providing a conservative estimate for the final landing phase.
- The aircraft mass (m) is 2,000 kg.
- Total airframe's drag [$C_{D_{\text{fuselage}}} \approx 0.02$ (Frawley, 2025) + $C_{D_{\text{wing}}} \approx 0.01081$ (from Table 12)] is negligible compared with the parachute's drag at these speeds as they were calculated to be (< 2% of target force)

Part B: Results

Reefed Parachute Sizing (Stage 1): The area required to produce a 1g deceleration at the instant of deployment is derived from Newton's Second Law, equating drag force to the decelerating force.

$$D_{\text{Para}} = (m \cdot a)$$

$$\frac{1}{2} \cdot \rho \cdot V_i^2 \cdot C_{D_{\text{para}}} \cdot A_{\text{reefed}} = m \cdot a$$

$$A_{\text{reefed}} = \frac{2 \cdot m \cdot a}{\rho \cdot V_i^2 \cdot C_{D_{\text{para}}}} \rightarrow \text{Substituting: } A_{\text{reefed}} = \frac{2 \cdot 2000 \cdot (-9.81)}{0.91 \cdot 83.33^2 \cdot 1.42} = 4.37 \text{ m}^2$$

Fully Deployed Parachute Sizing (Stage 2): The area required for a safe vertical descent rate is derived from the force equilibrium at terminal velocity, where parachute drag equals the aircraft's weight.

$$\text{Drag} = \text{Weight}$$

$$\frac{1}{2} \cdot \rho \cdot V_i^2 \cdot C_{D_{\text{para}}} \cdot A_{\text{full}} = m \cdot g$$

$$A_{\text{full}} = \frac{2 \cdot m \cdot g}{\rho \cdot V_{\text{vertical}}^2 \cdot C_{D_{\text{para}}}} \rightarrow \text{Substituting: } A_{\text{full}} = \frac{2 \cdot 2000 \cdot (9.81)}{1.225 \cdot 7.78^2 \cdot 1.42} = 372.69 \text{ m}^2$$

Table 17: Emergency Parachute Design Summary

Parameter	Reefed Stage (Deceleration)	Fully Deployed Stage (Descent)
Projected Area	4.37 m ²	373 m ²
Diameter	2.36 m	21.8 m
Design Constraint	Max 1g deceleration	28 km/h descent speed

Part B: Discussion

Operational Sequence and Safety Feasibility

The system functions through a precisely choreographed sequence: a rocket-deployed, reefed parachute of $\sim 4.4 \text{ m}^2$ initially inflates, violently decelerating the aircraft and converting its forward motion into a descending arc. Once horizontal velocity is sufficiently dissipated, the reefing mechanism releases, allowing the canopy to expand to its full 373 m^2 size. This larger area is then optimised to interact with the thicker sea-level air and achieve the target terminal velocity of 28 km/h.

A deployment altitude of 400 m is more than sufficient for this two-stage process. The aircraft would reach its target descent rate within the first 20-30 meters of vertical fall after full deployment, leaving over 370 meters to descend safely at 7.78 m/s. The total time from deployment at 400 m to landing would be approximately 50 seconds, providing a stable and controlled descent profile that ensures a high probability of occupant survival and minimal airframe damage.

Force Management:

The results shown in Table 17 clearly demonstrate the need for a two-stage reefing system. The large difference between the reefed area (4.37 m^2) and the fully deployed area (373 m^2) is driven by the relationship between drag force and velocity squared ($D \propto V^2 A$). Deploying the full 373 m^2 canopy at the cruise speed of 83.33 m/s would generate a catastrophic deceleration force. A preliminary calculation shows this force would exceed 190 kN, subjecting the aircraft and its occupants to over 9.5g, which would likely cause structural failure and serious injury. The reefed stage limits this initial force to 19.6 kN ($2,000 \text{ kg} \cdot 1\text{g}$).

Validation against empirical data:

The calculated fully deployed parachute area of 373 m^2 was validated through comparison with an existing certified system for a geometrically similar aircraft. The Ballistic Recovery System (BRS) for the Cessna 172 aircraft (Ballistic Recovery Systems Inc., n.d.) provided a reliable empirical comparison for the design.

