

Mission Vold-e Design of Prandtl-Plane eVTOL

Final Report

DSE Aerospace Design
Group 23



Mission Vold-e

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by

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Executive Overview

Over the years, a solid market has established for electric vertical take-off and landing aircrafts (eVTOLs). However, in order for eVTOLs to be truly effective, they will need to be efficient and have a small ground footprint. This creates a problem: high efficiency requires a larger wing span, increasing the ground footprint. The goal of this project is to design an eVTOL that redefines air mobility by enabling autonomous, near silent, sustainable transport across cities and geographic barriers. This resulted in the design of Vold-e: a fully electric eVTOL that aims to fit in tight urban spaces without compromising on aerodynamic efficiency, by making use of a folding Prandtl wing. A render of Vold-e, showing the amount of space that can be saved by folding the wings, can be seen in Figure 1.

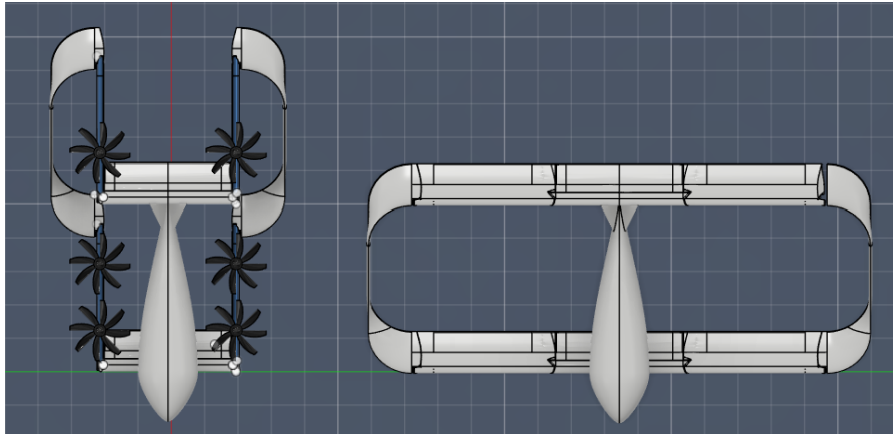


Figure 1: *Vold-e in folded (left) and unfolded (right) configuration*

Vold-e's Mission

Vold-e will be a fully electric, autonomous eVTOL able to operate within cities and across geographical barriers. It will be able to carry four passengers, over a maximum range of 200 [km], cruising at a max speed of 200 [km/h]. The market for Vold-e can be split in two: the current market, and the future market. In the current market, Vold-e is targeting premium transport, similar to helicopters today. More specifically, it is envisioned to be used as a winter resort shuttle in the Courchevel area, showcasing its ability to cross mountain ranges and land in cities. In the future market, Vold-e is envisioned to be widely adopted in European cities as a new mode of transportation. Here, the folding wing concept becomes extra important, as it reduces the infrastructure needed and allows it to land virtually anywhere.

Stakeholder Requirements and Configuration Selection

In the baseline, requirements have been imposed on the vehicle to be designed. The highest level of these requirements are the stakeholder requirements. These requirements are shown in Table 1.

Table 1: *Categorised stakeholder requirements for the eVTOL concept.*

ID	Stakeholder requirement
Performance	
REQ-STK-01	The vehicle shall be able to transport four passengers and their luggage.
REQ-STK-02	The vehicle shall have a ground footprint adequate for the usage of vertiports and landing pads as landing and take-off sites.
REQ-STK-03	The vehicle shall fly between regional city pairs without intermediate stops.
REQ-STK-04	The vehicle shall provide a faster travel time between cities than conventional ground based transportation.
REQ-STK-05	The vehicle shall safely operate within existing commercial airspace.
Regulatory & safety	
REQ-STK-06	The vehicle shall produce acceptable noise levels.
REQ-STK-07	The vehicle shall be safe to be used for transporting passengers.
Cost & sustainability	
REQ-STK-08	The main structure of the vehicle shall be repurposable.
REQ-STK-09	The final product cost shall not cost more than the costs of a H125 helicopter
REQ-STK-10	The vehicle shall have zero in-flight CO ₂ emissions.

Continued on next page

ID	Stakeholder requirement
Operations	
REQ-STK-11	The vehicle shall not need an on board pilot.
REQ-STK-12	The vehicle shall be able to be manoeuvred from hangar to hangar.
REQ-STK-13	The vehicle shall operate in urban areas without posing a safety hazard to uninvolved.
REQ-STK-14	The vehicle lifetime shall exceed 10 years.
REQ-STK-15	The vehicle shall be able to perform at least six stretches of the proposed mission per day.
Public acceptance	
REQ-STK-16	The vehicle shall have a willingness to use rate amongst the intended users of at least 50% as measured by pre-launch user acceptance surveys.

Based on these requirements, in the midterm, four concept vehicles have been designed and analysed on their feasibility in terms of aerodynamic efficiency, structural performance, stability, controllability, sustainability and noise. These aspects were then used to trade-off the four concepts. The weights assigned to each aspect was respectively: 0.25, 0.20, 0.15, 0.10 and 0.10. This trade-off included both quantitative estimates, such as cruise L/D and battery mass; as well as qualitative assessment, such as design complexity and aero-propulsive interaction. Based on this trade-off, the configuration that is now known as Vold-e was chosen. Although this configuration introduces additional risks related to the folding mechanisms and structural integration, it gives the best overall balance between compact ground footprint, aerodynamic performance, controllability, noise, and mission feasibility. In order to make sure that this configuration would be a robust choice, a sensitivity analysis was performed. This analysis confirmed that the chosen configuration, would remain the winner in the majority of the tested weighting various. As such, in the final report, this configuration was analysed and become Vold-e.

Subsystems

In this report, the design of the subsystems is central. The following five subsystems have been analysed: Aerodynamics, power & propulsion, structures, vehicle dynamics and autonomous operations.

Aerodynamics

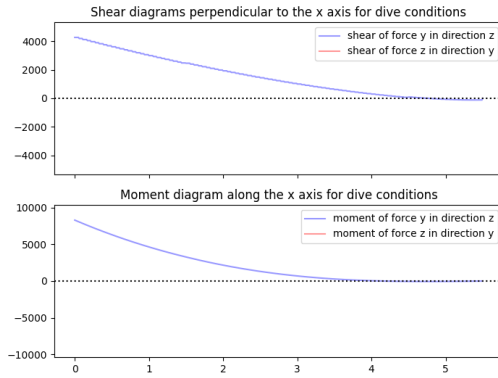
In terms of aerodynamics, the planform parameters have been sized, the airfoil has been chosen, the box wing has been sized and lastly, the fuselage shape has been determined. First, the high level planform parameters were determined to be: 0 [°] leading edge sweep, 0 [°] dihedral and, 0 [°] twist and a taper ratio of 0.45. Afterwards, the airfoil was chosen to be the NASA LANGLEY LS(1)-0417 due to its favourable aerodynamic efficiency and maximum lift coefficient. The vertical tail airfoil was chosen based on stability and structural considerations, and chosen to be the NACA 0010. The box wing was sized by means of positioning the aft wing relative to the front wing in terms of vertical spacing. In preliminary design, the vertical gap was chosen to be 2.1 [m], in order to comply with the footprint constraint. Theoretically, this would give an Oswald efficiency factor of 1.34: a 34% increase compared to a single wing. To determine the interference between the two wings, a VLM analysis was performed with different values for the vertical gap. From this, it was found that a vertical gap of 2.1 [m] would only decrease C_L by roughly 5% compared to one wing only, which was deemed acceptable. With this gap, a preliminary cruise angle of attack of $\alpha = 2.6^\circ$ was determined. The fuselage was sized by first establishing some constraints. These were that the fuselage diameter was assumed to be 1.8 [m], and the cabin length 2.5 [m]. With these constraints, different fuselages were designed, by varying the length, the shape of the nose, the longitudinal position of the passenger cabin, and the vertical position of the tail. These fuselages were then analysed using CFD at the cruise angle of attack $\alpha = 2.6^\circ$. From this, it was found that a fuselage of 7 [m], with a 1.5 [m] nose length, a blunt nose and the tail in the middle, would have the lowest drag coefficient of 0.0588.

Power and Propulsion

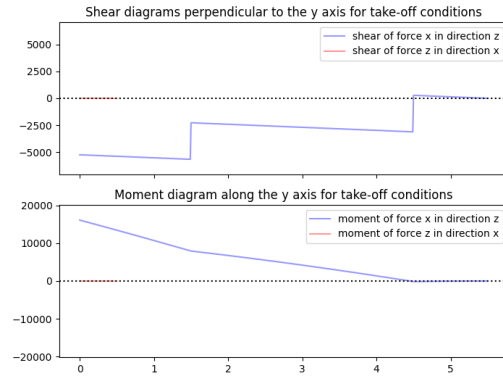
The battery is energy driven, and sized by the mission energy and optimized to a pack of 540 [kg] pack of 3995 silicon anode SA504 cells on a 800 [V] bus delivering around 142.4 [kWh] with the reserve energy included. Propulsion is done by six wing mounted rotors driven by electric motors to provide upward thrust in hover and forward thrust in climb and cruise. The propellers use a variable pitch propeller ensuring maximum efficiency during both hover and cruise. The high specific energy of the silicon anode cell is what lets the pack close within its mass budget.

Structures

Considering the structures of Vold-e different analyses were completed to assess different components. These components include analysing the loads on the wing and hinges, analysing the wing mass, sizing the vertical tail, landing gear sizing, cabin pressurization and finally the fuselage sizing and mass estimation. Firstly, the load analysis allowed for shear diagrams to be made for differing phases of the mission profile. Initially the shear diagrams for dive conditions and take-off conditions are seen in Figure 2a and Figure 2b.



(a) Moment diagrams for dive conditions w.r.t the wing axis



(b) Moment diagrams for take-off conditions w.r.t the wing axis

In order to size the wing and estimate its weight, the lift distribution along the wings are drawn and estimated how much loads the wings must be able to withstand. This allowed for the wing box to be defined. Next a detailed analysis of the hinge was constructed and the actuators required analysed.

The folding mechanism is paramount to Vold-e's small ground footprint. Combined with the Prandtl wing, it works by having eight hinges in total: each wing half has one close to the root and one at the tip. This allowed the majority of the wing to fold, whilst staying attached to the winglet. The hinges are designed such that all sides will stay parallel during the whole range of motion and ensures that the two wings fold simultaneously, without conflicts between the joints. A visualisation of the folding mechanism can be found in Figure 3. A detailed load analysis was carried out on the entire wing structure in order to properly assess the reaction forces on the hinges. This included studying the direction and magnitude of the wing's lift, weight and drag, as well as the reactions from the winglet and the propeller thrust. From these reactions, the hinges were properly sized to ensure they can endure the maximum loads imposed on the aircraft with a reasonable safety factor.

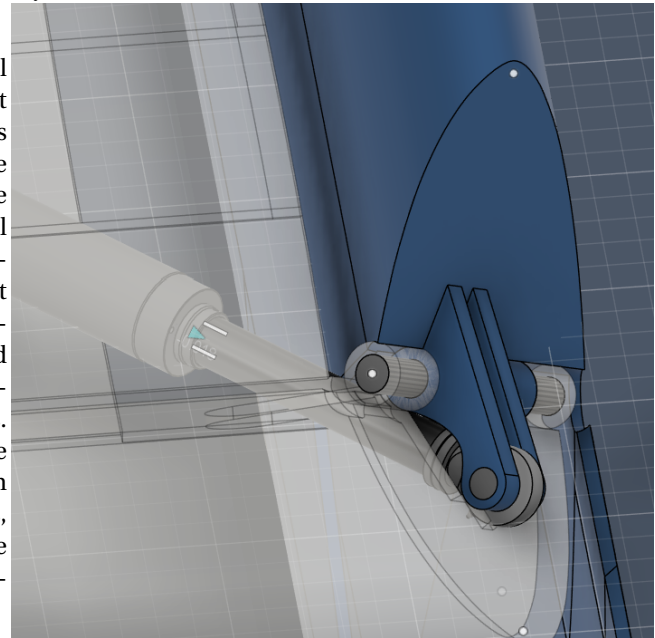


Figure 3: Folding mechanism

The landing configuration was determined by using a sensitivity trade-off method which determined that an elastomeric landing gear would be the most suitable. Lastly, a monocoque CFRP based fuselage was sized after considering various load cases; namely a pull-up manoeuvre, hard landing, and dive manoeuvre. Pressurisation of this fuselage was found to contribute negligible weight considering the altitude is not much higher than the pressurisation altitude.

Vehicle Dynamics

The aircraft has been assessed both statically and dynamically. Static stability results showed that a shift in CG is needed to incorporate with the OEI requirements in hover mode. Dynamic stability concluded that the aircraft is stable in trimmed cruise conditions for every classical eigen-motion except spiral.

Autonomous Control

Vold-e operates autonomously under remote pilot supervision, with the flight software structured as a closed sense, estimate, decide and act loop that fuses the navigation and perception sensors into a single state and obstacle map. Reliability is built in through redundancy: a x3 flight computer, radar backed all weather sensing, and a ballistic recovery parachute on an independent power reserve as the final failsafe.

Subsystem Integration

In order to integrate the subsystems, a central MTOW convergence loop was used, taking inputs from each department for known parameters. An initial guess for the MTOW is given, and this is iterated to a fixed point so that all sizings are mutually consistent. On top of that converged model sits a multi-objective genetic optimiser (NSGA-II) that minimises two objectives at once, namely the MTOW and the worst-case stability margin. This is done in order to trace the trade-off between a light aircraft and a stable one. It tunes eight design variables (front-wing position and height, the two

sliding-battery stations, vertical-tail span, the wing-area split, wing loading W/S , and the wingbox skin gauge) with bounds determined either via department constraints or computed via tools such as an updated matching diagram. These parameters were selected by utilising a robust Sobol sensitivity analysis. The result of the optimiser is that all stability related results match minimum targets and the MTOW reached in 1980 [kg].

Design Feasibility Assessment

The feasibility of the Vold-e eVTOL design was assessed through an in-depth evaluation of its aerodynamic and structural performance.

Aerodynamic Performance

To capture the aerodynamic interactions across the complete airframe, the integrated Vold-e eVTOL configuration was evaluated using steady-state RANS CFD simulations in ANSYS Fluent at a cruise altitude of 12,500 [ft] and a freestream velocity of 55.6 [m/s].

By constructing aerodynamic polars across a wide range of angles of attack (-4 [°] to 22 [°]), the Prandtl-plane layout's lift capabilities were validated, determining a maximum lift coefficient of 0.9917 at a stall angle of 19 [°]. Furthermore, the cruise angle of attack was estimated from the fitted lift curve using the cruise lift coefficient, $C_{L,cruise}$, obtained by imposing the $L = W$ condition at the final converged MTOW of 1980 [kg]. Running the simulations for the obtained cruise condition confirms that the vehicle achieves a highly efficient lift-to-drag (L/D) ratio of 13.167 at its designated cruise angle of attack of $\alpha = 5.49$ [°], operating smoothly without large-scale surface flow separation. Flow field analysis demonstrated that wake disturbances remain tightly constrained behind the fuselage and recover to freestream velocities withing the considered test domain. Mesh independence studies and turbulence model sensitivity assessment ensured the high fidelity and robustness of the obtained computational results, but remaining modelling limitations are still accounted for while interpreting the results.

All significant aerodynamic parameters obtained during the CFD analysis are outlined in Table 2.

Table 2: Main Aerodynamic Parameters

Parameter	Symbol	Value
Maximum Lift Coefficient	$C_{L,max}$	0.9917
Angle of Attack at Max Lift	$\alpha_{C_{L,max}}$	19°
Zero-Lift Angle of Attack	α_0	-1.83°
Zero-Lift Drag Coefficient	$C_{D,0}$	0.01702
Cruise Angle of Attack	α_{cruise}	5.49°
Cruise Lift Coefficient	$C_{L,cruise}$	0.47089
Cruise Drag Coefficient	C_D	0.03576
Cruise Lift-to-Drag Ratio	L/D	13.167
Effective Oswald Efficiency Factor	e	0.692

Structural Performance

The feasibility of the hinge design was tested by using finite element analysis on the 3D model. The results were used to detect the weak areas of the structure and reinforce them. A visualisation of the results can be found in Figure 4. In order to make sure the study provides valid results a mesh independence study was done to prove that the meshing can capture the stress correctly in all regions of the component. The results of this can be found in Figure 5.

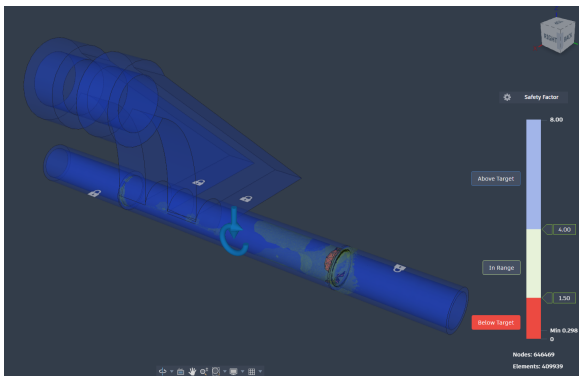


Figure 4: Results from FEA simulation

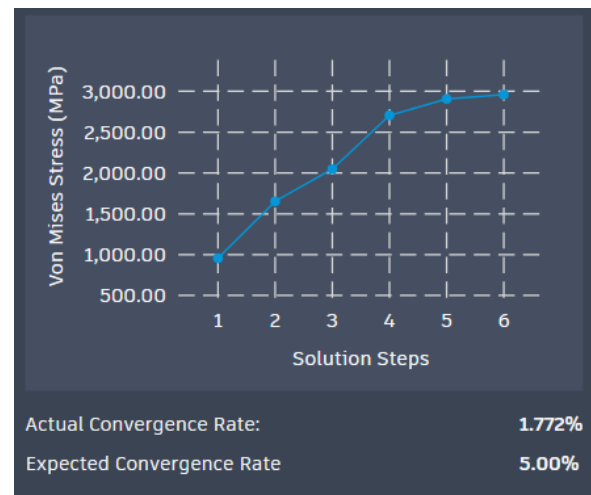


Figure 5: Mesh independence study.

Sustainability Assessment

The sustainability assessment was carried out by analysing the impact of the design on emissions. The two categories of emissions were:

- **Recurring Emissions** represent the constant emissions as a by-product of the aircraft's flight operations as well as the maintenance required. Since the aircraft has zero emissions, a focus was placed on the energy per kilometre and the source of electrical energy. Furthermore, the lifetime of aircraft components played a significant role in the design choices in order to lower maintenance costs.
- **Non-Recurring Emissions** represent the effects of aircraft production on the environment, as well as its infrastructure, and its impact on the land use during its operational lifetime. Although the design's current detail is insufficient to assess the sustainability of the aircraft's general production, thought was put into the battery sizing as it is considered very impactful on production emissions. As for land use, Vold-e's eVTOL and folding mechanism allows for a very small ground footprint, greatly reducing the land required for operations.

Financial Analysis

In order to assess Vold-e's feasibility in the future and current market a financial analysis was completed. Both the production and operational costs were analysed and the conclusion was made that during the initial phase of the market, Vold-e could be made for a value of 2.0 Million [€] and sold for a value of 3.0 Million [€] to ensure enough revenue. Looking at the future market these were expected to decrease to 1.1 Million [€] and 1.4 Million [€] respectively. To assess the validity of the models use, the costs for Vold-e in phase 1 were compared to the costs associated with Joby in Table 3.

Table 3: Comparison of Vold-e against the Joby eVTOL

Parameter	Vold-e	Joby ¹
Cost of production [Million €]	2.0	1.3
Selling price [Million €]	3.4	1.8
Operational cost per flight [€]	242	190
Ticket price per km [€]	1.7	3.0 (JFK to Manhattan)

Project Management & Systems Engineering

Throughout this project, care has been taken to implement project management and systems engineering principles. As such, the project started with identifying stakeholders, each having their own needs. These needs then flowed down into stakeholder requirements, as shown in Table 1, which in turn flowed into mission requirements and system requirements. Simultaneously, a functional flow and functional breakdown have been set up to determine which functions the vehicle shall be able to perform. Subsequently, the design was split into subsystems and different departments were created. The sub-departments are closely interconnected, demonstrating the collaborative effort made by members across all areas. Per subsystem, subsystem requirements were then set up. In terms of work to be done, a detailed work flow diagram, work breakdown diagram and Gantt chart were created to keep track of the work. At the end of this project, however, it was concluded that the design is not fully finished. As such, a detailed project design and development diagram and Gantt chart were created for post DSE activities. These activities will show what steps still need to be performed in order to complete the design.

Usage of AI

In order to ensure a fair and correct usage of AI, certain rules were set within the group. These rules are as follows:

- AI can be used to help with the formatting and layout of the overleaf document.
- AI can be used to find sources only when these sources are thoroughly checked to ensure they were valid.
- AI can be used to help with coding of certain elements.