The Cessna 172 has a maximum gross weight of 1,150 kg and is equipped with a parachute of 223 m^2 (Ballistic Recovery Systems Inc., n.d.). To enable a direct comparison between the two different aircraft, the key design parameter of the mass-to-area ratio was calculated.

- Cessna 172 Parachute Loading: $\frac{1,150 \text{ kg}}{223 \text{ m}^2} \approx 5.16 \text{ kg/m}^2$
- Designed Aircraft Parachute Loading: $\frac{2,000 \text{ kg}}{373 \text{ m}^2} \approx 5.36 \text{ kg/m}^2$

The close agreement between these parachute loading values (5.16 kg/m² vs. 5.36 kg/m²) demonstrates that the calculated area of 373 m² is consistent with industry practice for aircraft of this class. This strong correlation with a proven real-world system provides high confidence in the realism and accuracy of the design calculation, serving as a successful external validation.

8. Part C: High-Mach Suitability Analysis

Part C: Methodology

The critical Mach number (M_{cr}) was determined using a rigorous analytical approach based on compressible flow theory. The methodology comprised three key phases:

Phase 1: Incompressible Baseline Establishment

The minimum pressure coefficient ($C_{p,min}$) for the NACA 3213 airfoil was obtained through XFLR5 analysis at the design cruise angle of attack ($\alpha = 1.35^\circ$) under incompressible flow conditions ($M = 0.01$). This yielded the fundamental parameter $C_{p,min,0} = -0.81$, representing the most negative pressure coefficient on the airfoil surface at low speed.

Phase 2: Compressibility Correction and Critical Threshold Analysis

Two distinct curves were generated to analyse the airfoil's behaviour across the Mach number spectrum:

1. Airfoil Cp Progression: The Prandtl-Glauert compressibility correction was applied to predict how the minimum pressure coefficient evolves with increasing Mach number [Eqn. 14]. This relationship accounts for the increasing flow acceleration over the airfoil as compressibility effects become significant.
2. Critical Cp Curve: The theoretical pressure coefficient required for local sonic flow was calculated using the isentropic flow relation [Eqn. 15]. This curve defines the "sonic threshold" which is the minimum Cp value that can exist before local flow reaches Mach 1.

Phase 3: Intersection Analysis and M_{cr} Determination

The two curves were computed across Mach numbers from 0.3 to 0.8 in fine increments of 0.025. Linear interpolation was applied to precisely identify the intersection point where:

$$C_p(M_{cr}) = C_{p,cr}(M_{cr})$$

Part C: Results

Critical Mach Number Determination

The analytical determination of the critical Mach number yielded clear and definitive results. Table 18 presents the comprehensive dataset used for the intersection analysis, showing the progression of both the airfoil's minimum pressure coefficient and the critical sonic threshold across the Mach number range.

Table 18: Critical Mach Number Intersection Analysis

Mach Number	Airfoil C_p (Prandtl-Glauert Corrected)	Critical C_p (Sonic Threshold)
0.3	-0.849111	-6.94732
0.325	-0.856496	-5.83758
0.35	-0.864692	-4.95643
0.375	-0.873763	-4.24494
0.4	-0.883782	-3.66202
0.425	-0.894836	-3.17827
0.45	-0.907026	-2.77225
0.475	-0.920469	-2.42798
0.5	-0.935307	-2.1334
0.525	-0.951707	-1.87923
0.55	-0.969869	-1.65826
0.575	-0.990034	-1.46481
0.6	-1.0125	-1.29434
0.625	-1.03763	-1.14323
0.65	-1.065882	-1.00853
0.675	-1.097832	-0.8878
0.7	-1.134227	-0.77907
0.725	-1.176045	-0.68066
0.75	-1.224605	-0.59121
0.775	-1.281724	-0.50953
0.8	-1.35	-0.43464

The intersection point between the two curves was precisely determined through linear interpolation between Mach 0.625 and 0.650, where the airfoil's pressure coefficient transitions from being less negative than the critical value to more negative. This analysis established the critical Mach number for the NACA 3213 airfoil as $M_{cr} = 0.641$ using linear interpolation.

Figure 8.1 graphically illustrates this intersection, clearly showing the point at which the airfoil's minimum pressure coefficient reaches the theoretical sonic threshold, indicating the onset of local supersonic flow on the airfoil surface.