¹ Risk Premium, <https://riskpremiumresearch.substack.com/p/joby-unit-economics>, Accessed on 23rd of June 2026

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Nomenclature

Abbreviations

ADS-B	Automatic Dependent Surveillance-Broadcast
AR	Aspect Ratio
ATC	Air Traffic Control
BEMT	Blade Element Momentum Theory
BMS	Battery Management System
CAD	Computer-Aided Design
CFD	Computational Fluid Dynamics
CFRP	Carbon Fibre Reinforced Polymer
CG	Centre of Gravity
CSC	Cell Supervision Circuit
DATCOM	Data Compendium
DSE	Design Synthesis Exercise
EASA	European Union Aviation Safety Agency
ECS	Environmental Control System
EKF	Extended Kalman Filter
eVTOL	Electric Vertical Take-Off and Landing
FAR	Federal Aviation Regulations
FEA	Finite Element Analysis
GEO	Geostationary Earth Orbit
GNSS	Global Navigation Satellite System
GPS	Global Positioning System
IMU	Inertial Measurement Unit
ISA	International Standard Atmosphere
LEO	Low Earth Orbit
Lidar	Light Detection and Ranging
MAC	Mean Aerodynamic Chord
MMOI	Mass Moments of Inertia
MTOW	Maximum Take-Off Weight
NACA	National Advisory Committee for Aeronautics
NASA	National Aeronautics and Space Administration
NCA	Nickel Cobalt Aluminium
NMC	Nickel Manganese Cobalt
NSGA-II	Non-dominated Sorting Genetic Algorithm II
OEI	One Engine Inoperative
PMS	Propulsion Management System
PPE	Personal Protective Equipment
RAMS	Reliability, Availability, Maintainability and Safety
ROI	Return on Investment

RPM	Revolutions Per Minute
SATCOM	Satellite Communication
SLSQP	Sequential Least Squares Programming
SWOT	Strengths, Weaknesses, Opportunities and Threats
TAWS	Terrain Avoidance and Warning System
TBD	To Be Determined
UAM	Urban Air Mobility
V&V	Verification and Validation
VCA	VTOL Capable Aircraft
VLM	Vortex Lattice Method

Greek Symbols

α	Angle of attack	rad
β	Sideslip angle	rad
η	Propulsive efficiency	-
η_{SF}	Safety factor	-
Γ	Dihedral angle	deg
Λ	Sweep angle	deg
λ	Taper ratio	-
ν	Poisson's ratio	-
ω_n	Undamped natural frequency	rad/s
ρ	Air density	kg/m ³
σ	Stress	Pa
$\sigma(\hat{r})$	Local blade solidity	-
σ_{allow}	Allowable stress	Pa
σ_{cr}	Critical buckling stress	Pa
τ	Time constant	s
ε_u	Ultimate strain	-
ζ	Damping ratio	-

Roman Symbols

$\bar{c}$	Mean aerodynamic chord	m
$\bar{V}'$	Tail volume coefficient	-
$\hat{r}$	Non-dimensional blade radius	-
a	Speed of sound	m/s
$a.c$	Aerodynamic centre	m
AR	Aspect ratio	-
B	Number of propeller blades	-
b	Wingspan	m
c	Chord length	m
C_ℓ	Rolling moment coefficient	-
C_D	Drag coefficient	-
c_d	Sectional drag coefficient	-

C_L	Lift coefficient	-	t_{TO}	Take-off time	s
c_l	Sectional lift coefficient	-	V	Flight velocity	m/s
C_M	Pitching moment coefficient	-	V_C	Cruise speed	m/s
C_n	Yawing moment coefficient	-	V_D	Dive speed	m/s
c_P	Power coefficient (BEMT)	-	v_{hover}	Hover velocity	m/s
c_T	Thrust coefficient (BEMT)	-	W	Weight	N
C_Y	Side force coefficient	-	Z	Elastic section modulus	m ³
C_{D_0}	Parasite drag coefficient	-	z	Vertical vortex position	m
C_{D_i}	Induced drag coefficient	-	Z_p	Plastic section modulus	m ³
$C_{D_{fp}}$	Flat plate drag coefficient	-			
$C_{L\alpha}$	Change in C_L with α	1/rad			
$C_{L_{max}}$	Maximum lift coefficient	-			
D	Drag	N			
E	Young's modulus	Pa			
e	Oswald factor	-			
g	Gravitational acceleration	m/s ²			
G_{IIc}	Mode-II interlaminar fracture toughness	J/m ²			
h	Centre-of-gravity position (fraction of MAC)	-			
h_0	Aerodynamic-centre position (fraction of MAC)	-			
I	Mass moment of inertia	kg m ²			
I_{xx}	Mass moment of inertia about the x -axis	kg m ²			
I_{yy}	Mass moment of inertia about the y -axis	kg m ²			
I_{zz}	Mass moment of inertia about the z -axis	kg m ²			
J	Advance ratio	-			
k	Drag polar	-			
L	Lift	N			
l'	Tail arm	m			
M	Mach number	-			
M_t	Propeller tip Mach number	-			
$M_{battery}$	Battery mass	kg			
n	Load factor	-			
n_p	Propeller RPM	rev/s			
N_{props}	Number of propellers	-			
P	Power	W			
q	Dynamic pressure	Pa			
$Q_{battery}$	Battery specific energy	J/kg			
R	Propeller (or fuselage) radius	m			
Re	Reynolds number	-			
S	Wing area	m ²			
S_{ref}	Total reference wing area	m ²			
S_{wet}	Wetted surface area	m ²			
T	Thrust	N			
T_2	Time to double amplitude	s			

Introduction

Every day, millions of people across Europe are lost to traffic. In Utrecht alone, drivers spend an average of 75 hours per year stuck in congestion, time that could be spent productively, or simply not wasted ¹. Beyond the personal cost, the environmental toll is severe: over 60% of Europe's CO₂ emissions from transport stem from personal car use, with the average vehicle carrying only 1.6 occupants ². The inefficiency is striking large as it is emission-heavy and vehicles are moving with barely more than one person per time.

The need for an alternative is clear. Urban air mobility (UAM) is emerging as a promising solution to ground-level congestion, offering a third dimension of travel that is entirely unconstrained by road infrastructure or geographic obstacles such as rivers, mountains and forests. Short-distance public transport already provides partial relief, but for inter-city distances in the range of 100-200 [km], viable zero-emission alternatives remain scarce. Current electric vertical take-off and landing (eVTOL) solutions for this range either require large ground infrastructure incompatible with dense urban environments, or lack the aerodynamic efficiency necessary for meaningful range.

This project addresses that gap directly by creating Vold-e. A silent, autonomous, sustainable eVTOL based on the Prandtl-plane configuration capable of transporting four passengers for at least 200 [km] at a cruise speed of 200 [km/h], while occupying no more ground space than nine parked cars. By combining electric propulsion, autonomous guidance, and a transformative wing geometry, this vehicle aims to redefine what urban and inter-city air mobility can look like: quiet, clean, and accessible. Vold-e as shown in Figure 6, is a foldable eVTOL. It's main goal is to act as an air taxi in the future market. However, as this is currently not feasible, Vold-e will act as a premium winter sport resort shuttle service in the Courchevel region.



Figure 6: *Vold-e during transition*

In this report the overall design process of Vold-e is shown. At first in Chapter 1, a market analysis is completed outlining the key economical aspects which must be followed to make Vold-e a feasible option. Next, the analysed mission is presented in Chapter 2 describing certain aspects which must be taken into consideration throughout the design process. Afterwards, in Chapter 3, the actual internal and exterior design of Vold-e is shown. The aerodynamic properties of Vold-e are described in Chapter 4 which is then followed by its power and propulsion characteristics in Chapter 5. In Chapter 6 key stability criterion are given and Vold-e's response to certain conditions are assessed. To ensure that Vold-e is feasible to work without a pilot, Chapter 7 describes how the autonomous control could be implemented. To assess the structural aspects of Vold-e Chapter 8 analyses all structural components including the Hinge required for the folding mechanism. To optimise certain parameters, an optimizer was made and its functionality explained in Chapter 9.

Once all design aspects were described, Vold-e's feasibility is analysed in Chapter 10. Following this, a verification and validation plan is carried out in Chapter 11. Next, a full risk and safety analysis is presented in Chapter 12. Next, a sustainability analysis is completed in Chapter 13. Closing off the feasibility analysis of Vold-e, is a thorough financial analysis shown in Chapter 14. Finally, a conclusion is written summarising all key aspects of the design process and the feasibility of Vold-e written in Chapter 15.

¹Euobserver: <https://euobserver.com/84530/drivers-in-eu-capital-spend-three-days-a-year-stuck-in-traffic/> Accessed on: April 22 2026

²European Parliament: <https://www.europarl.europa.eu/topics/en/article/20190313ST031218/co2-emissions-from-cars-facts-and-figures-infographics> Accessed on: April 22 2026

1 Market Analysis

This chapter focuses on providing a clear overview of how the current market is structured and how the future market is expected to operate. Initially a general overview is given in Section 1.1 of how the market is split into two large segments. Next the most vital stakeholders are analysed and their importance in the design is defined. This is shown in Section 1.2. Afterwards, the key competitors within the current market are provided in Section 1.3 and key aspects are compared. Finally to give a general overview of the key aspects in the market a general summary is provided in Section 1.4 to highlight important factors that must be considered during the design phase.

1.1. Market Overview

To correctly identify the stakeholders involved in the development and deployment of an eVTOL, it is essential to examine both the current state and future state of the urban air mobility (UAM) market. According to Porsche Consultancy, it is expected that eVTOLs will roll out into the market in two distinguished phases: an initial phase targeting premium travellers and basic functionality, followed by a broader phase in which eVTOLs become accessible to the general public [1]. This clear difference in the current market and the future market is also present in the estimation of the market size. Currently the UAM industry is valued at \$ 9.3 billion and is expected to grow to \$ 29.2 billion in 2030 ¹.

Current Market: Premium and Structured Travel

Currently in the UAM market, there are three primary use cases for eVTOLs: cargo handling, premium travel and emergency services.

Cargo handling was done quite early in the market deployment as this has less issues with safety and public acceptance of eVTOLs. This is supported by EASA survey data indicating that a significant portion of the public remains apprehensive about travelling as a passenger in an eVTOL [2]. Deploying eVTOLs in an uncrewed cargo role first serves a dual purpose: it generates early revenue for operators while simultaneously building public familiarity and trust in the technology.

Emergency services represent another high value early application. Critical scenarios such as organ transportation have to be done as fast as possible. eVTOLs would be significantly faster than alternatives as it has no issues with congestion, and therefore enable organ transportation to be completed 73% faster [2].

Finally, premium travel, while limited in customer volume during this phase, plays an important commercial role by establishing the possible operational framework of eVTOLs. A prime example are companies such as Blade, a service operating all over the world allowing for short transfers by helicopters. Blade has even mentioned that they will want to partner up with eVTOL providers in the future ². This signals that established operators see eVTOLs as a natural evolution of their existing services.

Future Market: Mass Market Adoption

As eVTOL technology matures and public acceptance of eVTOLs improve, the market is expected to transition into a second phase characterised by widespread public adoption. In this phase, eVTOLs would be used similar to ride sharing services do today. They will work on demand, be affordable and accessible to the general public. The customer base broadens significantly, moving beyond only premium travellers to include daily commuters seeking to bypass ground congestion, or overcome geographic barriers such as rivers, mountains and forests. The speed of this transition is, however, dependent on the continued reductions in noise levels and operational costs, both of which directly influence public willingness to adopt the technology. This underlines the importance of designing eVTOLs that are not only technically capable, but also compliant with European noise and emission regulations.

Overall, UAM is transitioning from an emerging concept to an anticipated standard in daily transportation. For the Netherlands, with its dense urban structure, there is a compelling opportunity to take a leading role in this transformation.

1.2. Stakeholders

Following the market analysis, the key stakeholders can be identified and categorised. Based on the EASA framework [2], stakeholders are grouped into four categories: The UAM industry, potential users, governments, public institutions & regulators, and indirectly affected third parties. In addition to this, an additional category was implemented concerning the investors of eVTOLs. Each category carries distinct motivations that directly impact the requirements imposed on the eVTOL design.

The **UAM industry** involves all parties directly involved in the development, manufacturing, operation and servicing of the eVTOL. Their primary motivation is commercial viability; creating profit and consequently advancing technologies.

¹ Grand View Research, <https://www.grandviewresearch.com/horizon/outlook/urban-air-mobility-uam-market-size/global>, Accessed on 13th of June 2026

² EVE: <https://www.eveairmobility.com/embraers-eve-to-deploy-electric-vertical-aircraft-for-blade-in-key-southern-florida-and-west-coast-markets/>. Accessed on 24th of April 2026

The **potential users** represent all parties that could directly utilise eVTOLs. Their main motivation is convenience, safety, comfort and affordability. These factors will ultimately determine whether eVTOLs are accepted in the broader public.

The **government, public institutions & regulators** consider the parties at different political levels (national and local). Their motivation is ensure the community is well supported. Certain parties within this group also care more about the environment and want to ensure that certain regulations set in place are followed.

Next, the **indirectly affected third parties** consider individuals, that do not actively use eVTOLs but are in the nearby environment. Their main motivation would most likely be related to noise levels and visual obstruction.

Finally, the **investors** consider the larger parties actually providing money for the development of eVTOLs. Their main motivation is the technical advancement, return of investment and all other matters concerning the financial budget.

An overview of all the key stakeholders which will be included in the project analysis are depicted in Figure 1.1. This figure provides examples of what type of stakeholders belong in each category.

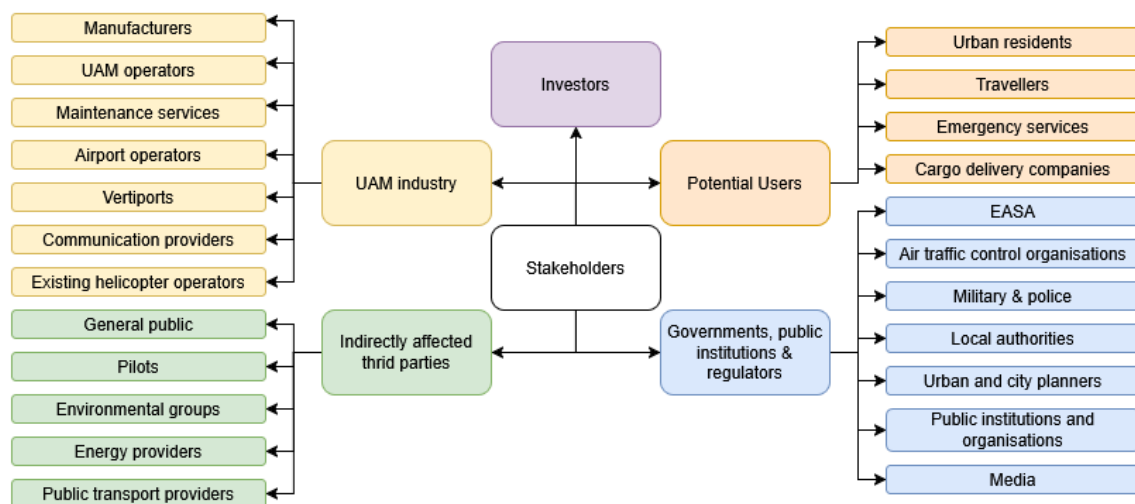


Figure 1.1: Stakeholder web

Once a list of stakeholders is given, an analysis can be completed of their importance in the project. This is done through creating a stakeholder map. Where stakeholders are ranked based on their interest in the project outcome, and their influence on the project outcome. This analysis is completed in Figure 1.3.

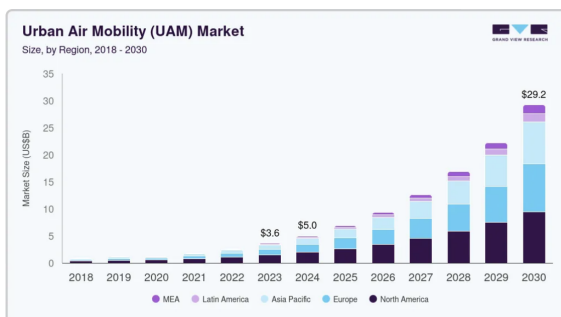


Figure 1.2: Expected UAM Market Growth³

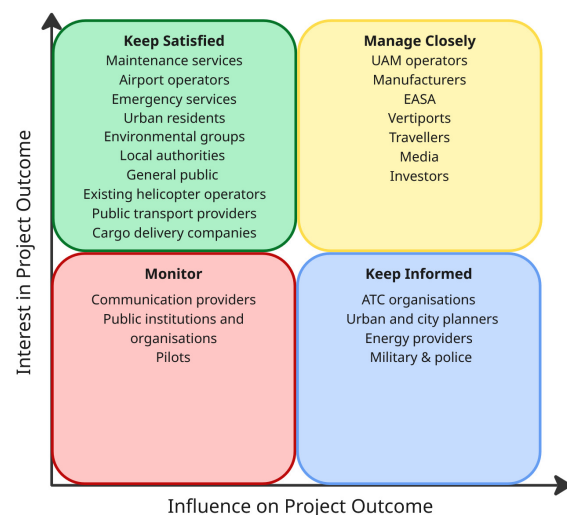


Figure 1.3: Stakeholder map

³Grand View Research: <https://www.grandviewresearch.com/industry-analysis/urban-air-mobility-market-report>, Accessed on 29th of April 2026

1.3. Competitors

In order to further inspect the market gap of eVTOLs one must take a look at the current competitors in the field to see what aspects are still missing. As eVTOL is still an emerging concept, there are only few competitors with a proper working design already in Table 1.1, an overview is given of certain main players in the eVTOL industry with certain parameters compared. Also, a helicopter is added to the table as currently this is the largest competitor. In Table 1.2 a few additional competitors are given that have a more different structure. This gives a large variety of inspiration to begin the design phase.

Table 1.1: Main Alternatives for Comparison

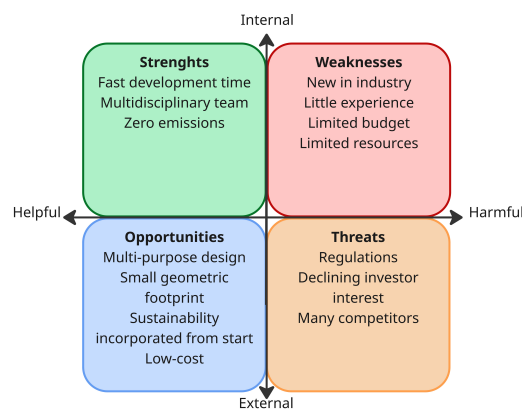
	Joby [3], [4] ¹  <i>Joby S4 aircraft ²</i>	E20 (Tcab Tech) [3]  <i>Tcab Tech E20 aircraft ³</i>	Vertiiia (AMSL Aero) ⁴  <i>Vertiiia AMSL aircraft ⁵</i>	Transwing P4 (PtereoDynamics)  <i>Transwing P4 aircraft ⁶</i>	ACH125 (Airbus)  <i>Airbus ACH125 ⁷</i>
Country	USA	China	Australia	USA	France
Reason for Choosing	Most information available; most similar mission profile	Successful transition phase; tilt rotors at wing tips	Prandtl-plane with full tilt rotor concept	Transformative folding wing; no passenger payload	Conventional helicopter benchmark for comparison
Cost [\$]	2 million	960,000	4.6 million	-	3 million
Lift Method	6 tilt rotors with 5 blades each	4 tilt rotors + 2 lift rotors, fixed wing	Prandtl-plane, 8 full tilt rotors	Transformative folding wing	Single main rotor + tail rotor
Tail Shape	V-tail	Conventional	V-shape	Inverted V-tail	Conventional tail rotor
Piloting	Onboard pilot	Onboard pilot	Onboard pilot	Autonomous	Onboard pilot
Charging Method	4 lithium-ion polymer battery packs	800V, ~20 min, 4 battery packs integrated into wing	Hydrogen, ~10 min	-	Jet fuel (Safran Arriel 2D turboshaft)
Payload [kg]	450	450	500	6.8	~700
MTOW [kg]	2,400	-	-	41	2,370
Passengers	4	4	4	0	4–5
Range [km]	320	200	1,000	51	630
Max Speed [km/h]	-	320	300	184	252
Cruise Speed [km/h]	320	-	-	110	253
Noise [dB]	45 (cruise, 500m); 65 (landing, 100m)	-	65–70	-	~85–90
Ground Footprint	7.5 × 14.05 [m]	-	-	2.1 × 1.3 [m]	Rotor diameter: 10.69 [m]
Cruise Lift coefficient	0.52	-	-	-	-
Material used	Carbon-fiber composite	Carbon-fiber composite	-	-	Aluminium alloy & composites
Notes	-	-	-	Dihedrally folding wing system	Fossil fuel powered; used as performance baseline

Table 1.2: Reference Alternatives for Comparison

	EH216-S (EHang) [3]  <i>EHang EH216-S</i> ⁸	Lilium Jet [3]  <i>Lilium Jet</i> ⁹	Elytron 4s (NASA) [5]  <i>NASA Elytron 4s</i> ¹⁰	Cary All (Autoflight) [11]  <i>Autoflight Cary All</i> ¹²
Country	China	Germany	USA	China
Reason for Choosing	Can be inspected for efficiencies in vertical take-off	Different approach using propellers embedded in wing	Detailed aerodynamic data for Prandtl concept	Certified eVTOL in commercial operation
Cost	340,000	-	-	-
Lift Method	Multi-rotor (16 rotors)	Tilt wing & tilt canard	Prandtl-plane	Fixed lift rotors + glider wing
Tail Shape	None	Front canard	V-shape	-
Piloting	Autonomous	Onboard pilot	Piloted	Autonomous
Payload [kg]	220	-	-	400
MTOW [kg]	620	-	-	-
Passengers	2	6	3	-
Range [km]	30	175	1,600	250
Max Speed [km/h]	130	248	-	200 (cruise)
Ground Footprint	H: 1.93 m, W: 5.73 m	Wingspan: 13.9 m	-	-
Notes	Aerial tourism use	Went bankrupt	-	Fully certified

SWOT Analysis

To evaluate the strategic position of Vold-e within the UAM market, a SWOT analysis is presented in Figure 1.4. This framework identifies the internal strengths and weaknesses and the external opportunities and threats that may influence its market viability. Together these four parts provide a structured basis for understanding where the design offers a competitive advantage and where development efforts should be focused. Each entry is explained in Table 1.3.

**Figure 1.4:** SWOT Analysis

¹Joby Aviation: <https://www.jobyaviation.com>, Accessed on 29th of April 2026

²eVTOL news: <https://evtol.news/jobby-s4>, Accessed on 29th of April 2026

³eVTOL news: <https://evtol.news/tcab-tech-e20>, Accessed on 29th of April 2026

⁴AMSL Aero: <https://www.amslaero.com/our-product>, Accessed on 29th of April 2026

⁵Financial Review: <https://longyield.substack.com/p/the-last-gate-joby-aviation-clears>, Accessed on 8th of May 2026

⁶Pteroo Dynamics: <https://evtol.news/pteroodynamics-transwing/>, Accessed on 29th of April 2026

⁷Airbus ACH125: <https://www.ach.airbus.com/en/ach-offerings/helicopters/ach125>, Accessed on 29th of April 2026

⁸eVTOL News: <https://evtol.news/ehang-216/>, Accessed on 29th of April 2026

⁹eVTOL News: <https://evtol.news/lilium/>, Accessed on 29th of April 2026

¹⁰eVTOL News: <https://evtol.news/elytron-aircraft-elytron-4s>, Accessed on 29th of April 2026

¹¹Autoflight: <https://www.autoflight.com/en/>, Accessed on 29th of April 2026

¹²eVTOL News: <https://evtol.news/autoflight-carryall>, Accessed on 29th of April 2026

Table 1.3: *SWOT analysis rationale*

Entry	Rationale
Strength: Fast development time	Given the development time of 10 weeks, the feasibility of the design will quickly be assessed and no unnecessary resources are wasted.
Strength: Multidisciplinary team	The team consists of people with interests in different fields within aerospace, meaning that design can be approached from many different viewpoints.
Strength: Zero emissions	The envisioned vehicle will emit no CO ₂ . This is quite important in modern day society ¹ .
Weakness: New in industry	As this is the team's first project, there is no brand recognition within the eVTOL market. This leaves the team at a commercial disadvantage in comparison to other well known competitors such as Airbus and Joby.
Weakness: Little experience	This is the first project at this scale that the team will take on. Therefore, there is a lack of experience.
Weakness: Limited budget	The budget available to the team is very limited. This makes for little prototyping possibilities and scaled tests.
Weakness: Limited resources	The resources are limited in terms of time and available materials. This too makes prototyping and scaled tests hard to perform.
Opportunity: Multi-purpose design	The vehicle will be designed to take on a multitude of different missions in order to enlarge its economic viability and industry relevance.
Opportunity: Small geometric footprint	The vehicle will be designed such to have a small footprint when on ground. This increases its ease of use in tight urban areas.
Opportunity: Sustainability incorporated from the start	Sustainability forms a vital pillar in the design of the vehicle. This is deemed to ease public acceptance.
Opportunity: Low cost	The vehicle will be designed with a maximum acquisition cost of 4 million euros in mind. Given Table 1.1, this is deemed a competitive cost.
Threat: Regulations	The vehicle must to comply with European regulations, that might impose constraints on the design.
Threat: Declining investor interest	Research from Roland Berger and the DLR [6] has shown that investor interest has been declining. This poses the threat of not being able to get enough cash to design and build the vehicle.
Threat: Many competitors	There are quite a lot of competitors in the market. Some have been listed in Table 1.1, but this list is far from exhaustive. The envisioned vehicle must be sufficiently better than the competition to ensure commercial viability.

1.4. Discussion

Following the market analysis and SWOT assessment certain key findings are critical to the successful market entry and long term viability of the eVTOL. In addition to this, these key aspects are also needed to outline how to outperform the competitors analysed in Table 1.1.

- **Autonomous flight:** According to EASA, the general public is not extremely confident in flying in an eVTOL themselves if there is no pilot present [2]. This means that there must be visible safety systems within the cockpit to reassure the passengers or implement an education strategy to demonstrate how reliable and safe autonomous flight systems are.
- **Noise levels:** Noise levels are consistently identified as one of the primary barriers to public acceptance. As helicopters in general are considered as quite intrusive, careful consideration must be put into place to ensure to truly minimize the sound not only inside the cockpit but also the sound during take off and landing.
- **Geographical factors:** A key competitive advantage would be to ensure that the eVTOL can operate in remote environments, including areas separated by natural barriers, such as mountains. Ensuring that the eVTOL can take off and land in remote locations significantly broadens the addressable market and differentiates the design from other competitors that mainly focus on urban environments.

¹ Intertraffic, <https://www.intertraffic.com/news/sustainability/a-positive-outlook-for-sustainable-mobility-in-2025>, Accessed: 11-05-2026

2 Mission Overview & Operations

This chapter is focused on giving an overview of the mission which will be completed by Vold-e and inspecting the operational and functional side of the eVTOL. At first the selected mission is discussed in Section 2.1. This outlines the structure of the mission and analysis its feasibility in the current market. Next, the operational side of this mission is discussed in Section 2.2 and how it will change over the changing market. Finally, the functional aspects of Vold-e are depicted in Section 2.3 through a series of functional flow diagrams. Finally, Section 2.4 summarises the operational requirements and limitations of the proposed mission with regard to its future deployment.

2.1. Mission Overview

This main goal of this project is to design a fully autonomous eVTOL that is able to serve as an air taxi between regional city pairs in the future market. To ensure its market viability in the future, our designed eVTOL (Vold-e) must be able to fly the route from Nice to Courchevel as it has extremely challenging geographical barriers. This challenging route designs Vold-e to the extreme, and ensures that Vold-e is able to fly everywhere on earth, and hence fully fulfil its purpose as an urban air taxi. Therefore with this in mind, the mission statement for Vold-e is as follows:

The goal of this project is to design an eVTOL that redefines air mobility by enabling autonomous, near silent, sustainable transport across cities and geographic barriers.

In order for Vold-e to be a viable option in the current market, it was decided to use Vold-e as a premium ski resort airport shuttle in the Courchevel area. This service qualifies as a viable option for the current market as premium travel, yet its point to point, on demand structure closely mirrors future possible operations. With this in mind, the Courchevel area with operations spanning all major cities within a 200 km radius is selected. The eVTOL must fly through mountain ranges, be able to withstand the high winds present by the sea, and fly through a busy city. It is therefore considered a representative worst case scenario, if the eVTOL can reliably complete this mission it is capable of serving all nearby cities within a 200 km radius as illustrated in Figure 2.1 ¹.

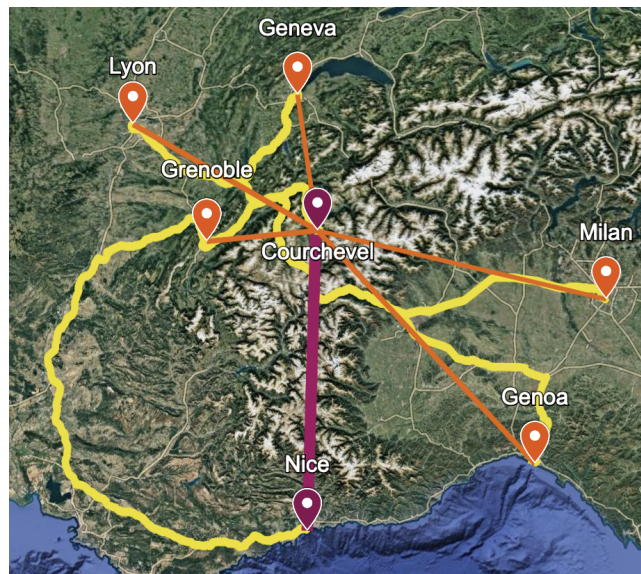


Figure 2.1: Range that can be covered by the designed eVTOL. Yellow lines indicate the current path travelled by ground transportation, and orange and red lines indicate path travelled with an eVTOL.

In order to ensure that the selected route is feasible, and that sufficient demand exists, an analysis was completed of all other methods of transportation for this route. Courchevel is located inside Les Trois Vallées which is the largest ski area in the world ². Meaning transportation to Courchevel is needed. In Table 2.1 an overview is given of all the existing transportation methods travelling between Nice and Courchevel and comparing them to an eVTOL. This clearly shows that although a helicopter is just as fast as an eVTOL, it is more expensive and is worse for the environment when looking at CO₂ emissions. When looking at a taxi, the time taken for this route is extremely long in comparison to other available methods deeming an eVTOL as a viable alternative.

In Table 2.1, FH means flight hour and shows that the estimated operational cost per person is roughly € 340 in the current market.

¹ Google Earth, https://earth.google.com/web/@45.46654003,6.33110432,1723.41898881a,1272781.7847003d,35y,344.40227446h,0t,0r/data=CgRCaggBMigKJgokCiAxWXAwc3R2Q1ZheGh2RG5uRWF0VHh6dGE2YVRDVV9JQSAC0gMKATBCAggASggI30iFxmMQAQ?authuser=0&utm_source=mymaps, Accessed on 29th of May 2026

² Igls Ski, <https://www.iglski.com/ski-resorts/ski-areas/three-valleys>, Accessed on 12th of May 2026

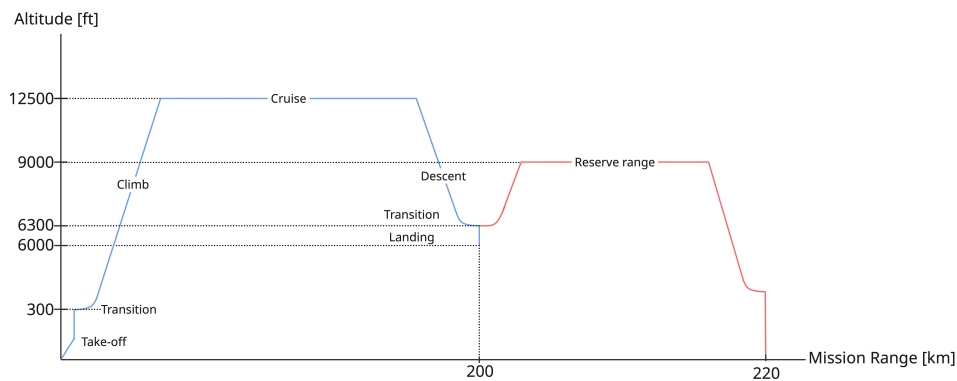
Table 2.1: Existing Premium Transportation Options in Courchevel

	eVTOL	Taxi ⁵	RoboTaxi ⁶	Helicopter ⁷
Cost per person [€]	340	480	530	780
Distance travelled [km]	200	590	590	200
CO ₂ emissions [kg]	0	100 ⁸	0	340 ⁹
Time [hours]	~ 1	6	6	1
Number of Passengers [-]	4	4	3	6
Price per km per passenger [€]	1.7	0.8	0.9	3.9

The ticket price per person was calculated in Section 14.3 to account for a profit and a realistic return on investment. This shows that looking at all other alternatives, the eVTOL is the cheapest and takes the shortest amount of time.

2.1.1. Mission Profile

Based on the mission from Nice airport to Courchevel, a mission profile has been developed. This mission profile is shown in Figure 2.2, and schematically shows the mission the envisioned eVTOL will need to perform. The cruise altitude is based on the highest mountains in the region which are approximately 10000 feet tall¹¹. Taking into account a safe margin of 2500 feet¹² of clearance above the mountains. Another reason for this chosen altitude, was there was no other safe flyable route between Nice and Courchevel within a 200 [km] range that does not travel across the mountain range with the highest mountains. In addition to this, it is found that Courchevel rests at an altitude of 6000 feet¹³. The reserve altitude is determined based on safely being able to clear the mountains around Courchevel, transition and fly to another destination. Therefore a altitude of 9000 ft is used.

**Figure 2.2:** Mission Profile for Route Between Courchevel and Nice Including Reserve

It should be emphasised that the value of 12500 feet is a preliminary estimation of the cruise altitude, adopted for the present sizing case and does not represent the final operational cruise altitude of the aircraft. The cruise altitude will in practice vary on a mission-by-mission basis with the origin–destination pair, the underlying terrain and obstacle environment, the prevailing airspace structure along the route, weather conditions, and noise-abatement requirements imposed by local authorities.

2.2. Operations and Logistics

In order to assess the operational side of performing the intended mission, a diagram is made in Figure 2.3 outlining the key logistical elements involved in a mission.

It must be noted that the operations and infrastructure involved in the mission is highly dependent on the phase of the market in which the mission is carried out. As mentioned in Chapter 1, the UAM market is expected to increase drastically in size in 2035 onwards. The operational and logistics diagram is therefore split into two scenarios, one being a mission carried out in the current market, and the second being a mission carried out in the future market. Currently, the carried out mission consists only have flights within the area of Courchevel and acts as a premium travel service. In the future market, there will be hubs in all large cities in Europe such as Amsterdam, Bern and London.

⁵Taxi Courchevel, <https://taxicourchevel.com>, Accessed on 12th of May 2026

⁶It is assumed, that the RoboTaxi is a Tesla Y model, and has the same costs as taxis currently in the USA, <https://rideobi.com/teslaq1/>, Accessed 27th of May 2026

⁷Hoper, <https://flyhoper.com/>, Accessed 12th of May 2026

⁸Vehicle score, <https://vehiclescore.co.uk/co2-emissions-check/mercedes-benz/v-class>, Accessed on 12th of May 2026

⁹Stuff, <https://www.stuff.co.nz/environment/129935388/double-standard-on-climate-change-helicopters-allowed-but-car-use-needs-to-be-cut>, Accessed on 12th of May 2026

¹¹Google Earth, <https://www.google.nl/intl/nl/earth/index.html>, Accessed 13-05-2026

¹²UK Civil Aviation Authority, <https://www.caa.co.uk/general-aviation/safety-topics/mountain-flying/>, Accessed 13-05-2026

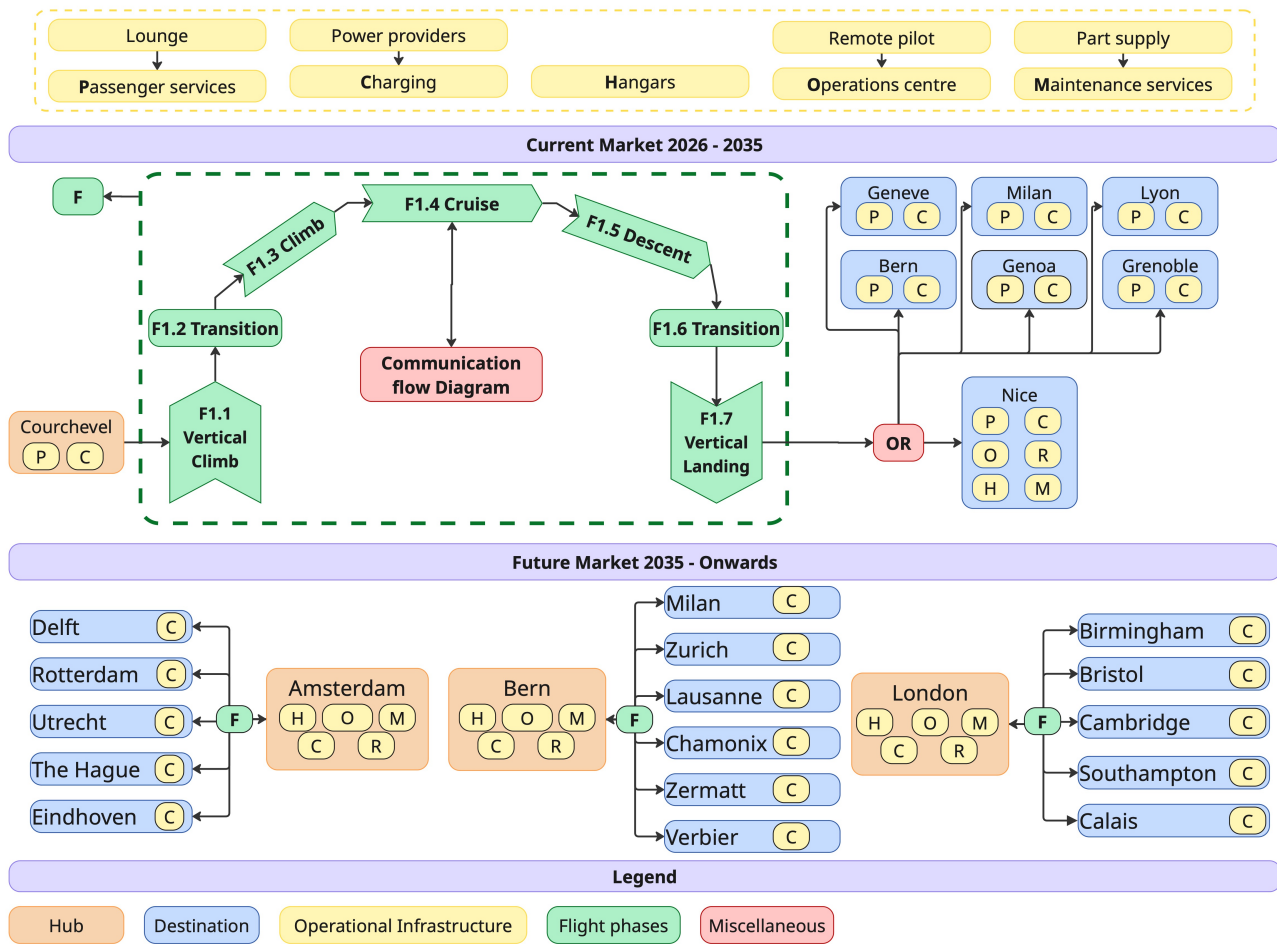


Figure 2.3: Operational and Logistics diagram

These hubs act as a central point for the eVTOLs. In these hubs, a remote pilot is placed to control all the eVTOLs around that hub. At each destination it is required to have a charging centre to allow for Vold-e to later continue its journey to a different destination.

In addition to the infrastructure required to make the desired missions feasible, a communication diagram is created in Figure 2.4 to show all the communication paths which are required to make the mission work. These paths are divided into three separate sections: automatic communication paths, manual communication paths and future automatic communication paths.

The communication path begins with the user, who manually contacts an operator to request an eVTOL. The operator then automatically assigns an eVTOL to the user based on proximity and availability. The operator is able to automatically communicate to the eVTOL to go to the user through either sending information through a LEO satellite or a cellular tower because research [7] has shown that the use of cellular towers is a promising way of connecting eVTOLs as it has a large coverage and low latency. However, current cellular towers are not enough to connect eVTOLs as these are optimised for on ground use. Therefore, the use of LEO satellites enables coverage for inter-city trips, near continuous connectivity and allows for high data rates. It was decided to use the LEO satellite as the main communication link between the vehicle and the operator, whereas the cellular tower will be used as a backup if satellite communication fails. The eVTOLs position and altitude are determined using GEO satellite data, which is continuously and automatically available to the operators.

Throughout the flight, the eVTOL must constantly be aware of surrounding airborne vehicles and likewise, other airborne vehicles must also be aware of nearby eVTOLs. This is achieved through an automatic link minimising the risk of mid air collisions. As an additional safety measure, operating centres are manually able to take over the eVTOL, again sending information through either LEO satellites or cellular towers.

For safety and regulatory compliance, ATC must always be informed about the whereabouts of the eVTOL particularly during airport transfers which is done through an automatic link. In addition to this, the ATC requires the possibility to manually contact the vehicle¹³. This is needed in case of emergencies, as well when transferring at airports. ATC is also able to manually contact the operators if needed.

¹³Airport Technology, <https://www.airport-technology.com/features/automation-air-traffic-control/>, Accessed on 18th of May 2026

Finally the eVTOL must interface with the U-Space framework. U-space is similar to the ATC but specifically designed for unmanned vehicles. According to EASA [8], U-Space will be used to define rules for passenger operations of vehicles, however these rules are not currently made. Once in place, the eVTOL will be required to communicate automatically with U-Space operators.

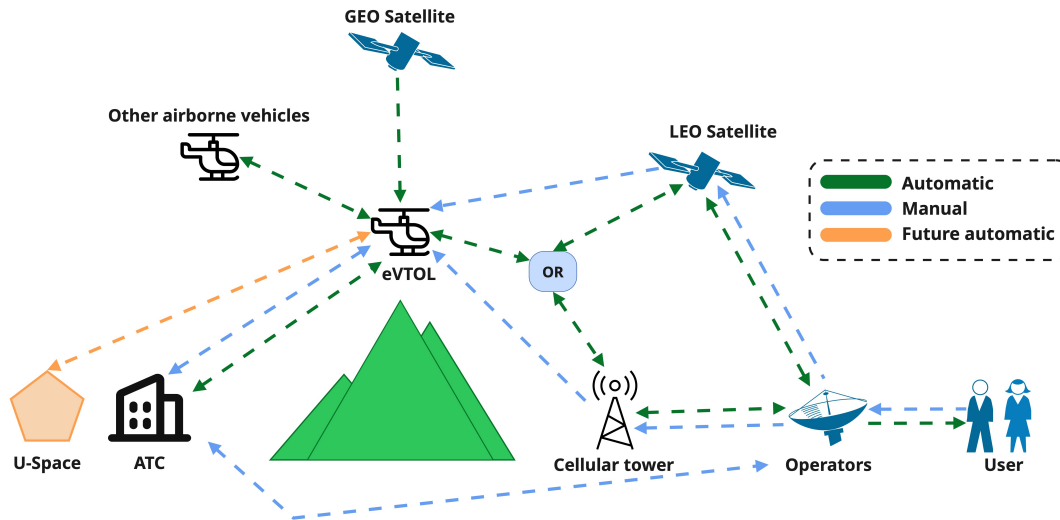


Figure 2.4: Communication Paths for the Designed eVTOL

2.3. Functional Diagrams

To assess how Vold-e will actually carry out the desired mission, a functional flow diagram and functional breakdown diagram are created to outline all the functions which must be carried out by certain systems. These diagrams are added in Appendix A.

2.4. Discussion

Overall, this section aimed to outline the key operational factors involved in the chosen mission. It creates a thorough overview of the logistical infrastructure needed for the eVTOL to be implemented into the current and future market. Certain limitations exist however in this overview, specifically relating to how the future eVTOL market will fully be developed. Hence, in the future to fully assess Vold-e's feasibility in the market an analysis in the future market must be completed to ensure that it is a profitable business as well as determining any additional infrastructure that might be needed to be implemented into the U-space framework.

3 Final Configuration and System Architecture

This chapter focuses on outlining the key design aspects of the final chosen design. Initially the final configuration is presented in Section 3.1 consisting of showing the internal structure key dimensions as well as a closer look at the folding mechanism required. Finally a data handling architecture is drawn in Section 3.2 to analyse the internal components required for communicational flow.

3.1. Final Configuration

This section presents the final physical configuration of Vold-e. First, the overall dimensions of Vold-e are shown in Subsection 3.1.1. Next, Subsection 3.1.2 outlines the internal fuselage layout, including passenger and battery integration, then Subsection 3.1.3 describes the folding mechanism that enables the Prandtl-plane wing system to meet the ground-footprint requirement.

3.1.1. External Layout

The external layout of the final design in VTOL mode can be seen in Figure 3.2 and in cruise mode in Figure 3.1. It consists of the aircraft being seen from side and up with all the necessary dimensions stated in meters.

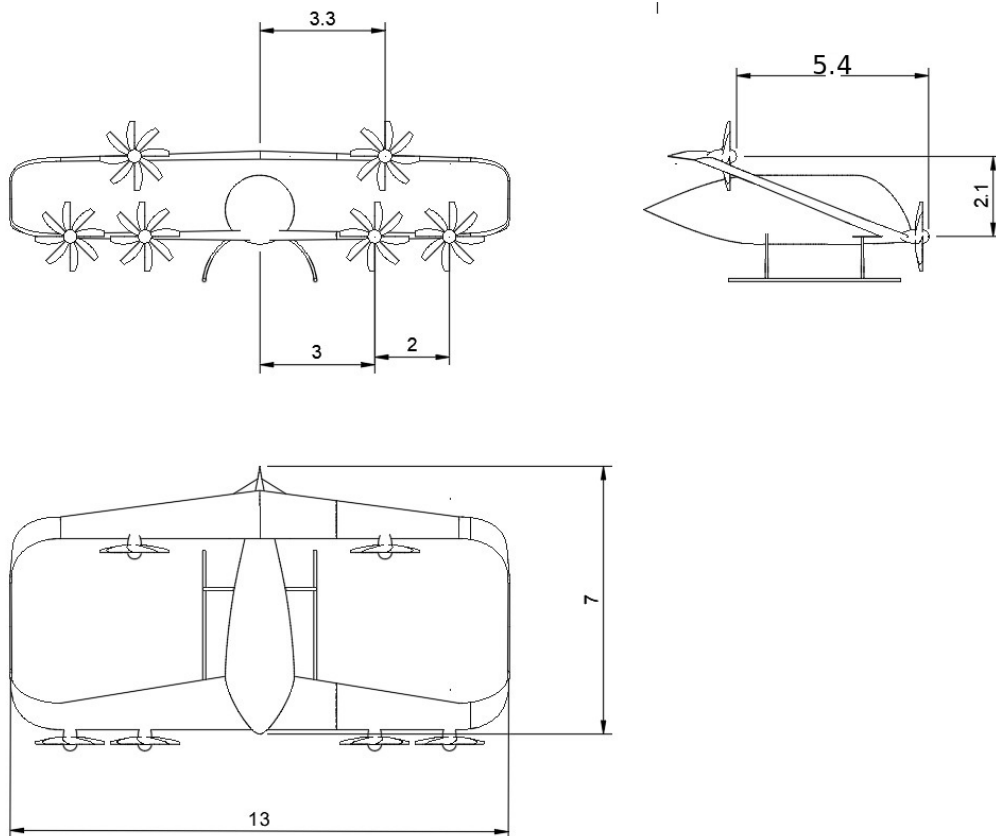


Figure 3.1: External layout of the aircraft in cruise configuration

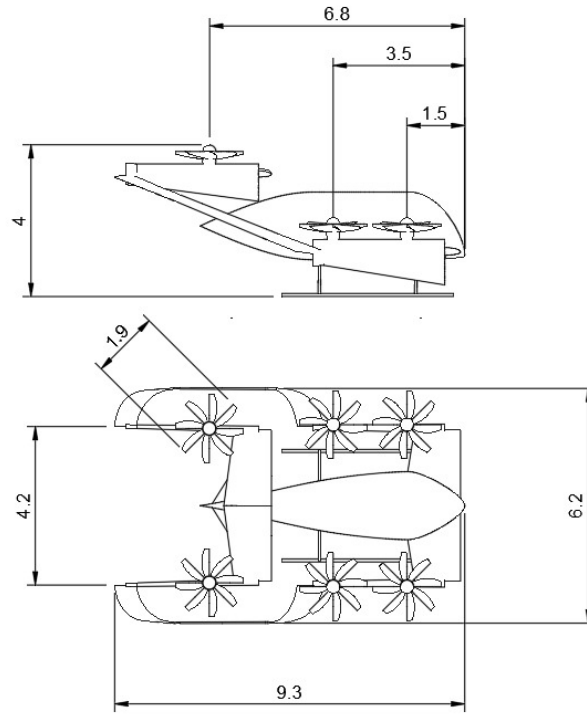


Figure 3.2: External layout of the aircraft in VTOL configuration

3.1.2. Internal Layout

The internal structure of the fuselage is presented in Figure 3.3. This image aims to portray the sizes of the chairs within the fuselage, the battery, the sliding mechanism and the overall room available inside. All dimensions and calculates are based on examples performed in [9].