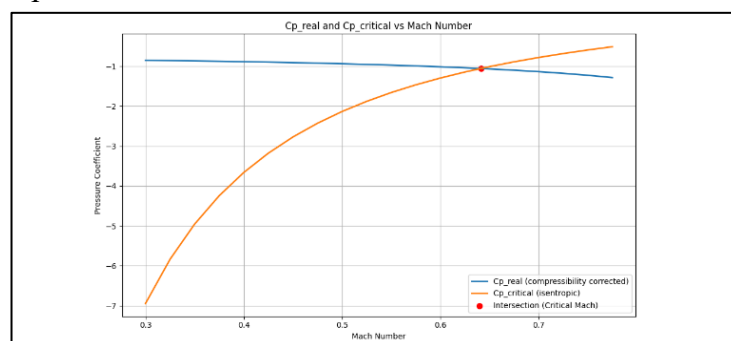


Figure 8.1: Determination of Critical Mach Number for NACA 3213 Airfoil via $C_{p,min}$ and $C_{p,critical}$ Intersection Analysis

Part C: Discussion

The comprehensive analysis confirms that the NACA 3213 airfoil, while optimal for a low-speed electric aircraft, is fundamentally unsuitable for high-speed flight at Mach 0.8. This unsuitability is demonstrated by the airfoil's critical Mach number of 0.641, indicating that local supersonic flow would develop well below the target cruise condition. Furthermore, at Mach 0.8, the substantial disparity between the airfoil's pressure coefficient (-1.350) and the critical threshold (-0.435) indicates extensive regions of supersonic flow that would inevitably lead to strong shockwaves, transonic drag rise, and potential control degradation.

Root Cause Analysis

The causes of this limitation are inherent in the airfoil's geometric characteristics that were optimised for low-speed performance. The 13% thickness ratio, while beneficial for structural feasibility and high lift at low speeds, creates significant flow acceleration over the airfoil surface. Combined with the 3% camber, these features produce pressure coefficients that readily exceed the sonic threshold as Mach number increases. This confirms the fundamental design compromise between high-lift, structurally efficient airfoils for low-speed applications and the thin, minimally cambered sections required for high-speed flight.

Compressibility Effects Validation

The onset of compressibility effects is evident well before the critical Mach number, as validated by analysis at $M = 0.5$. As shown in Table 19, lift, drag, and pressure coefficients all increased by approximately 15.4 - 15.7%, aligning closely with the 15.5% increase predicted by the Prandtl-Glauert correction. This consistency between XFLR5 results and theoretical predictions confirms that aerodynamic performance begins to degrade progressively as Mach number increases, underscoring the severity of operating an airfoil far beyond its M_{cr} .

Table 19: Sub-Critical Aerodynamic Performance at $M = 0.5$

Parameter	Incompressible Value ($M = 0.01$)	Compressible Value ($M = 0.5$)	Percentage Change
$C_{p,min}$	-0.810	-0.935	+15.4%
C_L	0.345	0.398	+15.4%
C_D	0.0108	0.0125	+15.7%

Changes Required for High-Mach Flight

To achieve efficient flight at Mach 0.8, three fundamental modifications are necessary. First, reducing thickness to 6 - 8% would directly decrease flow acceleration over the airfoil, thereby increasing M_{cr} . Second, adopting a 25 - 35° wing sweep would reduce the effective normal Mach number component given by $M_n = M \cdot \cos(\Lambda)$, effectively delaying drag divergence. Finally, transitioning from a cambered airfoil to a supercritical airfoil, which has a characteristically flattened upper surface designed to delay shock formation, as illustrated in Figure 8.2, helps minimise wave drag and produce more favourable pressure distributions at transonic speeds (Harris, 1990). These specialized airfoils see widespread use in UAVs and experimental aircraft, with notable implementation in the Boeing C-17 Globemaster III, where

the NASA SC(2)-0414 airfoil (Figure 8.2) demonstrates proven transonic capability up to Mach 0.74 (United States Air Force, 2018). Collectively, these modifications would transform the wing from a low-speed optimized configuration to a high-speed platform, accepting the inherent compromises in low-speed lift and structural efficiency required for transonic flight.