It was estimated that the inner diameter of the fuselage is 1.7 [m]. This leaves for a thickness of 0.05 [m] to ensure the cabin remained pressurised as well as to decrease the noise levels inside the fuselage. In addition to this, the battery compartment including the sliding mechanism was placed below the cabin floor and the battery was sized to have an area of 0.27 [m²].

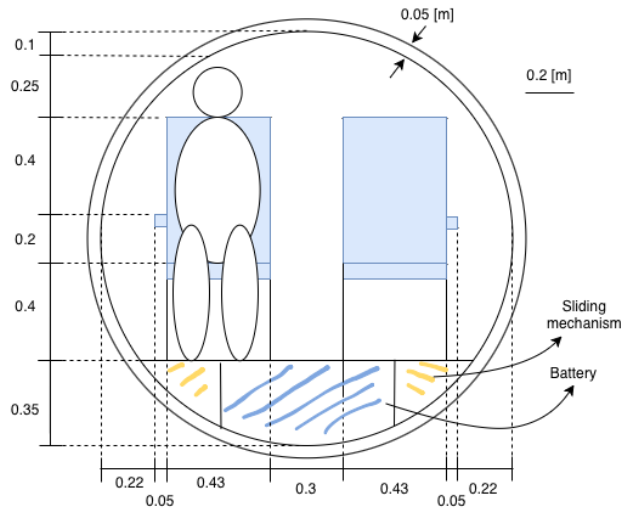


Figure 3.3: Cross section of the Fuselage

3.1.3. Folding Mechanism

In order to comply with the ground footprint requirements, without sacrificing aerodynamic performance, the configuration is equipped with folding mechanism for the wings. This subsection will go more in depth on how this folding is achieved. As it could be seen in Figure 3.4 there is some significant difference in the ground footprint between the folded and unfolded configurations.

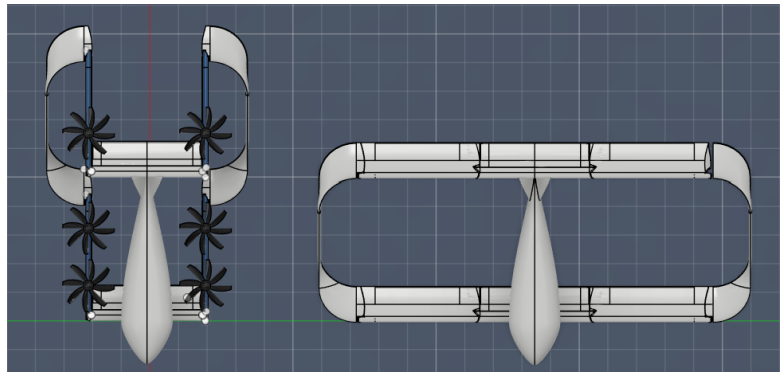


Figure 3.4: Size comparison between the folded and unfolded preliminary configurations.

In order to achieve this type of folding every wing has a hinge connection at the tip and close to the root as in Figure 3.5. This dual hinge design allows the wings to fold at the root and to stay attached to the vertical winglet. In order for these hinges to work simultaneously, without causing stress in the wings and the winglets, the axes of rotation of all the hinges need to be parallel to one another. This means they should all have the same angle with respect to the wing. Additionally, the virtual lines connecting the hinges, shown in Figure 3.5, must be parallel and opposite lines should have the same length.

If the motion is looked at from the direction in which the hinges axes point it will look like folding a rectangular frame with hinged vertices. All the sides will stay parallel during the whole range of motion. This ensures the two wings fold simultaneously without conflicts between the joints. This connection was validated in Fusion 360. After recreating the joint setup, the motion was free of conflicts, meaning the geometry works in the way it was intended.

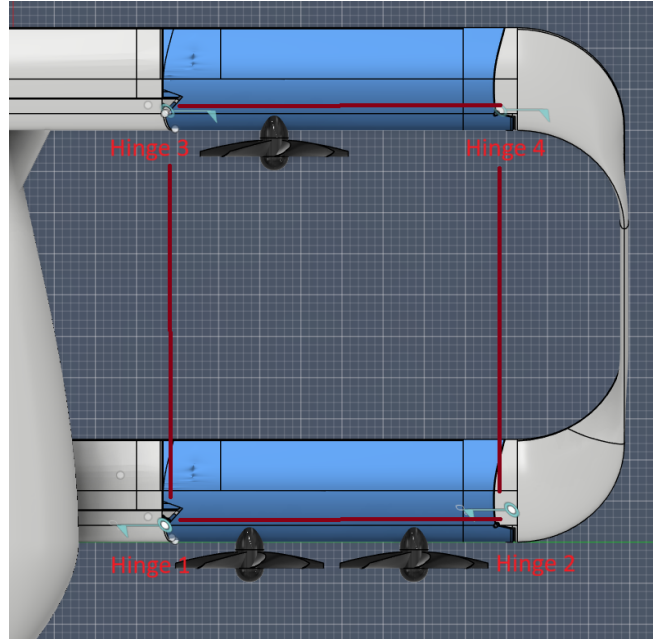


Figure 3.5: Hinge placement

The spanwise locations of the inner hinges are at 2 meters and of the outer ones at 6 meters with respect to the root chord. The chordwise locations of both the inner and outer hinges are at the same distance from the leading edge of the wing. This distance equals a quarter of the chord at the location of the inner hinges.

The angles of the hinge axes could be represented easily with the help of the cube in Figure 3.6.

The orange line represents the hinge axis, while the coordinate system axes are aligned with the fuselage and the span of the wing. The rotation around this axis allows the wings to fold parallel to the fuselage and point the engines in the correct direction for the cruise and vtol modes as could be seen in Figure 3.7. The transformation done by rotating around this axis is explained in more detail in Subsection 8.2.1.

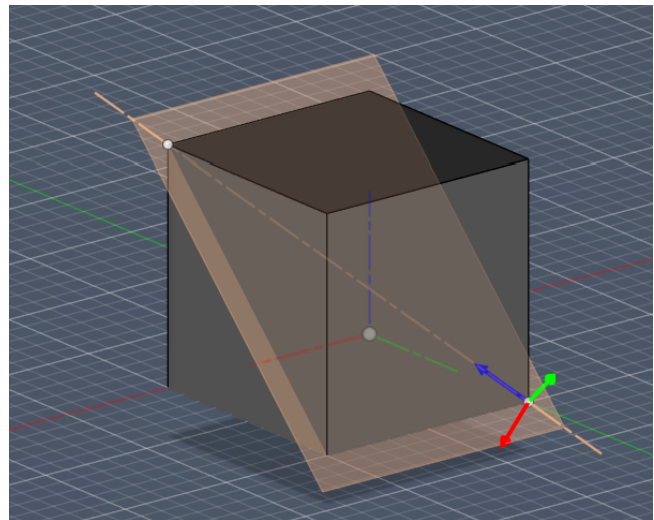
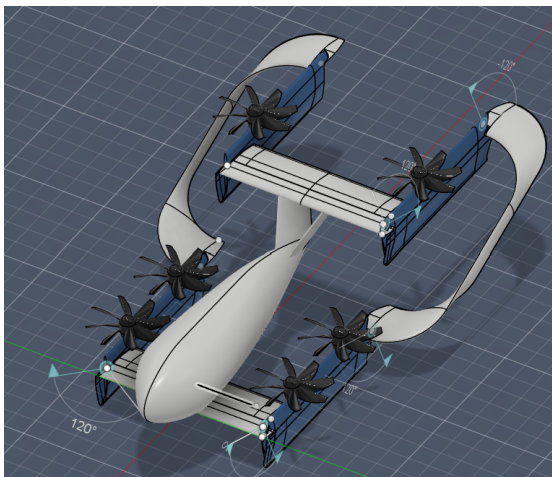
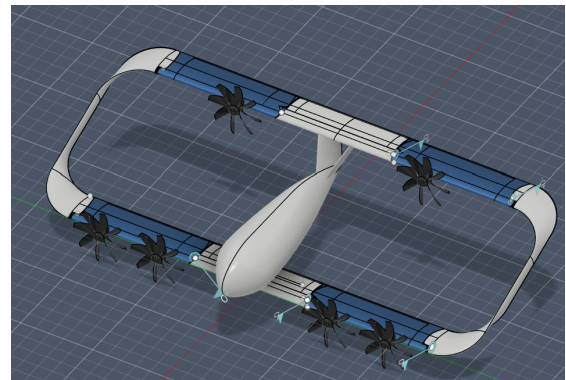


Figure 3.6: Representation of hinge axis orientation with cube geometry. (red - X-axis, green - Y-axis, blue - Z-axis)



(a) Evtol 3D model in vtol configuration.



(b) Evtol 3D model in cruise configuration.

Figure 3.7: 3D models of the evtol in the two modes of operation.

3.2. Data Handling Architecture

To establish how data flows through the system, a data handling diagram is created in Figure 3.8. In this diagram, all devices that provide data are grouped in two categories: attitude & navigation, responsible for the flight of the aircraft, and avoidance, which makes sure that Vold-e doesn't hit anything. The GPS, IMU, pitot tube, barometer, angle of attack sensor and external cameras are part of the attitude & navigation group. The radar and TWS are used for avoidance, where the TAWS is used for terrain avoidance and the radar used for the avoidance of other vehicles that do not have an ADS-B transmitter on board [10, 11]. The LiDAR returns distances to nearby objects¹ and is envisioned to be used both for navigation around cities and avoidance of buildings and other nearby obstacles. The ADS-B system in turn is used to send information such as the vehicle's position and velocity to other vehicles and ground stations. In addition, ADS-B also allows the vehicle to receive information from other vehicles and weather and traffic information from ground stations². The data rate of ADS-B transmissions is 1 Mbit/s³. In the diagram, it is assumed that the flight management computer handles all computing. In order to make sure that the aircraft is controllable, even in case of emergency, a remote pilot in the operations centre, is able to take control over the aircraft. In order to make this happen, a camera is also added to the aircraft. All necessary data is in turn sent to the ground station, via either the LEO satellite, or cellular tower. This data link is assumed to be one of the largest in terms of data rate, reaching 100 Mb/s [7]. In addition, the flight computers will also send system health data to the operations centre, to monitor the eVTOLs status and detect any faults.

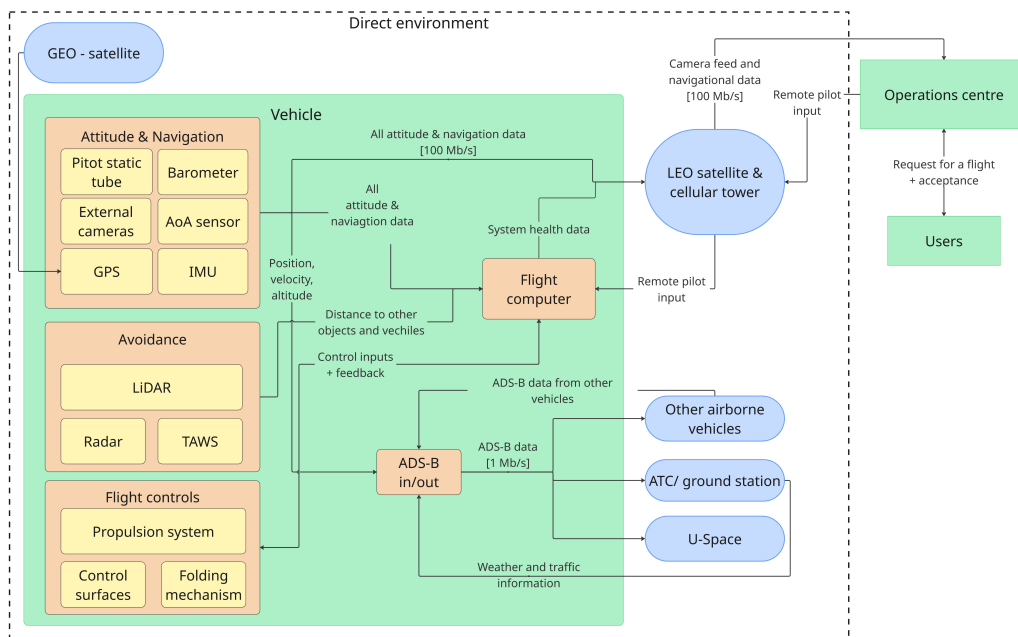


Figure 3.8: Internal communication flow diagram

¹IBM, <https://www.ibm.com/think/topics/lidar>, Accessed 18-05-2026

²FAA, https://www.faa.gov/air_traffic/technology/equipadsb/capabilities/ins_outs, Accessed 18-05-2026

³MATLAB, <https://www.mathworks.com/help/comm/ug/airplane-tracking-using-ads-b-signals.html>, Accessed 18-05-2026

4 Aerodynamics

This chapter presents the aerodynamic design and sizing of Vold-e. The objective is to define the main aerodynamic geometry of the aircraft and to justify the design choices that influence cruise efficiency, stability, integration, and overall vehicle performance. First, Section 4.1 outlines the planform design and defines the main wing geometric parameters. Section 4.2 then presents the airfoil selection process, including the trade-off setup and results, and a verification of the airfoil analysis method. Section 4.3 discusses the sizing and airfoil choice for the vertical tail. The box-wing layout is addressed in Section 4.4, where both the horizontal and vertical gaps between the front and aft wings are determined. Section 4.5 presents the fuselage selection process and assesses the aerodynamic characteristics of the selected fuselage geometry. Finally, Section 4.6 discusses the main limitations of the preliminary aerodynamic design methods and identifies key areas for future validation.

4.1. Planform Design

The aerodynamic design and sizing of the eVTOL includes the determination of the planform design parameters. These parameters with their chosen value and their rationale are presented in Table 4.1:

Table 4.1: *Planform design parameters*

Parameter	Value	Rationale
Leading edge sweep	0 deg	According to Raymer [12], leading edge sweep is determined by, amongst others, the angle of the Mach cone. Since the full flight will be subsonic, no sweep angle is necessary for aerodynamic purposes.
Dihedral angle	0 deg	This is determined from geometrical constraints. As the wings need to be folded, having a dihedral angle is not possible.
Twist angle	0 deg	A wing twist angle was not implemented to avoid additional complexity in the folding mechanism and to avoid integration constraints with the propellers.
Taper ratio (λ)	0.45	According to Raymer [12], for unswept wings, a taper ratio of 0.45 is the closest to an elliptical lift distribution.

4.2. Airfoil Selection

Having determined the wing planform design parameters, airfoil design parameters should be determined by selecting an airfoil. This airfoil trade-off is completed in the following few subsections.

4.2.1. Trade-Off Setup

First, the criteria used for the trade-off will be defined, together with the assigned weight and rationale in Table 4.2. The trade-off is split into 2 flight regimes; cruise and transition. The criteria are assessed in the flight regime in which they most relevant. In addition to the criteria mentioned in the table, the pitching moment coefficient was also investigated as a possible criterion. However, at this stage of the design, it was not possible to know what value of the pitching moment coefficient would be desirable. As such, this criterion was excluded from the trade-off, while it still is an important airfoil parameter.

Table 4.2: *Explanation of trade off criteria and their assigned weights*

Criteria	Flight regime	Weight	Rationale
Lift to drag ratio	Cruise	0.35	This is deemed the most important as it will lay the basis for the aerodynamic efficiency of the eVTOL and as such determine the energy needed during the flight.
Maximum lift coefficient	Transition	0.25	This is deemed important for determining how slow the aircraft is able to fly; i.e. it determines the stall speed. As such, it will also determine the speed at which transition will take place.
Thickness to chord ratio	n.a.	0.25	Wing structure weight is inversely proportional to the square root of the thickness to chord ratio [13]. The thickness to chord ratio is therefore important for the weight of the wing.
Stall angle	Transition	0.15	This is deemed important for transition, as this will determine the amount of lift the wing will still be able to provide during transition. However, as the mechanics of the transition are still largely unknown, the weight for this criterion is low.

Having defined the trade-off criteria, a list of airfoil candidates was selected by looking for airfoils used in other eVTOLs and those having similar reynolds regimes as Vold-e [14–17]. These candidates are shown in Table 4.4. The geometry data of these airfoils is extracted from Airfoil Tools ¹, except for the NACA44017, which was extracted from BigFoil ².

¹Airfoil Tools, <http://airfoiltools.com/>, Accessed 03-06-2026

²BigFoil.com, https://bigfoil.com/9b709ab0-97de-4c45-8223-d8ad075e32fc_info.php, Accessed 03-06-2026

4.2.2. Trade-Off Process and Outcome

In order to perform the tradeoff, the airfoils have been analysed in Flow5 (V7.56) ³. To perform the analysis the Reynolds number and Mach number are needed as inputs. As the trade-off is performed during multiple flight regimes, the regime-specific Reynolds number and Mach number were computed, which are presented together with the design lift coefficient and the parameters needed for their calculation in Table 4.3. For transition, the landing altitude of Courchevel, being 6000 feet, is assumed. For cruise, an altitude of 12500 feet is used, as specified in Chapter 1. It should be noted that preliminary values for aircraft mass and wing area were used. These values, together with the chord and therefore the Reynolds number, are still expected to change over the duration of the design process.

The formulas used to arrive at the Reynolds number, Mach number and design lift coefficient are found in Equation 4.1. In these formulas, the lift coefficient of the wing is assumed to be equal to the lift coefficient of the airfoil. According to Raymer [13] this can be used as a first approximation.

$$Re = \frac{\rho V c}{\mu} \quad M = \frac{V}{a} \quad C_{L,design} = \frac{W}{0.5 \rho_{cruise} V^2 S} \approx C_{l,design} \quad (4.1)$$

Table 4.3: Input parameters for Reynolds and Mach number calculations

Parameter	Cruise	Transition
Preliminary mass aircraft [kg]	2000	2000
Preliminary wing area [m^2]	25.82	25.82
Altitude, h [ft]	12500	6000
Air density, ρ [kg/m^3] ⁴	0.83568	1.02393
Temperature T [K] ⁴	263.385	276.263
Dynamic viscosity, μ [Pa s] ⁵	1.6731×10^{-5}	1.7478×10^{-5}
Speed of sound, a [m/s] ⁴	325.34	333.2
Flight velocity, V [m/s]	55.6	20
Mean aerodynamic chord, $\bar{c}$ [m]	1	1
Reynolds number, Re [-]	2.8×10^6	1.2×10^6
Mach number, M [-]	0.17	0.06
Design lift coefficient, $C_{l,design}$ [-]	0.59	-

With these parameters, the analysis in Flow5 can be performed. To complete the analysis, the geometry data of the airfoils were imported and the panel size was increased to 300, with a bunching parameter of roughly 0.25. Only the S1091 airfoil had a reduced panel count of 100 due to convergence issues. With these settings, two analysis were run, both of which had an Ncrit value of 9 to simulate an average wind tunnel ⁶. In the case of cruise, an analysis with C_l varying from 0.4 to 0.7 with a step size of 0.01 was run. For the transition, an analysis with α varying from 0 to 20 [°] was run, with a step size of 0.1. This lead to the values as shown in Table 4.4. Based on these values, a score was awarded to each airfoil ranging from 1-9: 1 for the worst, 9 for the best scoring airfoil. These scores were then multiplied with the weight and summed up for each criteria to arrive at a final score for each airfoil. The results of this trade-off can be seen in Table 4.5. From this table, it becomes clear that the NASA LANGLEY LS(1)-0417 won and will be chosen for Vold-e.

Table 4.4: Airfoil trade-off parameters

Airfoil	C_l/C_d at $C_{l,design}$	$C_{l,max}$	C_m at $C_{l,design}$	t/c	Stall angle [°]
EPPLER 793	91	1.76	-0.124	0.16	14.4
NACA 23018	50	1.01	0.007	0.18	12.5
NACA 4412	63	1.43	-0.099	0.12	11.8
NACA 64(3)-618	104	1.52	-0.119	0.18	16.9
NACA 65410	100	1.43	-0.083	0.10	13.0
NACA 44017	88	1.66	-0.003	0.17	16.2
NACA 6409	98	1.57	-0.151	0.09	11.7
NASA LANGLEY LS(1)-0417	134	1.82	-0.123	0.17	17.8
SELIG S1091	93	-	-0.143	0.05	-

³flow5, <https://flow5.tech/flow5.html>, Accessed 04-06-2026

⁴Aerotoolbox, <https://aerotoolbox.com/atmcalc/>, Accessed 3-6-2026

⁵The Engineering Mindset, <https://theengineeringmindset.com/properties-of-air-at-atmospheric-pressure/>, Accessed 3-6-2026

⁶Airfoil Tools, <http://airfoiltools.com/airfoil/details?r=polar/index/#xfoil>, Accessed 09-06-2026

Table 4.5: Airfoil trade-off scores

Airfoil	C_l/C_d at $C_{l,\text{design}}$	$C_{l,\text{max}}$	t/c	Stall angle [°]	Total
EPPLER 793	4	8	5	6	4.75
NACA 23018	1	2	9	4	2.6
NACA 4412	2	4	4	3	2.6
NACA 64(3)-618	8	5	8	8	6.05
NACA 65410	7	3	3	5	4.15
NACA 44017	3	7	6	7	4.4
NACA 6409	6	6	2	2	4.1
NASA LANGLEY LS(1)-0417	9	9	7	9	7.35
SELIG S1091	5	1	1	1	2.25

4.2.3. Airfoil Analysis Verification

To verify the Flow5 analysis setup, the aerodynamic curves of the NASA Langley LS(1)-0417 airfoil are compared to reference XFOIL data from Airfoil Tools ⁷. Since both Flow5 and Airfoil Tools use XFOIL ⁸ this comparison does not include full validation against experimental data. It however serves as a consistency check to confirm the Flow5 inputs such as panel settings were implemented correctly. Both datasets were generated with the same conditions: a Reynolds number of 1 million, a Mach number of 0, and a Ncrit of 9. The relative differences between the Airfoil Tool reference data and the Flow5 results in percentages for different angle of attacks are presented in Figure 4.1. Lift coefficient values near 0 were omitted from the lift coefficient difference plot as the relative difference in this region becomes artificially large, while it doesn't indicate aerodynamic disagreement.

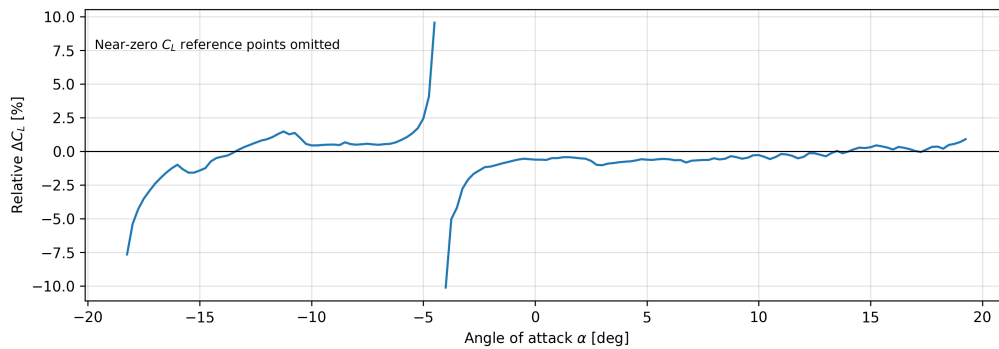


Figure 4.1: Relative difference between the Flow5 data and the AirfoilTools reference data for the NASA Langley LS(1)-0417 airfoil.

From the figure it can be deduced that within the operating range relevant for the design, the agreement between the 2 datasets is good for C_L . The same analysis was also performed for C_D and C_M , and all 3 curves show good agreement. It should be noted that larger differences occur at the high angle of attack and stall regions. This is deemed irrelevant for our validation, as XFOIL based methods, such as Flow5, do not model stall behaviour accurately [18]. Overall, the comparison shows that the Flow5 aerodynamic curves are consistent with the Airfoil Tools reference data in the range relevant for the design. This indicates that the Flow5 setup is sufficiently reliable for the airfoil trade-off.

4.3. Vertical Tail Sizing

For the vertical tail, a symmetric airfoil is selected to ensure yaw stability and control authority for both positive and negative sideslip angles without introducing a side-force bias at zero sideslip. Since the vertical tail is not required to generate a steady side force during cruise, camber is unnecessary and would result in undesirable trim forces. The thickness-to-chord ratio is therefore governed primarily by structural requirements, including sufficient stiffness, space for internal attachments, and integration with the fuselage structure. Based on these constraints, the minimum acceptable airfoil thickness-to-chord ratio is set to 0.1. Since the NACA 0010 airfoil satisfies both the symmetry and thickness requirements, it was selected for the vertical tail.

4.4. Box-Wing Sizing

The box-wing layout is sized by defining the positions of the front and aft wing relative to each other. This positioning is influenced by aerodynamic and integration constraints. Aerodynamically, the vertical spacing between the wings affects the induced drag, while from an integration perspective, the vehicle must fit within the ground footprint and allow passenger access when in the folded configuration.

⁷Airfoil Tools, <http://airfoiltools.com/>, Accessed on 4th of June 2026

⁸XFOIL, <https://web.mit.edu/drela/Public/web/xfoil/>, Accessed on 23th of June 2026

4.4.1. Box-Wing Horizontal Gap

The horizontal gap between the front and aft wing was initially set to 5 [m]. This preliminary value was mainly driven by the folded configuration and passenger access requirements. In the folded configuration, entry from the sides is blocked by the wings themselves, while rear entry is not possible due to the vertical tail. Therefore, passenger entry was assumed to take place through the nose.

During the design optimisation described in Chapter 9, the longitudinal wing positions were updated to satisfy the stability and integration requirements of the final configuration. From this it followed that the final front-wing LEMAC position is located at $x_{LEMAC, fw} = 0.17$ [m] from the nose, while the aft-wing LEMAC position is located at $x_{LEMAC, aw} = 5.60$ [m] from the nose. This gives a final horizontal gap of 5.43 [m].

Since the preliminary 5 [m] gap was only used to establish the initial box-wing layout. The final value of 5.43 [m] is used in the final configuration and should be considered as the definitive horizontal spacing between the front and aft wing.

4.4.2. Box-Wing Vertical Gap

The vertical gap h between the front and the aft wing is an important parameter for any box-wing configuration. It affects the aerodynamic interaction between both lifting surfaces, but also affects other departments. A larger vertical gap will influence controllability, stability, and structural mass. The decision was made to first size the vertical gap from an aerodynamics point of view, after which the selected value was checked against control, stability, and structural requirements.

Increasing the vertical gap improves the aerodynamic efficiency in 2 ways. First, it improves the induced drag behaviour of the aircraft. For a fixed wingspan b , increasing h improves the Oswald efficiency factor, and therefore reduces the induced drag. Second, it reduces the interference between the front and aft wing. As the aft wing is moved upward, it is less in the wake of the front wing, allowing it to maintain more of its single-wing lift performance. Given these 2 effects, the largest possible vertical gap is preferred.

The vertical gap can however not be increased without a limit as the aircraft must remain within the maximum allowed ground footprint height of 3 [m]. From the initial preliminary geometry, it followed that given the maximum height of 3 [m], the largest vertical gap possible was 2.1 [m]. Similar to the horizontal gap, after the design optimisation done in Chapter 9, it turned out that a vertical gap of 2.1 [m] would exceed the height requirement of 3 [m] due to geometric changes needed to comply with the stability requirements. The decision was made to keep the vertical gap of 2.1 [m], due to the earlier mentioned advantages, and instead to not comply with the height footprint constraint.

The induced-drag benefit was estimated using the Rizzo relation, as reported by M. Waeterschoot [19]. This relation links the induced-drag factor k , and therefore the Oswald efficiency factor e to the vertical gap-to-span ratio h/b , and is given by Equation 4.2:

$$e = \frac{1}{k} \quad k = \frac{0.44 + 0.9594 \frac{h}{b}}{0.44 + 2.219 \frac{h}{b}} \quad (4.2)$$

Since the wingspan b is fixed, the Oswald efficiency factor only depends on the vertical gap h . The resulting curve is shown in Figure 4.2. It increases rapidly for low values of h , after which the curve flattens out. In the limit where the vertical gap tends to infinity, the Oswald efficiency factor for a box-wing configuration reaches its asymptote of 2.31. This asymptote represents the theoretical upper bound, not a practically achievable design target.

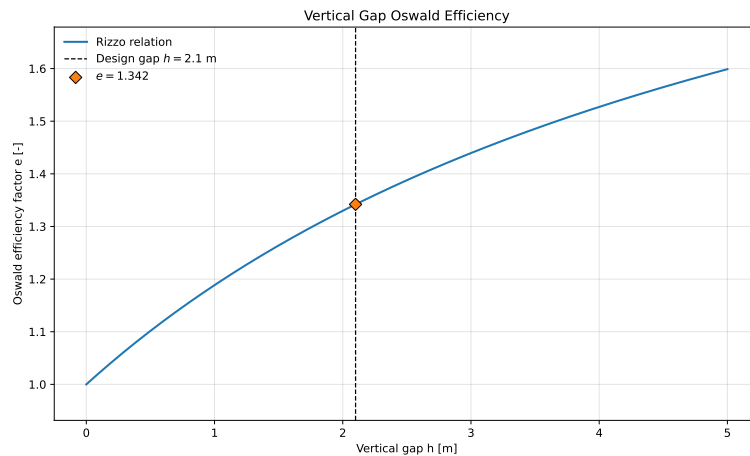


Figure 4.2: Rizzo relation: Oswald efficiency factor for different values of the vertical gap h .

With the selected wingspan of 13 [m] and a vertical gap of 2.1 [m], the design h/b has a value of 0.1615, which gives an Oswald efficiency factor of 1.34. This is an improvement of 34% compared to the zero-gap reference. Although

the theoretical limit cannot be reached due to the ground footprint constraints, the design point still captures the available induced-drag benefits within the available aircraft height.

Besides the induced-drag benefits, the vertical gap also affects the interference between the front and aft wing. For this, an analysis was performed in XFLR5 using the Vortex Lattice Method (VLM)⁹ to assess the lift interference between the front and aft wing. The front and rear wings were modelled as two separate lifting surfaces placed behind each other. The rear wing was then moved up step by step, and at each step the lift coefficient of the two-wing configuration was compared to the main-wing reference lift coefficient.

According to Cipolla et al. [20], VLM methods are suitable for preliminary aerodynamic analysis of Prandtl plane configurations. Their work applied the Athena Vortex Lattice (AVL) to a Prandtl configuration and validated the results against windtunnel data for the *IDINTOS* aircraft, resulting in errors lower than 10% for small angles of attack and sideslip. Therefore, the VLM is considered suitable for estimating the preliminary inviscid wing-to-wing interaction.

The XFLR5 analysis was done for wing gap values of 0, 0.5, 0.75, 1, 1.4, 1.8, 2.2, 2.6 and 3 meters. In addition to this, a root chord of 1 meter was used, with a taper ratio of 0.45 as defined in Table 4.1. Furthermore, it is assumed that both wings are identical and the horizontal gap was fixed at 5 [m]. All configurations have been run with the VLM1 setting at cruise condition ($h = 3810$ [m] and $T = -9.8$ [°C]) and with a panel count of 20 in the x direction with cosine spacing and 40 in the y direction with -sine spacing. As shown by [21], refining the panels any more has negligible effects on the results. The resulting ratio between the lift coefficient of 2 wings over the lift coefficient of only the main wing, $\frac{C_{L,2W}}{C_{L,1W}}$, is plotted in Figure 4.3. This ratio reaches its asymptote of 1 when h tends to infinity; i.e. when h is large enough, the aft wing will not be in the wake of the front wing, and therefore the lift coefficient of the two-wing configuration is the same as the main-wing-only configuration.

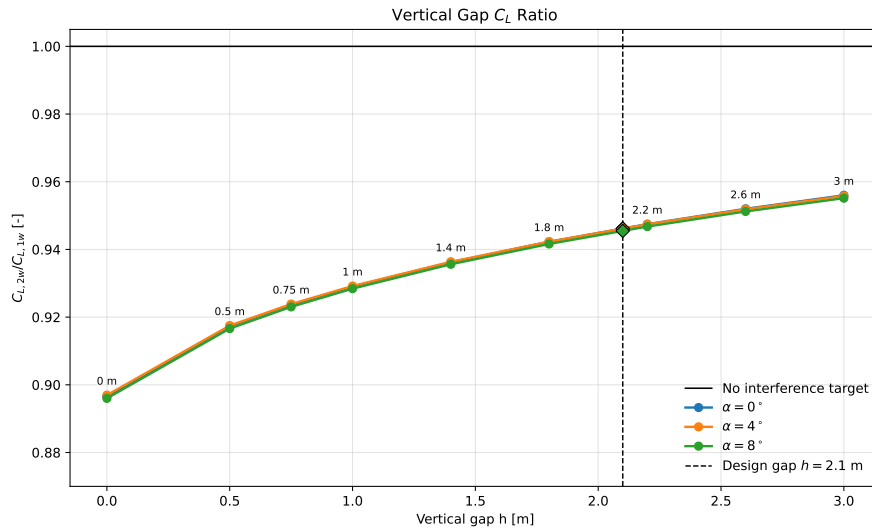


Figure 4.3: C_L ratio for different values of the vertical gap h , analysed at different angle of attacks.

At the selected design gap of $h = 2.1$ [m], interpolation gives a C_L ratio of approximately 0.95 for the analysed angles of attack. This corresponds to a lift-interference error of 5% relative to the main-wing-only reference, which was considered acceptable.

From this design gap analysis, it was also possible to determine the cruise angle of attack. For this, the same $C_{L,design}$ that was calculated in Subsection 4.2.2 was used, which is 0.64. By inspecting the XFLR5 data, a corresponding angle of attack could be found: $\alpha_{cruise} = 2.6$ [°].

4.5. Fuselage Selection

In this section, the geometry of the fuselage will be determined. The objective will be to minimise the drag coefficient of the fuselage shape. In order to do this, a baseline design will be developed, from which the different design options will flow. These design options are then modelled in CAD software (Autodesk Fusion¹⁰), after which their drag coefficient was determined using Ansys Fluent¹¹ at the cruise angle of attack of $\alpha_{cruise} = 2.6$ [°].

The fuselage candidates were generated by varying a baseline geometry. The length of the fuselage is limited by the ground footprint, specified by REQ-SYS-02. This limits the total length of the fuselage to be 10 [m]. In terms of the passenger compartment, the Cessna Citation Ascend was used as a reference for the diameter of the passenger compartment. As this aircraft has an inner cabin diameter of 1.73¹², a fuselage diameter of 1.8[m] is assumed for

⁹XFLR5, <https://www.flow5.tech/xflr5/xflr5.html>, Accessed on 5th of June 2026

¹⁰Autodesk Fusion, <https://www.autodesk.com/nl/products/fusion-360/overview>, Accessed on 23th of June 2026

¹¹Ansys Fluent, <https://www.ansys.com/>, Accessed on 23th of June 2026

¹²Textron Aviation, <https://cessna.txtav.com/en/citation/ascend>, Accessed 10-6-2026

Vold-e. The fuselage length needed for the four seats was based on KLM's Premium Comfort seat pitch of 97 [cm]¹³. Adding a margin of 0.5 [m] for luggage that cannot be fit elsewhere gives a cabin length of 2.5 [m]. The parameters that were varied are listed below, a visualisation of these parameters can be seen in Figure 4.4.

1. **Length of the fuselage.** According to Raymer [13], a fineness ratio of 3.0 is optimal for minimising subsonic drag. However, Roskam [22] mentions that most general aviation aircraft have a fineness ratio of 5-6. As such, it was chosen to test both a fuselage length of 6 [m], and a fuselage length of 10 [m], corresponding to an fineness ratio of roughly 3.3 and 6.3 respectively.
2. **Shape of the nose.** According to Abbott [23] for a fuselage with a fineness ratio of 5, bodies with a sharp nose and tail have the lowest drag coefficient. However, the effect of the nose size for fineness ratio of 3 remains unknown. Therefore, different nose shapes were tested: blunt and sharp.
3. **Longitudinal position of passenger cabin.** Depending on the allowable c.g. range, the cabin can be shifted more forward and more aft. This will have an effect on the sharpness of the nose and the tail. Again, two configurations will be tested: one with the cabin more forward and one with the cabin more aft.
4. **Vertical position of aft part.** Depending on the required size of the vertical tail. The aft part of the fuselage might need to be placed lower than the middle of the fuselage. The determine the effect of this, two configurations were tested: the tail in the middle and the tail level with the lowest part of the fuselage.

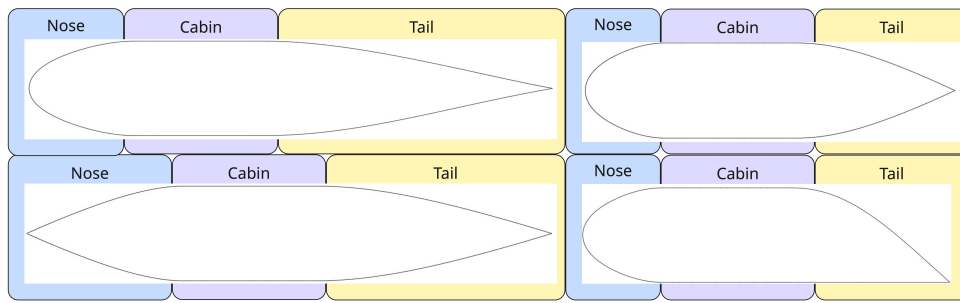


Figure 4.4: Fuselage design variations

In order to reduce the number of CFD simulations needed, the design variations were done in series. In this approach, only the best performing candidates from a design step were taken to the next design step. This method greatly reduces computational cost, but it also comes with a limitation. Not all possible combinations of design parameters are tested, which means the selected fuselage design may not be the global optimum. This method is summarised in Figure 4.5 outlining which designed were deemed favourable and which weren't. In this diagram, the naming convention works as: Fuselage_length_nose length_nose type (Blunt or Sharp) _ Tail location (Middle or Lower). In addition, the computed C_d value for each configuration is given in the orange boxes.

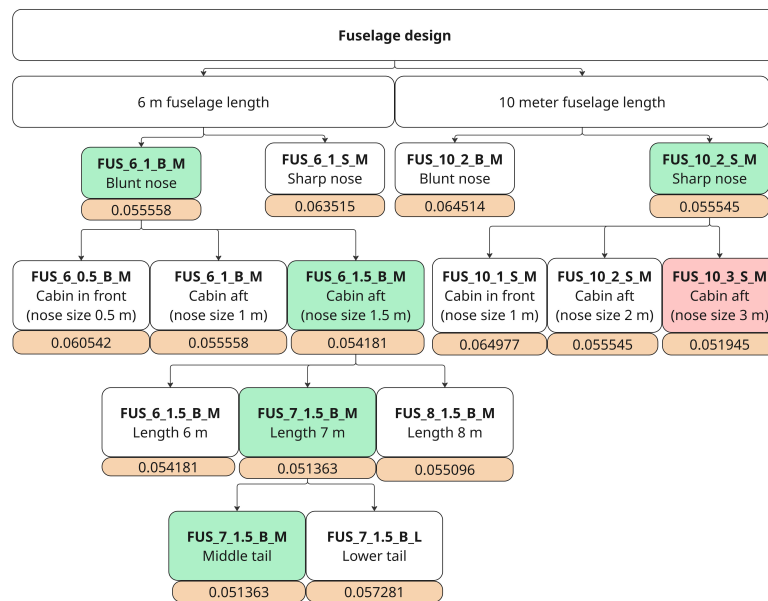


Figure 4.5: Schematic of the fuselage design process

¹³KLM, <https://www.klm.nl/en/information/travel-class-extra-options/aircraft-types/boeing-777-300er>, Accessed 10-06-2026

4.5.1. Final Fuselage Characteristics

To assess the characteristics of the final fuselage, the CFD simulation was repeated using a more detailed turbulence and near-wall modelling approach. Further information on the meshing strategy and solver setup is provided in Appendix A. The resulting fuselage drag components at the estimated cruise angle of attack, $\alpha_{\text{cruise}} = 2.6^\circ$, are presented in Table 4.6. In ANSYS Fluent, the total drag is decomposed into pressure drag and viscous drag, separating the contribution of form effects and wall-shear effects. The drag values are slightly higher than those obtained for the same fuselage geometry during the trade-off process. This is expected, as the trade-off simulations used a less detailed but more computationally efficient setup, which produced a more optimistic drag estimate. The lower-fidelity setup used during the trade-off process is also described in Appendix A.

Table 4.6: Drag force and drag coefficient components for the fuselage

Zone	Forces [N]			Coefficients		
	Pressure	Viscous	Total	Pressure	Viscous	Total
Fuselage	98.100	95.389	193.490	0.02983	0.02901	0.05884

Figure 4.6 shows the velocity streamlines around the selected fuselage at the cruise condition. The streamlines remain mostly aligned with the freestream direction and follow the fuselage contour without showing large-scale flow reversal. Acceleration of the flow can be observed around the forward and mid-body regions, while a lower-velocity wake forms downstream of the tapered tail. The wake remains concentrated behind the fuselage, indicating that the selected geometry guides the flow smoothly towards the aft section. This supports the suitability of the final fuselage shape for cruise, since no large separated region is observed in the streamline field.

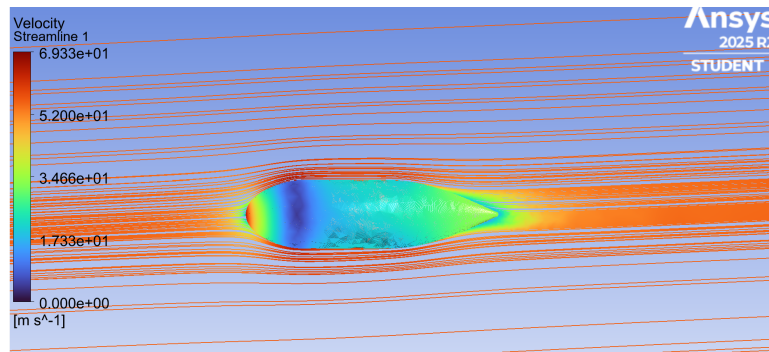


Figure 4.6: Velocity streamlines around the fuselage at cruise conditions, $V_{\text{cruise}} = 55.6 \text{ m/s}$ and $\alpha_{\text{cruise}} = 2.6^\circ$.

Figure 4.7 presents the pressure-coefficient distribution along the upper and lower fuselage surfaces. The pressure coefficient is highest near the nose, where the incoming flow is decelerated close to the stagnation region. Moving downstream, the pressure coefficient decreases as the flow accelerates around the fuselage, after which gradual pressure recovery occurs towards the tail. The upper and lower surface distributions are similar, which is expected because the fuselage is symmetric and is analysed at a relatively small angle of attack. The remaining difference between the two curves indicates a slight asymmetry in the pressure field caused by the positive cruise angle.

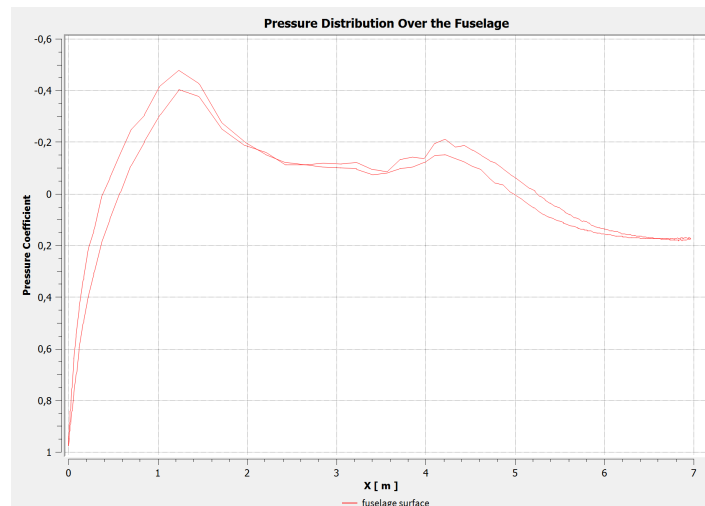


Figure 4.7: Pressure coefficient distribution along the upper and lower surfaces of the fuselage at cruise conditions, $V_{\text{cruise}} = 55.6 \text{ m/s}$ and $\alpha_{\text{cruise}} = 2.6^\circ$

Figure 4.8 shows the skin-friction coefficient distribution over the fuselage. The highest skin-friction values occur near the forward part of the fuselage, where the boundary layer is still relatively thin and the near-wall velocity gradients are high. The lowest values are found near the tapered tail, where the flow decelerates and the wall shear stress becomes smaller. Since the skin-friction coefficient remains positive over the visible fuselage surface, no major separated region is present on the fuselage at the analysed cruise condition.

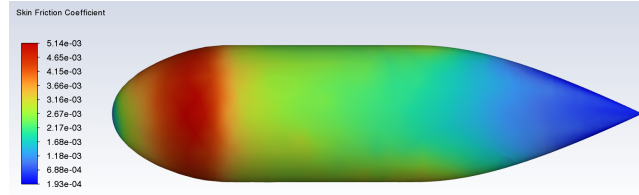


Figure 4.8: Skin-friction coefficient distribution over the final fuselage at cruise conditions, $V_{\text{cruise}} = 55.6 \text{ [m/s]}$ and $\alpha_{\text{cruise}} = 2.6 \text{ [}^\circ\text{]}$

4.6. Discussion

Although the aerodynamic design presented in this chapter provides a preliminary basis for the Vold-e, several limitations remain. The main limitation is that the analyses were performed at different fidelity levels. The airfoil selection was based on 2-dimensional XFOIL data, the wing-to-wing interaction was analysed using a VLM setup in XFLR5, and the fuselage was analysed separately using CFD with Ansys. As a result, important interaction effects between components were not captured at this stage. To reduce the impact of this limitation, the integrated aircraft geometry is analysed in more detail in Chapter 10, where a full-aircraft CFD is used to assess the final aerodynamic performance characteristics.