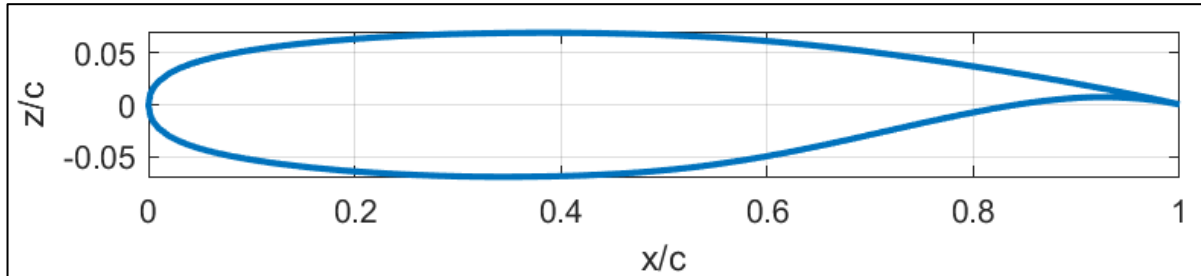


Figure 8.2: Section geometry of NASA SC(2)-0414 airfoil used in the Boeing C-17 Globemaster III. Source: (Prakasha, 2018)

9. Integrated Discussion

Looking back at the complete design, a clear story emerges where every part of the aircraft design is connected. The choices made for the wing didn't just affect its performance, they directly influenced the size of the parachute and clearly defined what the aircraft can and cannot do. The entire process was a balancing act, where optimising for one goal meant making trade-offs in other areas.

Summary of key integrated findings

- Airfoil chosen: NACA 3213. Balanced performance: $(L/D)_{\max} \approx 172.3$, $C_{l,\max} \approx 1.84$, $\text{avg } |C_m| \approx 0.0512$.
- Planform selected: Aspect ratio $AR = 10$, taper $\lambda = 0.70$. Cruise performance: $L/D_{\text{cruise}} \approx 31.46$, $C_{D,\text{cruise}} \approx 0.01081$, cruise $\text{AoA} \approx 1.35^\circ$.
- Flap configuration: Plain flap at 25% chord, 50% span; 5° deflection chosen for both take-off and landing (best composite score). Take-off achieved $C_L = 0.853$ (margin 4.3%) and landing achieved $C_L = 1.131$ (margin 3.1%).
- Range (battery model): Using $E = 600,000 \text{ J/kg}$, $\eta = 0.80$, $m_{\text{battery}} = 400 \text{ kg}$, $m_{\text{aircraft}} = 2000 \text{ kg}$, and $L/D = 31.46$, the estimated range is $\approx 308 \text{ km}$ (fuselage drag neglected).
- Emergency parachute: Reefed area $\approx 4.37 \text{ m}^2$ (1g deceleration @ 83.33 m/s at 3000 m), full area $\approx 373 \text{ m}^2$ (terminal $V \approx 7.78 \text{ m/s}$ at sea level). Parachute loading $\approx 5.36 \text{ kg/m}^2$, comparable to BRS/Cessna data ($\sim 5.16 \text{ kg/m}^2$).
- High-Mach suitability: $M_{cr} \approx 0.641$ for NACA 3213, This confirms the airfoil is unsuitable for Mach 0.8 without major redesign (thinner section, sweep, or supercritical section required).

Discussion of Main integrated findings:

1. Efficiency vs. stability vs. structural practicality.

NACA 6612 offered exceptional 2D L/D and $C_{L,max}$ but failed the pitching-moment constraint; adoption would demand a larger stabiliser and heavier structure, reducing net range. NACA 3213 gives high L/D while keeping $|C_m|$ small enough to avoid excessive trim drag which the right compromise for an electric general aviation aircraft.

2. Aspect ratio and taper selection.

Increasing AR from 7 to 10 produced significant induced-drag reduction and improved cruise L/D ($\approx 12\%$ improvement). AR = 10 was chosen to maximise range while keeping wing geometry realistic for the aircraft by avoiding excessive structural weight or flutter risk. Taper $\lambda = 0.70$ produced the best Oswald efficiency and the most favourable stall progression. It was noted that aerodynamic gains were small across λ , but λ influenced root loading and stall behaviour which were considered a safety-critical effect.

3. Flap sizing: small deflection outperforms large deflection in system terms.

Larger flap angles increase C_L but impose severe L/D penalties and increase required pilot AoA for cruise/approach transitions. The optimisation showed 5° gives the best net performance (meets lift thresholds while preserving range and handling), which is useful for an electrically limited aircraft where energy economy is primary.