A second limitation is that the airfoil selection and the box-wing sizing rely on preliminary methods. XFOIL based methods, which were used for the airfoil selection, are less reliable near stall, which makes the predicted maximum lift coefficient and the stall angle uncertain. Similarly, the VLM analysis used for the box-wing interaction neglects viscous effects and detailed wake behaviour. Therefore, the selected airfoil and the box-wing spacings should be validated in the future using CFD analysis or wind-tunnel testing.

Besides this, after the optimisation some geometric parameters of the aircraft changed to satisfy the stability requirements. The decision was made to keep the vertical gap of 2.1m as it has a strong aerodynamic influence. Due to time constraints, no further re-iteration could be performed to find a configuration that both satisfies the new geometry and remains within the ground footprint while preserving the aerodynamic benefits of the selected vertical gap. This non-compliance was accepted as the height constraint was considered less critical to the mission and operational use-case. Further work should explore alternative geometric designs or reconsider the vertical gap values to comply with the height requirement while inducing a minimal aerodynamic penalty.

Finally, transition aerodynamics were not included in the current aerodynamic sizing, as this was considered outside the scope of the project. Future work should therefore also focus on transition analysis before the aerodynamic performance can be considered fully validated.

5 Power and Propulsion

The first part of this chapter outlines how the battery is sized through estimating the power required for certain phases, while the second is focused on analysing the propulsion system required for Vold-e. Initially, a thorough analysis is completed over the power used through the mission in Section 5.1 as this is the section that requires the highest power. Next, a simulation is made to estimate the amount of power required by other systems which is completed in Section 5.2. With these power estimates, a battery type can be chosen and how the pack is structured can all be seen in Section 5.3. Next, in order to ensure the batteries are properly managed, a battery management system is defined in Section 5.4 and the thermal management system is explained in Section 5.5. This concludes the power-related part of the chapter, after which the propulsion part is presented. It begins with the determination of the propeller geometry required to minimise noise, as described in Section 5.6. Finally, based on this information, a suitable motor is selected to match the required characteristics, as discussed in Section 5.7. Finally, Section 5.8 discusses the limitations of the current power and propulsion system design and highlights areas for further development.

5.1. Mission Profile

In order to calculate the amount of power needed per mission phase, an analysis was conducted looking at the entire mission profile of flying from Nice to Courchevel and splitting it up into 6 phases. These phases are: take-off, vertical climb, climb, cruise, descent and landing. To analyse the amount of power required for each phase, analytical formulas were used to estimate these values.

Take-off, Vertical Climb & Landing

Power consumed during take-off and landing is the highest power consumption of all mission phases. The main reason is most of the acceleration is during the hover mode. During this phase it is assumed that it will last 5 seconds and have an average take-off velocity of 2 [m/s]. To calculate the power consumed at this stage, the average thrust, hover induced velocity and the average induced velocity was found. These equations come from dynamic equations looking at free body diagrams and research completed by O. Ugwueze, et al. [24]. From these two sources the following equations were calculated or found and then used:

$$T_{avg} = MTOW \cdot \left(g + \frac{v_{hover}}{t}\right) \quad v_{i_{hover}} = \sqrt{\frac{T_{avg}}{2 \cdot \rho_0 \cdot A_{disk} \cdot 6}} \quad (5.1)$$

$$v_{i_{avg}} = -\frac{v_{take-off}}{2} + \sqrt{\left(\frac{v_{take-off}}{2}\right)^2 + v_{i_{hover}}^2} \quad P_{take-off} = T_{avg} \cdot \frac{(v_{i_{avg}} + v_{take-off})}{FM} \quad (5.2)$$

Similarly, these equations were used for the vertical climb and landing. However the velocities taken were not the take-off velocity, but the average climb and landing velocity was taken.

Climb & Descent

In between the vertical climb and the climb phase the eVTOL will transition. After the transition, the remaining climb is completed with the eVTOL in cruise configuration. During the climb phase, it is assumed that the eVTOL will travel with an initial horizontal velocity of 20 [m/s] induced by the transition phase. It is then required to accelerate to the cruise velocity of 56 [m/s] in 20 seconds. To calculate the power during this climb phase, the principle of energy conservation was used, as so:

$$E = \frac{1}{2} \cdot MTOW \cdot (v_{cruise}^2 - v_{x_i}^2) + MTOW \cdot g \cdot v_{hover} \cdot t_{climb} + \frac{1 \cdot MTOW \cdot g \cdot a_{x_{climb}} \cdot t_{climb}^2}{2 \cdot L/D} \quad (5.3)$$

This give the energy during climb in Joules, however we want this energy to be converted into [kW] which is done by dividing the energy with a conversion factor of $3.6 \cdot 10^6$ and dividing it by the acceleration time. This method is also applied to the descent phase but then taken that the final velocity achieved is 5 [m/s]. For the climb stage, this calculated energy is only the energy needed to accelerate to the cruise velocity. To look at the energy required to actually perform the climb, the vertical distance travelled during the acceleration stage is added to the previous vertical distance, and then subtracted from the cruise altitude. This is the remaining vertical distance still required to climb. From the previously travelled horizontal distances and the needed remaining vertical distance, the flight path angle (γ) can easily be calculated. Then to find the power during the climb is calculated as follows:

$$P_{climb} = MTOW \cdot g \cdot \sqrt{(v_{cruise}^2 + v_y^2)} \cdot \left(\frac{D}{L} \cdot \cos(\gamma) + \sin(\gamma)\right) \quad (5.4)$$

Cruise

During cruise to calculate the power, the following equation can be used:

$$P_{cruise} = \frac{MTOW \cdot g}{L/D} \cdot v_{cruise} \quad (5.5)$$

Reserve Mission Power

In addition to only having power available during each phase of the mission, a reserve range was also determined in case of any emergencies and there was no possibility of landing in Courchevel. According to EASA, a reserve range is determined depending on the mission [25]. Hence, for the mission from Nice to Courchevel, a worst case scenario would be when the eVTOL is unable to land in Courchevel. In order to make a safe landing from this point onwards, the eVTOL would have to fly to a city at a lower altitude, this closest by city is Chambéry. To analyse the additional amount of power needed to make this landing, a similar method was carried out as for the initial battery sizing. In the end, this added up to requiring 20% of the mission energy for the reserve range. This added up to the total energy needed for the battery to be 168 [kWh].

5.2. Power Usage Across Systems

Besides propulsion, Vold-e requires electrical power for cabin conditioning, avionics, flight-control actuation, lighting and other auxiliary systems. Although these loads are small compared with the propulsion demand, they contribute to the total mission energy and are therefore included in the battery sizing. This section estimates the power use of the environmental control system and the remaining auxiliary loads throughout the mission.

5.2.1. Environmental Control System

The ECS maintains cabin pressure and temperature and provides ventilation to make sure the passengers have a comfortable flight. It is sized using the method provided in the electric ECS model of Fioriti and Di Fede [26], which couples a cabin heat-load balance to a dedicated electric compressor. The air cycle machine (ACM) cooling train of the reference model is omitted.

The cabin heat balance gets the sum of the metabolic load of the occupants, the waste heat, solar gain and the conduction through the fuselage skin:

$$\dot{Q}_{cabin} = \dot{Q}_{met} + \dot{Q}_{avionics} + \dot{Q}_{disp} + \dot{Q}_{lights} + \dot{Q}_{solar} + \dot{Q}_{cond} \quad (5.6)$$

The avionics heat term (539 [W]) is taken from the avionics power build up Subsection 5.2.2, of which it is the in cabin heat dissipation component. At low altitude on a hot day, the cabin gains heat (Equation 5.6 positive), requiring cooling; at cruise altitude, the conduction term becomes more negative and the cabin is net-heating, so no active cooling is required in cruise.

Table 5.1: Cabin heat balance, hot ground cooling case (sea level, ISA+25) [26].

Term	Value [W]
Metabolic (4 occupants)	280
Avionics	539
Displays (4 screens)	34
Lighting	34
Solar through windows	600
Conduction through fuselage	815
Total (heat to remove)	2302

This 2.3 [kW] of heat to remove is the basis for the cabin AC compressor sized in Subsection 5.2.2. At low altitude on a hot day, the cabin gains heat and requires cooling; at cruise altitude, however, the main conduction term becomes negative (the ambient is cold at 12500 [ft]) and the balance reverses to approximately 1240 [W], so the cabin is in net heating and no active cooling is required in cruise.

For the **compressor power and sizing**, the compressor performs pressurisation work, set by the pressure ratio β_c between cabin and ambient. At low altitude, where β_c is 0, no pressurisation is required; as the aircraft climbs into the pressurised regime the ratio increases, peaking at 1.26 in cruise (cabin held at a 6000[ft] equivalent). The electrical motor power follows from the compressor power and the motor/drive efficiency. The peak motor power and mission ECS energy are summarised in Table 5.2.

Table 5.2: ECS sizing results.

Parameter	Value
Cabin pressure (6000 ft equiv.)	0.813 bar
Peak pressure ratio (cruise)	1.26
Avionics heat into cabin	539 W
Peak motor power (cruise)	1.27 kW
Mission ECS energy	0.855 kWh
ECS share of propulsion energy	0.91%

The peak motor power of 1.27 [kW] occurs in cruise, driven by the pressurisation demand rather than the thermal load, as no active cooling is required in cruise. The total ECS pressurisation energy over the mission is 0.855 [kWh], drawn from the 270 [V] bus. At under 1% of the mission propulsion energy, it is not sized into the pack explicitly but

is comfortably covered by the 20% energy reserve in Subsection 5.3.1. It is distinct from the AC compressor, which performs cabin heat removal rather than pressurisation and is included in the auxiliary loads (Subsection 5.2.2). The ECS sizing is sensitive to the chosen cabin pressure as it is governed by the cruise pressurisation. The table below shows that a sea level cabin would roughly double both the peak power and the mission energy relative to the 6000 [ft] cabin adopted, while a higher cabin would reduce.

Table 5.3: *ECS sensitivity to cabin pressure altitude.*

Cabin altitude [ft]	Cabin p [bar]	Peak motor [kW]	ECS energy [kWh]
0	1.013	2.56	1.86
4000	0.875	1.69	1.18
6000	0.813	1.26	0.85
8000	0.753	0.84	0.54

The ACM cooling train of the reference model is omitted because the compressor exit temperature remains low at the altitudes flown, so no active air cycle cooling of the compressed air is required. This is consistent with the cruise altitude.

5.2.2. Auxiliary Loads

The vehicle should also be able to provide enough energy to supply the non-propulsive loads. This is supplied with two buses: the 270 V bus for the actuators, folding mechanism and AC compressor; and the 28 V bus for avionics, sensors, lighting, cabin electronics, and battery management. This split onto two buses is a detailed design refinement: the high power actuator and compressor loads draw impractically high currents on a low voltage bus (for example, a 6 [kW] actuator would draw over 200 [A] at 28 [V] versus 22 [A] at 270 [V]), so a higher voltage bus is introduced for these loads, while the low power avionics and cabin loads remain on the 28 [V] bus; and these loads are sized by a component build up.

Each component is assigned a power and a per phase duty cycle and the energy is integrated over the mission and divided by the conversion efficiency.

$$E_{aux} = \frac{1}{\eta_{DC}} \sum_{\text{phases}} \left(\sum_{\text{loads}} P_i d_i \right) t_{\text{phase}} \quad (5.7)$$

where P_i is the load power of the component and d_i is the duty cycle in that phase. The avionics powers are taken from their data sheets (Section 7.1) and are consistent with the ECS model (Table 5.2), the AC compressor is derived from the ECS cabin cooling load (around 2.2 [kW]) divided by an assumed coefficient of performance of 2, and is active when the vehicle's in the warm low altitude phases (the cabin is net heating in cruise). The resulting energy by phase and bus is given below. Table 5.4.

Table 5.4: *Auxiliary electrical energy by mission phase.*

Phase	Auxiliary energy [kWh]
take-off	0.013
Vertical climb	0.063
Climb	1.182
Cruise	1.000
Descent	0.282
Landing	0.181
28 V bus subtotal	0.85
270 V bus subtotal	1.87
Total	2.72

The auxiliary energy is governed by the steady avionics, cabin and AC loads. The AC compressor is a separate machine from the ECS pressurisation compressor, and the two are counted independently with no overlap. The pressurisation compressor performs pressure work Subsection 5.2.1 while the AC performs heat removal.

5.3. Battery Chemistry & Cell Selection

The choice of the cell is driven by the specific energy of the cell which will be able to provide enough power and energy during the mission and the high continuous specific power. Fay et al. benchmark [27] 300 commercial cylindrical Li-ion cells against eVTOL energy and power requirements and identify nickel-rich chemistries (NMC, NCA, NMCA) as the only current graphite-anode chemistries simultaneously meeting the cell level energy density threshold of 240 [Wh/kg] and power density threshold of 2000 [W/kg] with the Molicel INR21700-P45B identified as the most capable screened cell.

This screening is insufficient for the present design as shown in Section 5.1, the combination of both 142.4 [kWh] and the battery mass budget of around 20-25% of the MTOW (Chapter 9) requires a cell level specific energy of the order of 380 [Wh/kg] once packaging overhead is accounted for, roughly 50% above the best current nickel-rich cylindrical cells [28]. A higher-specific-energy cell technology is therefore required.

A silicon anode lithium ion cell is selected: the Amprius SiCore SA504. Silicon anode cells retain the liquid electrolyte maturity of the conventional lithium ion cells while raising specific energy through the higher lithium capacity of the silicon anode and the SiCore series is commercially available and tested for aviation. The SA504 is a high energy and power cell containing 386 [Wh/kg]. It has a 6C continuous discharge rate [29]. This combination is selected as the energy used in the mission profile is being used to size the battery, with the auxiliary and reserve energy.

Table 5.5: Amprius SiCore SA504 cell datasheet values used for sizing.

Parameter	Symbol	Value
Nominal voltage	V_{cell}	3.40 V
Capacity (typical)	C_{cell}	11.05 Ah
Energy per cell	E_{cell}	37.57 Wh
Specific energy (cell)	$E_{sb,cell}$	386 Wh/kg
Max continuous discharge	I_{cont}	66.3 A (6C)
Continuous power per cell	P_{cell}	225 W
Cell mass	m_{cell}	97.3 g
Cycle life (100% DOD to 80% SOH)	–	500 cycles
Packaging	–	pouch

It is a pouch cell which imposes a compression fixture and constrains the achievable cell-to-pack ratio. The rated life cycle is higher than the other variants [29], the implications for pack replacement frequency are discussed in Subsection 5.3.1.

5.3.1. Battery Pack Selection

This subsection calculates the number of cells required, resulting in the architecture and checks the constraints. The battery must be able to provide enough energy for the mission given from Section 5.1, the auxiliary loads and the reserve. The main energy is at 137.2 [kWh], the auxiliary energy is built up by the component in Subsection 5.2.2 and totals 2.79 [kWh]. A reserve of 20% of the mission energy is adopted as the energy margin:

$$E_{deliv} = (E_{mission} + E_{aux}) (1 + k_{res}) \quad (5.8)$$

with the 20% factor giving an energy of 168 [kWh].

To reach the number of cells needed, the deliverable energy divided by the energy per cell.

$$N_{energy} = \left\lceil \frac{E_{deliv}}{E_{cell}} \right\rceil = \left\lceil \frac{168 \times 10^3}{37.57} \right\rceil = 4472 \quad (5.9)$$

The cells are arranged to meet the 800 [V] high voltage bus [30] with $N_{series} = 235$ cells in series the pack nominal voltage is $235 \times 3.40 = 799$ [V] and having a string energy of 8.84 [kWh] [29]; and having $N_{parallel} = 20(168/8.84 \approx 19)$ cells in parallel, giving a total of 4700 cells. The table below shows the pack characteristics of the battery.

Table 5.6: Battery pack sizing (initial estimate used to develop the method).

Parameter	Symbol	Value
Deliverable energy	E_{deliv}	168.0 kWh
Cell configuration	–	235S20P
Total cells	N	4700
Pack mass	m_{pack}	635 kg
Battery mass fraction	–	0.274

The pack mass of 635 [kg] is obtained from the cell mass. The cell-to-pack ratio: $4700 \times 0.0973 = 457$ [kg] of cells, divided by $f_{c2p} = 0.72$, giving 635 [kg]. Because the battery is sized by this same method inside the mass estimation loop (discussed below), the converged take-off mass already carries this battery mass, so no separate mass allocation is required. Looking at power, the battery can deliver 1059 [kW] of continuous power (66.3 [A] $\times$ 3.4 [V] $\times$ $4700 = 1059$ [kW]). The take-off phase demands 870 [kW], giving a margin of 1.22. The hover discharge rate is 4.93C, which falls within the 6C continuous rating of the cells. The power constraint is met with less margin than the energy constraint, which is acceptable given the moderate discharge rate and is the basis for selecting the higher-energy, longer life SA504 over a higher-power, shorter life variant.

Integration Into The Mass Estimation Loop

The sizing above is performed on the initial take-off mass estimate to set out the method and verify the cell selection and pack architecture. Because the battery mass depends on the take-off mass through the mission energy, the method above was implemented directly into the mass optimiser method (Chapter 7), replacing the initially calculated value, so that the battery is sized consistently with the converged final mass.

As the mass estimation loop converges, the optimised design parameters reduce the take-off mass below the initial estimate, and with it the mission energy and the required battery mass. The cell count, configuration and pack mass quoted above therefore correspond to the initial estimate used to develop the method; the final, converged values obtained from the optimised take-off mass are reported with the final mass breakdown:

The converged take-off mass of 1980 kg lowers the mission energy to 115.9 kWh, and the same sizing method then gives a deliverable energy of

$$E_{deliv} = (115.9 + 2.79)(1 + 0.20) = 142.4 \text{ kWh}, \quad (5.10)$$

So the energy-driven cell count is:

$$N_{energy} = \left\lceil \frac{142.4 \times 10^3}{37.57} \right\rceil = 3791. \quad (5.11)$$

Retaining the $N_{series} = 235$ string for the 800 [V] bus, the parallel count is $N_{parallel} = \lceil 3791/235 \rceil = 17$, giving a total of $235S17P = 3995$ cells. The converged pack characteristics are listed below.

Table 5.7: Converged battery pack sizing at the final take-off mass.

Parameter	Symbol	Value
Mission energy	$E_{mission}$	115.9 kWh
Auxiliary energy	E_{aux}	2.79 kWh
Reserve fraction	k_{res}	0.20
Deliverable energy	E_{deliv}	142.4 kWh
Installed pack energy	–	150.1 kWh
Hover/take-off power	P_{hover}	691 kW
Pack continuous power	–	901 kW
Hover discharge rate (pack)	–	4.61C
Cell-to-pack ratio	f_{c2p}	0.72
Cell configuration	–	235S17P
Total cells	N	3995
Pack nominal voltage	–	799 V
Pack mass	m_{pack}	540 kg
Battery mass fraction	–	0.273
Cell size (H × W × T)	–	120 × 42.2 × 8.25 mm

Energy remains the driving constraint (the 691 kW hover demand needs only about 3070 cells, against the 3791 set by energy). The pack delivers 901 kW continuous ($66.3 \text{ [A]} \times 3.4 \text{ [V]} \times 3995$), a power margin of 1.30 over the take-off demand, at a hover discharge rate of 4.61C, comfortably within the 6C cell rating. The resulting pack mass of 540 [kg] ($3995 \times 0.0973 / 0.72$) is the battery line carried by the converged take-off mass.

5.4. Battery Management System

The battery management system is used for monitoring all the cells in the battery by measuring cell voltage, current and temperature. This is done to ensure balanced charge and discharge of the cells and keep them in the optimal operating temperature via controlling the thermal control system.

The battery consists of 3995 cells, which means they should be structured in separate packs that are monitored by smaller, local BMS, because of their high count. In Figure 5.1, the arrangement of the cells is shown.

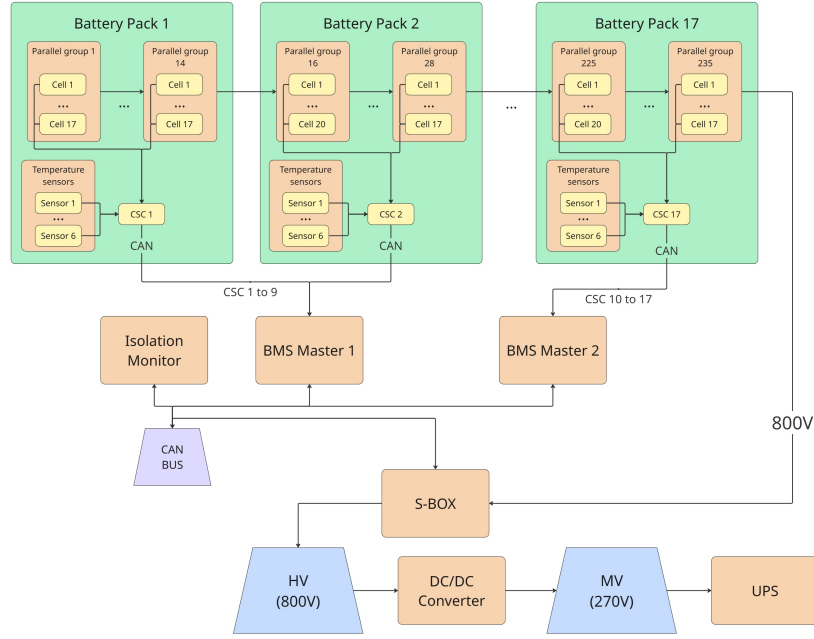


Figure 5.1: Battery management system diagram

In each pack, there are a maximum of 14 parallel groups that are connected in series. Each one of these groups consists of 17 cells that are connected in parallel. Because of that, the cells within each of these parallel groups will have the same voltage and can be connected to the same port of the CSC (Cell Supervision Circuit)¹. Based on these voltage measurements, the CSC balances the parallel groups by additionally discharging the ones with higher voltage. The information from all the CSC modules is being transferred to one of the two BMS Masters through a two-way High Speed Differential Isolated Communication¹. This includes the voltage of each parallel group and the information from the temperature sensors (6 per CSC). This information is used by each of the BMS Masters to determine the state of health and charge of the battery.

The S-BOX¹ is the main contactor of the battery. It is connected on the same bus as the BMS Masters and it receives information about the battery state of health and charge. It is also equipped with two separate current sensors. They ensure that the contactor will disconnect the battery in the case of a short circuit and communicate to the flight computer to enter the last stage of failsafe, which is the deployment of the parachute system. That way, the aircraft can perform an emergency landing without risking catastrophic battery failure.