4. Parachute design must have two stages.

The two-stage reefing approach is essential: a small, reefed canopy limits initial g loads on occupants then the full canopy addresses vertical descent. The reefed/full area pair ($4.37 \text{ m}^2/373 \text{ m}^2$) is consistent with empirical BRS data after normalising by mass-to-area loading.

5. Low-speed optimisation vs. high-speed incompatibility.

The chosen airfoil and wing geometry are tailored to low-subsonic cruise ($M \approx 0.25$) and provide excellent electric-aircraft range. The critical Mach analysis shows local supersonic flow onset near $M \approx 0.64$, thus, the wing is unsuitable for transonic operation. Any goal to operate at $M \approx 0.8$ would require a fundamentally different design. It was found that including an airfoil with a thin section, sweep, or supercritical airfoil greatly improves performance at high Mach.

10. Conclusion

This study successfully developed a preliminary design of a wing and emergency-parachute system for a four-seat electric aircraft of mass 2,000 kg operating at 300 km/h at 3,000 m. A structured computational workflow which included 2-D airfoil screening with XFOIL (automated by python), followed by 3-D planform and flap optimisation system in XFLR5 was used to navigate the complex design trade-offs.

The investigation found that the optimal configuration was the NACA 3213 airfoil, with an aspect ratio of 10 and a taper ratio of 0.70. A plain flap system deflected to 5° was identified as the optimal setting for both take-off and landing, successfully meeting the required lift coefficients ($C_L \approx 0.853$ and 1.131 , respectively) while preserving its aerodynamic efficiency (cruise $L/D \approx 31.46$). This optimised wing enables a theoretical aircraft range of approximately 308 km, based on the given battery parameters and neglecting fuselage drag.

The parachute analysis demonstrated that a parachute system for an aircraft needs a two-stage reefing system to ensure occupant safety. A small, 4.4 m^2 reefed canopy ensures safe deceleration ($\leq 1g$) from cruise speed, while a large, 373 m^2 fully deployed canopy guarantees a survivable descent rate of 7.78 m/s . The close alignment of the calculated mass-to-area ratio (5.36 kg/m^2) with certified systems (e.g., Cessna 172 BRS) validates the design.

The project also highlighted the inherent trade-offs in aircraft design. The wing, while perfectly suited for its low-speed, efficient mission, is fundamentally unsuitable for high-speed flight at Mach 0.8, with a critical Mach number of just 0.641. It was found that if a wing was needed high Mach conditions it would need to be thinner, have a larger sweep angle (25 to 30 degrees) and consider an airfoil from the NASA super critical wing series. This clearly demonstrates that aircraft design is a process of optimisation for a specific mission profile.

In summary, this report delivers a viable, well-justified preliminary wing and parachute design for an electric aircraft. The systematic methodology, particularly the use of weighted multi-criteria optimisation, effectively balanced competing objectives to maximise range and ensure safety. This work provides a solid foundation for future development, which should include the incorporation of fuselage drag and structural analysis for a more complete analysis. Furthermore, the computational framework established here is directly applicable to exploring advanced configurations, such as wings employing NASA supercritical airfoils or sweep, to extend the design's capabilities into the high-subsonic regime.

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12. Appendix

Appendix A: Code repository created and used for analysis

All code used throughout the report for data analysis can be found in this linked GitHub repository: [Click to View full GitHub repository ↗](#). A variety of supporting code is listed in the GitHub but one may use the hyperlink guide below to quickly navigate the repository and find the main solvers used:

- A.1. XFOIL NACA aerodynamic data extractor: [Click to View Code. ↗](#)
- A.2. Ideal NACA airfoil finder: [Click to View Code. ↗](#)
- A.3 Script for creating visualisation of any 4-digit NACA airfoil: [Click to View Code. ↗](#)
- A.4 Script for optimising flap deflection configuration of airfoil. [Click to View Code. ↗](#)
- A.5 Mach vs $C_{p,min}$ vs $C_{p,critical}$ plotter and intersection solver: [Click to View Code. ↗](#)

Appendix B – Excel Data Analysis Files

This appendix provides links to the Excel workbook used for data analysis.

- B.1 Ashen Wijayarathne's – [Excel Workbook ↗](#)