The UPS system is used in order to provide a power supply to the medium voltage bus in the case of an emergency when the battery is disconnected by the main contactor. The battery capacity is enough to ensure the flight computer and the control surfaces can perform the failsafe procedure and deploy the parachute in the desired conditions (Section 7.2).

5.5. Thermal Control of Batteries

In order to ensure that the batteries operate at their optimum state, thermal control of the batteries must be implemented. The range of -20 to 60 [°C] for discharge. For charge however, this range is decreased to 0 to 45 [°C]. In order to create a thermal control system for the batteries, it must be estimated initially how much energy is dissipated from the battery pack. This can be done using [31]:

$$Q_{tot} = R \cdot I^2 \cdot n_{cells} \quad (5.12)$$

where R is the internal resistance of the cells, I is the current at a discharge rate (discharge rate x capacity) and n is the total number of cells. This gives the total dissipated power from the heat of the entire energy pack. It is estimated that the heat dissipated is equivalent to 2 [kW] to account for critical cases. This was calculated using a resistance of 6 $m\Omega$ and current 8.6 [A] (0.78 x 10.8) during nominal operations. Hence, using the heat expelled during the cruise configuration, the required mass flow from the coolant can be calculated using:

$$\dot{m} = \frac{Q}{c_p \cdot \Delta T} \quad (5.13)$$

Using a 50/50 ratio of water to glycol coolant, c_p is approximately equivalent to 3500 [J/kg K]¹. As the cabin temperature is fixed to be 21 [°C], and the optimum operating temperature of the cells is 30 [°C], the temperature difference is only 9 [°C]. However, to size for redundancy, it is estimated that the temperature difference is only 4 [°C] ensuring

¹Futavis, <https://www.futavis.de/en/>, Accessed on 12th of June 2026

that the battery never goes above 30 degrees in case the ambient temperature gets warmer. This gives rise to a coolant flow rate of 0.14 [kg/s]. As a 50/50 coolant mixture of water and glycol has a density of approximately 1060 [kg/m³]². To now calculate the cool plate area, we must calculate the heat absorbed by the coolant as the plate transfers the heat from the cells to the coolant.

$$A = \frac{Q}{U \cdot \Delta T_{lm}} \quad (5.14)$$

Where U is the overall cell to coolant heat transfer coefficient and ΔT_{lm} is the log mean temperature difference equation to 6.8. This assumed the cell being kept constant at 30 [°C], and the coolant rising from 21 to 25 [°C]. To find the exact heat transfer coefficient a lot of unknown parameters must be known, therefore looking at the nominal range of this value for lithium ion batteries and a liquid coolant a value of 600 [W/m² K] [32] for a redundant estimate. Therefore, the required area is 0.5 [m²] and assuming a thickness of 4 [mm], the volume of the cold plate is 0.002 [m³]. Using the mass flow rate and the final cooling, the approximated mass estimation can be completed. The heat rejection radiator is needed to expel the heat into the surroundings and the cold case heater is needed to make sure that the battery stays above the 0 [°C] operating range when landing in Courchevel. In Table 5.8 an estimate is given of the total mass of the battery thermal management system.

Table 5.8: Mass of Final Cooling Components

Component	Formula required	Final Mass [kg]
Coolant	$0.5 \times 0.004 \times 1060$	2.1
Aluminium Liquid Cold Plate	0.002×2700 ²	5.4
Circulation Pump	Estimate ³	2
Pumps, radiators, cold case heater, plumbing and valves	80 % of Mass	30
Total Mass		39.5

5.6. Propeller Selection

To simplify the propeller sizing, it is assumed that a certain airfoil is used which is the CLARK-Y airfoil. This airfoil was deemed satisfactory for the estimated Reynolds number present during hover and during cruise. During hover, it is expected that the Reynolds number is higher than during cruise, as the velocities over the propellers are higher. This Reynolds number was then in the range from $0.9 - 1.1 \times 10^6$, therefore it is on the lower end where the CLARK-Y airfoil performs satisfactorily³. In addition to this, certain characteristics were fixed and were not iterated over. This includes the built in geometric twist at the root and tip being 14 [°] and 3 [°] respectively. The chord root and chord tip were simplified to be equivalent to the radius divided by 15, and the radius divided by 25 respectively. The tip Mach number M_t was also given a hard constraint of 0.75 to account for compressibility effects. It is said that if the tip Mach number exceeds 0.75, the drag increases significantly and reduces the overall propeller performance⁴.

With the help of these simplifications, certain characteristics still had to be found for both the cruise and hover situations, as it was deemed that these two flight phases were most crucial in sizing the propellers. For each of these flight phases, it was necessary to find the speed at which the propellers must be turning (rpm), the pitch angle, the power required per propeller and the noise levels for the chosen rpm.

In order to determine these parameters, certain methods were followed. This includes a simplified blade element momentum theory approach (BEMT) to calculate the required pitch angles, rpm and power required. Methods from [33], [34], [35] were used to perform the BEMT analysis and to find the required formulae. In order to analyse the noise levels, it was chosen to follow the procedure from NASA JPL Technical Report [36]. Both methods are first estimate methods and must be tested in theory. However, for the scope of this project, this isn't possible and hence leaves room for development.

At the beginning of this optimisation program, the thrust per propeller required for the hover and for the cruise phase was calculated as follows:

$$T_{hover} = \frac{MTOW \cdot \left(g + \frac{v_{hover}}{t_{TO}}\right)}{N_{props}}, \quad T_{cruise} = \frac{MTOW \cdot g \cdot \frac{D}{L}}{N_{props}} \quad (5.15)$$

This thrust formed the initial requirements which the parameters must converge to. In order to calculate the actual thrust delivered over the propeller, the propeller was split into specific sections to analyse just how much thrust is delivered by each small section. This was done with the help of analysing the lift and drag profile of the CLARK Y

²Excel Calcs, <https://www.excelcalcs.com/calcs/repository/Fluids/50-50-Ethylene-glycol-water--and--air-properties/>, Accessed on 11th of June 2026

²Thyssenkrupp, <https://www.thyssenkrupp-materials.co.uk/density-of-aluminium.html>, Accessed on 11th of June 2026

³Bosch, <https://www.bosch-mobility.com/en/solutions/pumps/electrical-coolant-pump-pce/>, Accessed on 11th of June 2026.

³Airfoil Tools, <http://airfoiltools.com/airfoil/details?airfoil=clarky-il>, Accessed on 8th of June 2026

⁴Learn to Fly, <https://learntoflyblog.com/aerodynamics-high-speed-flight/>, Accessed on 8th of June 2026

airfoil. By looking at the different sections over the propeller length, the stall characteristics over the length can be analysed as well imposing an additional constraint on the propeller sizing, the angle of attack must be below a certain point to prevent stalling of the propeller blades. Overall, this provides the geometry definition of the propeller blade.

5.6.1. BEMT Cruise

To calculate the thrust, power, advance ratio, efficiency and maximum angle of attack, a BEMT analysis is completed for the cruise flight. This method combines the aerodynamic force on each section of the blade from its local angle of attack with momentum theory relating those forces to induced velocities. However, both the blade aerodynamic characteristics impact the induced velocities and vice versa, meaning that the optimiser uses these iteratively.

The following equations are used to set up the method, they include finding the angular velocity, local geometric pitch, rotational and tangential speed at each section.

$$\Omega = \frac{n \cdot 2 \cdot \pi}{60} \quad \beta(r) = \theta_{twist}(r) + \theta_{collective} \quad U_t = \Omega \cdot r \quad (5.16)$$

Next, looking at the induced velocities, two velocities are considered: the axial velocity U_p , and the tangential velocity U_r .

$$U_p = V_\infty \cdot (1 + a) \quad U_r = \Omega \cdot (1 + b) \quad (5.17)$$

a and b in the above equation are iterated over each section and are the axial induction and the tangential induction. From these two velocities, the inflow angle and angle of attack are calculated along with the local relative velocity magnitude (W), which can be determined using $\sqrt{U_p^2 + U_r^2}$.

$$\Phi = \arctan\left(\frac{U_p}{U_r}\right) \quad \alpha = \beta - \phi \quad (5.18)$$

Using the CLARK Y airfoil lift and drag characteristics, the lift and drag coefficients can be found from the calculated angle of attack per section. These coefficients enable the lift and drag to be calculated per section which are later used to calculate the thrust (dT) and the torque (dQ) supplied over the blade.

$$dT = B \cdot (dL \cdot \cos(\phi) - dD \cdot \sin(\phi)) \quad dQ = B \cdot (dL \cdot \sin(\phi) + dD \cdot \cos(\phi)) \cdot r \quad (5.19)$$

If we were to use the formulas above, they would significantly overestimate the amount of thrust that can be delivered. This is because a real blade loses lift near the tip because of pressure differences, this leads to a correction factor having to be imposed or also known as the Prandtl tip loss factor (F) [37].

$$f = \frac{B \cdot (R - r)}{c \cdot r \cdot \sin(\phi)} \quad F = \frac{2}{\pi} \cdot \arccos(e^{-f}) \quad (5.20)$$

F ranges from 1 (centre of the blade) to 0 (at the tip) and reduces the effective momentum available, which raises the induced velocity near the tip. Next, the momentum theory is needed to create an iterative loop for the induced axial velocity, and the previously defined a and b parameters:

$$a = \frac{v_i}{V_\infty} \quad v_i = \frac{dT}{4 \cdot \pi \cdot r^3 \cdot \rho \cdot U_p \cdot \Omega \cdot F} \quad (5.21)$$

These a and b are the updates and are then put back into the induced axial and tangential velocity formulas. The iteration is then as follows: guess a,b -> compute ϕ, α -> Calculate C_l and C_d -> Compute dT and dQ -> update a,b -> repeat. This iterative loop is completed until the induction factors (a,b) stop changing with a tolerance of 10^{-6} . After all iterations are completed, the total thrust, torque and power are obtained by summing all individual components over the sections.

$$T = \sum dT_i \Delta r_i \quad Q = \sum dQ_i \Delta r_i \quad P = \Omega \cdot Q \quad (5.22)$$

And hence, the non dimensionalised propeller coefficients are found using:

$$c_T = \frac{T}{\rho \cdot n^2 \cdot D^4} \quad c_P = \frac{P}{\rho \cdot n^3 \cdot D^5} \quad J = \frac{V_\infty}{n \cdot D} \quad (5.23)$$

Where n represents the revolutions per second, D the propeller diameter in [m], and J the advance ratio. Finally, this enables the propulsive efficiency to be calculated as follows.

$$\eta = \frac{c_T \cdot J}{c_P} = \frac{T \cdot V_\infty}{P} \quad (5.24)$$

All in all, these iterations provide the available characteristics during the cruise from all the propeller blades.

5.6.2. BEMT Hover

To calculate the necessary parameters during hover, the same method was applied as with cruise with a few alterations following methods explained by Leishman in [38]. Mainly, during hover there is zero forward speed, simplifying the analysis which must be completed. The only flow through the disc is the induced velocity which is created by the propeller itself. This eliminates the needs for iterations, as the momentum equations are combined into a single closed form expression for the inflow, evaluated at each section of the blade [39]. To start, all quantities are non dimensionalised by the tip radius and tip speed to simplify the full analysis.

Hence the non dimensional radius, and local solidity (blade area fraction at that station) equate to:

$$\hat{r} = \frac{r}{R} \quad \sigma(\hat{r}) = \frac{B \cdot c(\hat{r})}{\pi \cdot R} \quad (5.25)$$

The airfoil's lift curve slope $\frac{dC_l}{d\alpha}$ is assumed to be a linear function and is taken from the CLARK Y aerodynamic properties⁵. To simplify the combined inflow equation, the airfoil's lift curve slope is said to be equivalent to c . This inflow equation finds the induced flow at each section which later must be added to determine the inflow ratio and hence the induced downward velocity normalised by the tip speed. This equation replaces the iterative loop which was required for cruise.

$$\lambda(\hat{r}) = \sqrt{\left(\frac{\sigma \cdot c}{16}\right)^2 + \frac{\sigma \cdot c}{8} \cdot \theta \cdot \hat{r}} - \frac{\sigma \cdot c}{16} \quad \lambda = \frac{U_p}{\Omega \cdot R} \quad (5.26)$$

Similar to as to cruise flight, the non dimensionalised Prandtl tip loss factor is also applied to the hover situation and applied to the inflow equation. Then from the inflow, the inflow angle and angel of each section are:

$$\phi(\hat{r}) = \arctan\left(\frac{\lambda}{\hat{r}}\right) \quad \alpha = \theta - \phi \quad (5.27)$$

Using this formula for the angle of attack, the airfoil polar gives $C_l(\alpha)$ and $C_d(\alpha)$, which can then be used to compute the thrust and power per section in non dimensional form:

$$dc_T = 0.5 \cdot \sigma \cdot C_l \cdot \hat{r}^2 \cdot d\hat{r} \quad dc_P = 0.5 \cdot \sigma \cdot (\phi \cdot C_l + C_d) \cdot \hat{r}^3 \cdot d\hat{r} \quad (5.28)$$

Inside the power coefficient, the two terms are the induced power (cost of producing lift against induced inflow) and the profile power (cost of dragging the blade through the air) respectively. Summing these over the entire blade span, give the total coefficients during hover. To then dimensionalise the thrust and power available, these coefficients are used as follows:

$$T = c_T \cdot \rho \cdot A \cdot V_{tip}^2 \quad P_{ideal} = \frac{T^{1.5}}{\sqrt{2 \cdot \rho \cdot A}} \quad (5.29)$$

Overall, these give the thrust and power available during a hover scenario. However, the power delivered by the propellers isn't ideal and a correction factor must be added to calculate the actual power usage per propeller. This is done by dividing the calculated power by a figure of merit that is now estimated to be 0.75 [40]. This then provides a realistic value for the actual power of the propellers.

5.6.3. Pitch Solver

Next, to find the pitch that is able to deliver the required thrust, a pitch solver optimiser is made to calculate the pitch during hover mode and during cruise mode. During this optimisation, the stall angle is also taken into account and it is ensured that the pitch position enables a thrust before this stall position. It was decided to have a variable pitch system to optimise the eVTOL's performance.

To iterate across different pitch angles, the rpm ceiling is defined by using:

$$n = \frac{M_{tip} \cdot a \cdot 2 \cdot \pi}{R \cdot 60} \quad (5.30)$$

Then using this maximum rpm initially and the thrust requirements, different pitch angles are matched to then estimate a pitch angle using the least power for both cruise and hover scenarios. In addition to this, especially during cruise, the rpm is also iterated to find an optimum combination of rpm and pitch angle.

5.6.4. Noise Constraints

Finally, in order to calculate the noise level produced from the propeller, research completed by NASA JPL laboratory to calculate the sound pressure levels [36]. The equation to calculate these sound levels is as follows:

$$SPL_{total} = L_1 + 20 \cdot \log\left(\frac{4}{B}\right) + 40 \cdot \log\left(\frac{15.5}{D}\right) + L_2 + L_3 - 20 \cdot \log(r - 1) \quad (5.31)$$

⁵Airfoil Tools, <http://airfoiltools.com/airfoil/details?airfoil=clarky-il>

In this equation, the first term L_1 is a correction term for the shaft horsepower which can be found from the motor used. The next two terms are corrections for the number of blades used, B and the diameter of the propeller, D . L_2 is a correction factor for the tip Mach number. This value is found using figure B-3 in [36]. Next, L_3 is the correction factor for the fuselage position with respect to the propellers. This value is obtained using figure B-4 in [36]. Finally, the last term is the correction term for the spherical spreading of sound where r is the distance from the centre of the propeller. It must be noted that in this formula, all variables must be provided in imperial units.

However, although this formula gives us the sound pressure levels per propeller, it is required to find the total sound pressure levels using:

$$SPL_{\text{overall}} = 10 \cdot \log_{10} \left(\sum_n 10^{L_n/10} \right) \quad (5.32)$$

This overall sound pressure level, is from a specific location [36]. To analyse the effects of a varying rpm, the blade passage frequency equation can be used. As this frequency, determines a correction factor using figure B-6 from [36] providing the harmonic level in dB. To comply with the noise restrictions, this overall noise level must be converted to dBA.

5.6.5. Final Propeller Design

It was decided to make the airfoil out of carbon fibre as this was deemed strong enough to complete the current thrust required according to H. Luong et al, 2025 [41].

The weight of a propeller is hard to calculate geometrically as there are multiple angles to take into account. Therefore, to calculate its weight, the propeller was plotted in Fusion and the weight was calculated there. This estimated the weight of 1 propeller to be 11.7 [kg].

In addition to this, when running the optimiser, it was realised that the rpm was extremely low during cruise which leads to a very high torque being generated by the motor shaft. For this reason, it was decided to fix the cruise rpm to 800 rpm. This led to a minimal impact on the actual power and thrust generated, hence seemed like a feasible option. Overall, all final characteristics of the propeller are shown in Table 5.9 and compared to characteristics found for Joby S4. It must be noted however, that Joby has 5 blades per propeller, whilst Vold-e has 8 blades per propeller. In addition to this, to assess the validity of the model used, some of Joby's parameters were put into the optimiser to find the difference between the actual values and the simulated values. To receive the values for Joby, multiple sources were used [42], [43]⁶.

Table 5.9: Final Propeller Characteristics

	Vold-e		Simulated Joby S4		Joby S4	
	Cruise	Hover	Cruise	Hover	Cruise	Hover
Diameter [m]	1.9		2.9		2.9	
Blade count	8		5		5	
Disc Loading [kg/m ²]	125		64		46	
Blade Pitch [°]	41	12	60	10	-	-
RPM	800	2200	300	1200	250-500	800
Power [kW]	15	100	20	72	14	70
Noise at 15 m [dBA]	73	94	73	89	-	85
Noise at 30 m [dBA]	66.4	88	67	83	-	80
Noise at 500 m [dBA]	42	64	43	60	45	75

From this table, it is clear that Vold-e does not comply with the imposed restrictions at the 15[m] distance. It is over by 14 [dBA], which is quite extreme. However, when comparing the simulated Joby to the actual Joby, the largest difference is the rpm simulated during hover. This may be the cause for the increase in noise prediction during the hover scenario. This may also be the case for Vold-e meaning that in reality, the noise emitted may be lower than what is predicted.

5.7. Motor Selection

In order to provide the required power to the propellers as mentioned in Subsection 5.6.5, a motor must be chosen that can provide this power as well as have a variable pitch system. To ensure that at any moment the propulsion system can provide enough power, it is estimated that each motor must be able to deliver 20% more power than what

⁶AOPA, <https://www.aopa.org/news-and-media/all-news/2023/april/pilot/joby-s4-coming-to-you-in-2025>, Accessed on 11th of June 2026

was calculated in Subsection 5.6.5. This is mainly because it is unknown how much power will be needed for the transition phase, as well as to account for any wind gust effects on the eVTOL and to overcome these forces.

Another important factor is to know the maximum torque exerted on the motor. This can be calculated by dividing the maximum power by the angular velocity Ω , in this case this would be the angular velocity during take-off and equate to 380 [Nm]. All motors chosen have also been ensured to be possible with only air cooling, which simplifies the system and ensures that no extra weight is needed for the propellers. With these few restrictions, a trade-off must be completed to choose the correct motor. In Table 5.10, all possible options are shown.

Table 5.10: Possible Motor options

Motor	P_{peak} [kW]	T_{peak} [Nm]	RPM_{peak}	Peak dur. [s]	P_{cont} [kW]	T_{cont} [Nm]	Mass [kg]	DxL [mm]	Power Dens. [kW/kg]
EMRAX 268 LV (no inv.) ¹	210	500	4500	60	94	200	21.4	268x94	4.4
EMRAX 348 HV (no inv.) ¹	145	1000	3250	30	82	350	43.1	348x112	1.9
AF340 EVO (no inv.) ²	450	1200	5000	60	x	780	122	390x333.4	x
MAGiDRIVE 150 (incl. inv.) ³	150	700	4200	x	120	400	24	381x152.4	5

In order to complete the trade-off, certain weights are assigned to certain categories. These categories are: P_{peak} , T_{peak} , the power density, if the motor includes an inverter and the last category is the overall mass of the motor.

Each trade-off option is scaled per category with a value of 1 - 4, with 4 being the best performing within the category and 1 the worst.

Table 5.11: Motor Trade-off

Motor/Criteria	P_{peak} [0.4]	T_{peak} [0.1]	Power Density [0.3]	Mass [0.2]	Total Score
EMRAX 268	3	1	3	4	3
EMRAX 348	1	3	2	2	1.7
EVO AF340	4	4	1	1	2.5
MAGiDRIVE 150	2	2	4	4	2.6

Using this table, it is clear that the EMRAX 268 is the best fit for Vold-e. The largest reason why this scores better than the MAGiDRIVE 150, is that the EMRAX provides a larger margin for safety during one engine inoperative. When looking at this condition, the maximum thrust which the propellers are able to achieve using the MAGiDRIVE is approximately 4500 [N], whilst the thrust needed for take-off is approximately 3600 [N]. This gives an approximate 20% margin for safety which is quite small. When using the EMRAX 268, the maximum thrust which can be achieved is as high as 5500 [N]. This extra margin of thrust is necessary for stability and control reasons and hence proves that EMRAX 268 is a good fit for Vold-e.

Choosing the EMRAX motor requires a controller that fits with it. Looking at the EMRAX specifications, the controller that matches this is the HV-850⁷. This controller also satisfies the peak voltage constraints and has a total mass of 6.8 [kg]. This leaves the motor with a weight of 28.2 [kg].

5.8. Discussion

The sizing is energy driven, but rests on a few assumptions to revisit in detailed design. The pack depends on the emerging SA504 cell, whose 500 cycle life sets a yearly replacement interval, and the ECS and thermal models are first-order estimates. The largest open item is propulsion noise: the blade element and NASA JPL acoustic methods are unvalidated estimates, and the predicted hover noise of 94 dBA at 15 m exceeds the limit by about 14 dBA. The Joby S4 comparison in Table 5.9 suggests the model overpredicts hover noise, so actual level is likely lower, but a quieter design and validated acoustic modelling are needed for compliance. The transition phase is not modelled explicitly and is covered by a 20% motor power margin pending a dedicated analysis.

¹Emrax, https://emrax.com/wp-content/uploads/2025/08/EMRAX_268_datasheet_v1.6.pdf, Accessed on 9th of June 2026

²Off the Grid Sun, <https://offthegridsun.com/Brushless-Motor/AF340-EVO-Axial-Flux-Standard-Electric-280kw-Motor>, Accessed on 9th of June 2026

³For this motor, the average value for the torque range, mass, and length was taken, Magically, <https://www.magically.biz/downloads/magic-all-magidrive-datasheet.pdf>, Accessed on 9th of June 2026.

⁷Emrax, https://emrax.com/wp-content/uploads/2025/03/List_of_recommended_controllers_V1.3.pdf, Accessed on 11th of June 2026

6 Vehicle Dynamics

This chapter provides an overview of the methods and analyses that were used to quantify and assess the flight performance of the aircraft in its main operating configurations. As such, the aircraft's static and dynamic stability in both the longitudinal and lateral directions are investigated, as well as the controllability characteristics.

First, Section 6.1 assesses the static stability of Vold-e in wing-borne and VTOL flight. Next, Section 6.2 evaluates the dynamic stability, control-surface design, and handling qualities of the aircraft at the cruise condition. Then, Section 6.3 describes how the static-stability requirements are incorporated into the design process. Finally, Section 6.4 and Section 6.5 evaluate the aircraft's mission-relevant performance through the payload-range, V-n, turn, climb and descent analyses, before concluding with a brief discussion of the transition procedure in Section 6.6.

6.1. Stability

The stability of an aircraft forms one of the key aspects that drive aircraft design. Stability considerations, namely, greatly impact the safety of a design; if an aircraft is stable and controllable, then it can be flown safely; otherwise it will have a hard time remaining airborne and is deemed unsafe.

The aircraft designed in this report switches between two different configurations and modes of operation throughout its mission profile. These different configurations require distinct investigations regarding aircraft stability, and will be presented in the following sections.

6.1.1. Description on Static Longitudinal Stability and the Neutral Point

The static longitudinal stability of an aircraft describes the behaviour of the aircraft when it is disturbed and deviates from its trimmed condition through an increase in the angle of attack. The aircraft is longitudinally stable when the moment curve has a negative slope, i.e. $C_{M_\alpha} < 0$. Just as for conventional aircraft, this condition is satisfied for a Prandtl Plane when the centre of gravity (CG) is placed in front of the neutral point (NP). The neutral point corresponds to the point where the total change in lift acts, and its location can be computed using the method from [44]. See Figure 6.1 for the free-body diagram for which this equation holds.

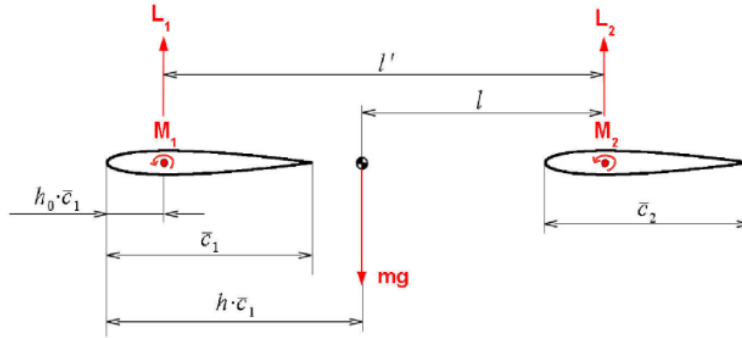


Figure 6.1: Forces and moments working on the box wing aircraft, as adapted from [44].

$$h < h_0 + \frac{dC_{L_2}}{dC_L} \frac{\bar{V}'}{\bar{c}_1'}, \quad \text{where} \quad \bar{V}' = \frac{S_2 l'}{S \bar{c}}, \quad \bar{c} = \frac{S_1 \bar{c}_1 + S_2 \bar{c}_2}{S_1 + S_2} \quad (6.1)$$

The quantity $\frac{dC_{L_2}}{dC_L}$ considers the change in lift of the aft wing with a change in total aircraft lift, and can be defined as a lift slope. This slope depends on the aerodynamic characteristics of the aft wing, as well as the system in which it is integrated. This quantity therefore considers many influencing factors, and its computation requires multiple steps. First, we rewrite the relation:

$$\frac{dC_{L_2}}{dC_L} = \frac{dC_{L_2}}{d\alpha_2} \frac{d\alpha_2}{d\alpha} \frac{d\alpha}{dC_L} \quad (6.2)$$

The first term on the right hand side is derived from the preliminary wing planform design and the selected airfoil, and has a value of 5.0 [1/rad.]. The second term can be rewritten to $1 - \frac{d\epsilon}{d\alpha}$. Here, $\frac{d\epsilon}{d\alpha}$ corresponds to the downwash of the first wing that alters the effective angle of attack that is perceived by the aft wing. This mathematical relation that is used to compute this quantity is derived from [45] as presented in [46]:

$$\frac{d\epsilon}{d\alpha} = 4.44 \cdot (k_{A,1} k_{\lambda,1} k_H \sqrt{\cos \varphi_{25,1}})^{1.19} \cdot \frac{(C_{L_\alpha})_{1,M}}{(C_{L_\alpha})_{1,M=0}} \quad (6.3)$$

with:

$$k_{A,1} = \frac{1}{A_1} - \frac{1}{1 + A_1^{1.7}}, \quad k_{\lambda,1} = \frac{10 - 3\lambda_1}{7}, \quad k_H = \frac{1 - \left| \frac{z}{b_1} \right|}{\sqrt[3]{\frac{2l'}{b_1}}} \quad (6.4)$$

Here, z corresponds to the vertical distance between the chord lines of the front and aft wings in [m]. The quantity $\varphi_{25,1}$ refers to the sweep angle of the quarter-chord line of the wing.

Furthermore, the quantity of $(C_{L\alpha})_{1,M}$ can be found through the Prandtl-Glauert correction factor:

$$C_{L\alpha,M} = \frac{C_{L\alpha,0}}{\sqrt{1 - M_\infty^2}} \quad (6.5)$$

Now, only the last term that is needed to find $\frac{dC_{L2}}{dC_L}$ remains. This term is the inverse of the slope of the lift curve of the whole aircraft, which is denoted by a . This quantity is not only dependent on the horizontal wings. The aircraft nose and fuselage namely interfere with the flow field around the wings and contribute to the overall lift, though these contributions are relatively minor. Furthermore, the downwash and wing tip vortices of the front wing interfere with the flow field of the aft wing. These effects are accounted for with the following relations:

$$a = s_1 \left(\frac{dC_L}{d\alpha} \right)_1 (K_N + K_{W(B)} + K_{B(W)})_1 \frac{S_{e,1}}{S_1} + s_2 \left(\frac{dC_L}{d\alpha} \right)_2 (K_{W(B)} + K_{B(W)})_2 \frac{q_2}{q_\infty} \frac{S_{e,2}}{S_1} + \left(\frac{dC_L}{d\alpha} \right)_{W_2(v)} \quad (6.6)$$

with:

$$\left(\frac{dC_L}{d\alpha} \right)_{W_2(v)} = \frac{\left(\frac{dC_L}{d\alpha} \right)_1 \cdot \frac{S_{e,1}}{S_1} \cdot \left(\frac{dC_L}{d\alpha} \right)_2 \cdot \frac{q_2}{q_\infty} \cdot K_{W(B),1} \cdot I_{V_{W_1(W_2)}} \cdot \left(\frac{b_2}{2} - \frac{d_2}{2} \right)}{2\pi A_2 \left(\frac{b_1}{2} - \frac{d_1}{2} \right)} \quad (6.7)$$

Here, $S_{e,i}$ refers to the exposed wing area, and d_i refers to the local effective fuselage diameter. The ratio $\frac{q_2}{q_\infty}$ is estimated to be 0.9 at the present stage in the design. Furthermore, the factors denoted by K represent additional lift effects generated by the aircraft nose, and the mutual interferences of the wing and the aircraft body (i.e. the fuselage). The effect of the nose is quantified through the following relation:

$$K_N = \frac{\pi d_1^2}{2 \left(\frac{dC_L}{d\alpha} \right)_1 S_{e,1}} \quad (6.8)$$

Please note that the aft wing is sufficiently placed backwards such that the aircraft nose affects only the front wing. Furthermore, the sum of the wing-body interference is computed through:

$$K_{W(B),i} + K_{B(W),i} = \left(\frac{d_i}{b_i} + 1 \right)^2 \quad (6.9)$$

The front wing is placed on the underside of the fuselage, and the wing-body interference effects here thus cannot be neglected. However, the aft wing is placed on top of the vertical tail. Furthermore, the fuselage is shaped like a droplet. This means that there are relatively minor wing-body interference effects that are complex to model. Therefore, the wing-body interference effects of the aft wing are neglected, and the sum $(K_{W(B)} + K_{B(W)})_2$ and are both set to a value of 1. Please refer to Figure 3.7 for a visualisation of the aircraft configuration.

The factor $K_{W(B),i}$ also appears separate from $K_{B(W),i}$ and can be computed through:

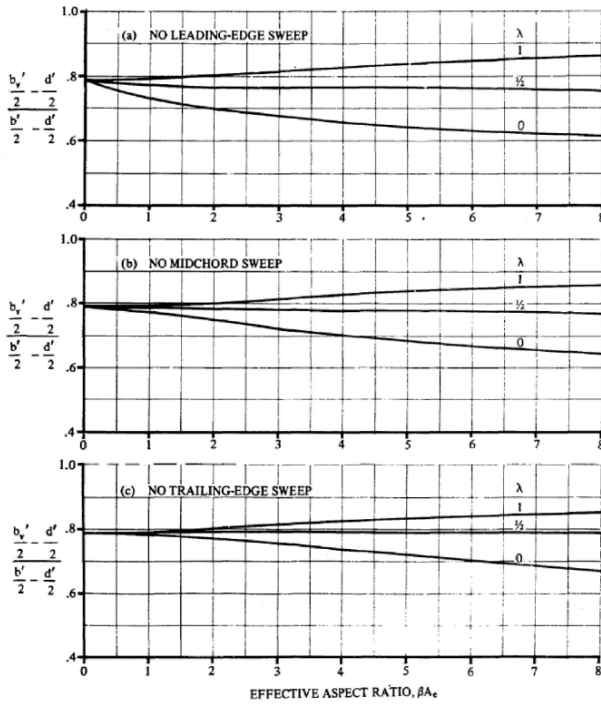
$$K_{W(B),i} = 0.8 \frac{d_i}{b_i} + 1 \quad (6.10)$$

The remaining interference factor is the vortex interference factor $I_{V_{W_1(W_2)}}$. This quantity considers the effect of the trailing vortices of the front wing regarding the total lift of the aircraft [45]. The determination of the value for this quantity is a somewhat complicated process that comprises multiple steps. First, the lateral position of the trailing vortices needs to be determined. At the current stage of the design, a taper ratio of 0.45 and no leading edge sweep have been selected. However, there is no graph that corresponds to this specific taper ratio. Therefore, the graph that corresponds to a taper ratio closest to 0.45 is to be used. This leads to the use of the middle line in chart a in Figure 6.2a.

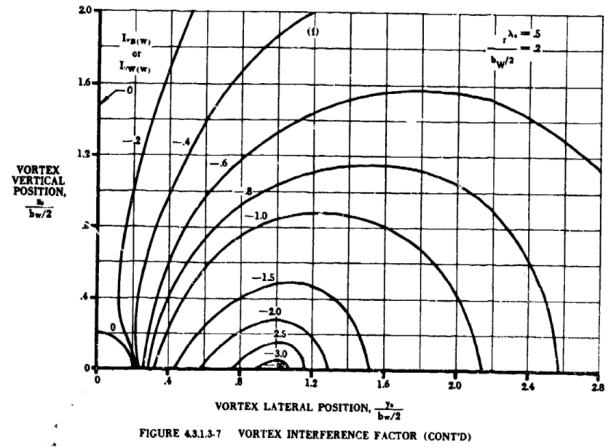
The parameter displayed on the vertical axes refers to the lateral vortex position. As can be seen in the chart, the graph that corresponds to a taper ratio of 0.5 and no leading edge sweep is relatively constant. Therefore, it can be assumed that the lateral vortex position has a value of 0.75, regardless of the effective aspect ratio.

Now that the trailing vortex position is known, the vortex interference factor can be determined. This is done through the use of another chart. Just like before, there is no chart that exactly matches the configuration of

the aircraft that is to be designed. Furthermore, DATCOM methods are, strictly speaking, only valid for idealised conventional aircraft. The aircraft design that is considered in this report deviates from conventional aircraft designs, and the charts are thus used for preliminary rough estimations. This warrants the use of the chart displayed below in Figure 6.2b.



(a) Lateral position of the trailing edge vortices of the front wing, depending on effective aspect ratio and wing and body geometry.



(b) Vortex interference factor, depending on both lateral and vertical vortex position as well as wing and fuselage geometry.

Figure 6.2: DATCOM charts used for vortex position and interference estimation [45].

According to Schiktanz, 2011:

“The vertical distance between the forward wing trailing vortices and the aft wing is assumed to be the same as the vertical distance between these two wings. This can be justified with the vortices trailing in the free stream direction assuming that the angle between the center line of the aircraft and the free stream is zero.”

By using a value of 0.16 for h/b in combination with the statement from the quote above, a value of -1.75 for the vortex interference factor I_v is determined.

Now that the methodologies to determine the values of the relevant parameters are known, the stability of each aircraft configuration can be determined.

6.1.2. Hover Stability With OEI

Apart from being stable in flight where the wings generate lift, the aircraft needs also to be stable in its VTOL-configuration. The most important flight states for which this stability needs to be investigated are hover and steady vertical climb or descent. In this context, stability refers to moment equilibrium about the CG. Then, the stabilisation problem comes down to determining the required thrust per propeller for a given aircraft configuration and weight such that the following conditions are satisfied:

1. $\sum T_i = W$
2. $\sum M_x = 0$
3. $\sum M_y = 0$
4. $\sum M_z = 0$

It is important to note that the last condition is not considered at the current stage of the design and should be treated in later design iterations. The remaining conditions can only be satisfied for a specific CG-envelope that depends on several factors. These factors are given below:

1. Number of propellers;
2. Longitudinal and lateral positions of the propellers;
3. Maximum propeller thrust at sea level;
4. MTOW.

The design must be adjusted such that the actual CG always lies in this envelope during the aforementioned flight states to guarantee safe flight.

As mentioned, the size and location of the CG-envelope depends on the number of propellers. Thus, when the number of available propellers changes, then also the CG-envelope changes. Based on the risk analysis performed in Subsection 12.1.1, the case in which one engine fails (TR-PP-06), needs to be investigated and designed for. This is indeed the critical case. Namely, the moment generating capability of the system decreases when one engine goes inoperative, consequently decreasing the allowable CG-envelope. According to the same logic, the failure of the out-board engines is critical, and the problem scope can thus be limited to failure of these engines. It is important to note that the noise constraints are considered irrelevant in an emergency situation, and are therefore neglected in this analysis.

At the current state of the design, the aircraft contains six engines in total. For the OEI-case, five engines remain available. Furthermore, there are three kinetic conditions which need to be satisfied to achieve steady flight. Thus, there are more variables than equations to solve. This means that the problem is under-constrained and is thus not uniquely solvable. The problem becomes a linear optimisation problem: find the CG-envelope of maximum size by altering the generated thrust of the separate propellers, while satisfying the aforementioned conditions.

Consequently, a model has been constructed that finds the CG-envelope of maximum possible size for a given aircraft configuration. The verification and validation of this model is discussed in Chapter 11.

The model makes use of a Cartesian coordinate system. The x-axis coincides with the aircraft symmetry line, and the system origin is located at the aircraft nose. Naturally, the y-axis is perpendicular to the x-axis and intersects at the origin. See Figure 6.3.

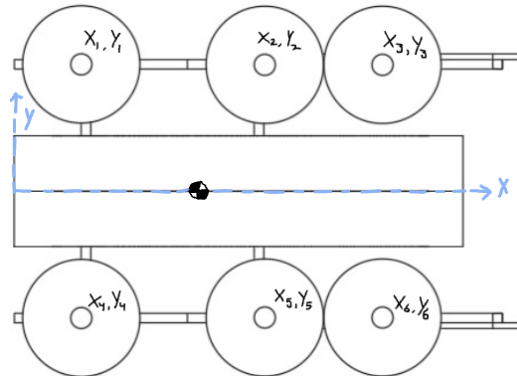


Figure 6.3: Top-view of the aircraft in its VTOL-configuration.

The solutions that the model yields adhere to the aforementioned constraints and conditions. The CG lies on the x-axis, and the second condition can thus be rewritten to:

$$\Sigma T_{+y} = \Sigma T_{-y} = \frac{W}{2} \quad (6.11)$$

Furthermore, the third condition can be rewritten to the form in Equation 6.12.

$$x_{CG} = \frac{\Sigma x_i T_i}{W} \quad (6.12)$$

This relation yields the CG-location for which the aircraft is trimmed in pitch for a given thrust distribution. This thrust distribution also inherently satisfies the condition that total thrust equals the aircraft weight. Furthermore, this relation is used to define the solution procedure. Namely, a linear solver is used that finds the minimum and maximum allowable CG-locations of all possible thrust distributions that satisfy the predefined conditions and the thrust boundary. To this end, the method 'linprog' of the Python module SciPy is used.

This Python method is only capable to minimise a given objective. This means that it can only find the minimum allowable CG-location. To find the maximum trimmable CG-location, the negative objective is minimised. (Making the objective negative effectively produces the aircraft configuration that is mirrored in the y-axis.) This yields the maximum CG-location as a negative number. To obtain the correct value, the found quantity is simply multiplied by -1. The implementation of this method and the corresponding model are described in Section 6.3.

6.2. Dynamic Stability: Cruise

To assess the overall dynamic stability, certain aerodynamic coefficients and stability derivatives have to be calculated. These derivatives are all calculated by empirical and semi-empirical methods that are present in the USAF DATCOM Stability and Control book [45]. This section addresses the dynamic response that includes the natural eigenmodes of the linearised cruise model and their compliance with the stability requirements. The two main requirements include whether the aircraft is dynamically stable about the cruise trim point and whether the modes fall within accepted handling quality bounds. The whole process is carried out in the cruise design point. Under these symmetric trim conditions the equations of motion includes longitudinal (symmetric) and lateral (asymmetric) sets.

6.2.1. DATCOM Methods

The aerodynamic and control derivatives were estimated using the subsonic methods of the USAF Stability and Control DATCOM [45], built up from the section level through the wing, wing-body and full aircraft. This handbook level fidelity has been found to be appropriate for the current design stage whose purpose is to confirm the modes exist and are well behaved.

Before evaluating and determining all of the derivatives, two extensions are added because of the Prandtl-plane configuration. First, the overall lifting complex is treated as two staggered surfaces, with per wing rolling, sideslip and rotary contributions evaluated individually and summed. Second, the directional surfaces consist of a twin vertical tail and the winglets that are modelled as additional vertical lifting surfaces contributing to the lateral derivatives. These treatments fall outside DATCOM's validated envelope. Therefore, the derivatives that are effected by those surfaces should be confirmed against CFD analyses. The current analysis is conducted to be as close as possible to the reality. However, this can be noted as a limitation of the analysis and can be addressed in more detail in the future. The derivative set was verified against the sign requirements for a stable aircraft. All twelve most crucial stability signs are satisfied and their required signs are stated in Table 6.1.

Table 6.1: Required signs of the stability derivatives.

Derivative	C_{m_α}	C_{m_q}	C_{L_q}	C_{n_β}	C_{ℓ_β}	C_{Y_β}	C_{ℓ_p}	C_{n_r}	C_{Y_p}	C_{n_p}	C_{Y_r}	C_{ℓ_r}
Required sign	< 0	< 0	> 0	> 0	< 0	< 0	< 0	< 0	< 0	< 0	> 0	> 0

6.2.2. State Space Representation

The model is trimmed at the cruise condition by solving for the angle of attack, elevator deflection, and throttle setting that null the body axis force and moment derivatives. The nonlinear model is then linearised about this trim point to obtain the symmetric and asymmetric state space systems.

$$\begin{bmatrix} C_{xu} - 2\mu_c D_c & C_{x\alpha} & C_{z0} & C_{xq} \\ C_{zu} & C_{z\alpha} + (C_{z\dot{\alpha}} - 2\mu_c) D_c & -C_{x0} & C_{zq} + 2\mu_c \\ 0 & 0 & -D_c & 1 \\ C_{mu} & C_{m\alpha} + C_{m\dot{\alpha}} D_c & 0 & C_{mq} - 2\mu_c K_{yy}^2 D_c \end{bmatrix} \begin{bmatrix} \hat{u} \\ \alpha \\ \theta \\ \frac{q\bar{c}}{V} \end{bmatrix} = \begin{bmatrix} -C_{x\delta_e} \\ -C_{z\delta_e} \\ 0 \\ -C_{m\delta_e} \end{bmatrix} \delta_e \quad (6.13)$$

The asymmetric equations of motion can be represented in matrix form as given as the following,

$$\begin{bmatrix} C_{Y\beta} + (C_{Y\dot{\beta}} - 2\mu_b) D_b & C_L & C_{Yp} & C_{Yr} - 4\mu_b \\ 0 & -\frac{1}{2} D_b & 1 & 0 \\ C_{\ell\beta} & 0 & C_{\ell p} - 4\mu_b K_{xx}^2 D_b & C_{\ell r} + 4\mu_b K_{xz} D_b \\ C_{n\beta} + C_{n\dot{\beta}} D_b & 0 & C_{np} + 4\mu_b K_{xz} D_b & C_{nr} - 4\mu_b K_{zz}^2 D_b \end{bmatrix} \begin{bmatrix} \beta \\ \phi \\ \frac{pb}{2V} \\ \frac{rb}{2V} \end{bmatrix} = \begin{bmatrix} -C_{Y\delta_a} \\ 0 \\ -C_{\ell\delta_a} \\ -C_{n\delta_a} \end{bmatrix} \delta_a + \begin{bmatrix} -C_{Y\delta_r} \\ 0 \\ -C_{\ell\delta_r} \\ -C_{n\delta_r} \end{bmatrix} \delta_r \quad (6.14)$$

In the symmetric case $\bar{u}$ and $\bar{x}$ are defined as:

$$\bar{u} = [\delta_e] \quad \bar{x} = [\hat{u} \quad \alpha \quad \theta \quad q]^T \quad (6.15)$$

and in the asymmetric case:

$$\bar{u} = [\delta_a \quad \delta_r]^T \quad \bar{x} = [\beta \quad \phi \quad p \quad r]^T \quad (6.16)$$

The *state space representation* of the system that is aimed is as follows:

$$\dot{\bar{x}} = A\bar{x} + B\bar{u} \quad \bar{y} = C\bar{x} + D\bar{u} \quad (6.17)$$

where $\bar{x}$ the state vector, $\bar{u}$ the input vector and $\bar{y}$ the output vector.

$$C_1 \dot{\bar{x}} + C_2 \bar{x} + C_3 \bar{u} = \bar{0}. \quad (6.18)$$

The system matrices A and B can be determined from:

$$A = -C_1^{-1}C_2 \quad B = -C_1^{-1}C_3 \quad (6.19)$$

Both A matrices are assembled directly from the non-dimensional derivatives through the standard $A = -C_1^{-1}C_1$ construction, using a single reference triplet (S_{ref} , b_{ref} , c_{ref}) consistent between the derivative set and the equations of motion.

After that, the dynamic modes are obtained as the eigenvalues of the two A matrices. The symmetric system yields two complex conjugate pairs, identified by frequency as the short period and phugoid modes. The asymmetric system yields one complex pair (the Dutch roll) and two real poles (aperiodic roll and spiral). Each mode is characterised by its eigenvalue λ , undamped natural frequency ω_n , damping ratio ζ and period T or time constant τ . Unstable modes are quantified by their time to double amplitude T_2 . In Subsection 6.2.7 the modes are assessed against the Level-1 boundaries of MIL-F-8785C for Category B (cruise) flight¹.

6.2.3. Vertical Tail Sizing

The vertical tail is governed by two driving constraints: the directional stability requirement on C_{n_β} and a ground-footprint constraint that caps the overall height of the aircraft to 3 [m]. The fins that make up the vertical tail must deliver the required directional stiffness without exceeding this height.

The joiners that close the box are large vertical surfaces, called winglets, but they cannot be relied upon to provide directional stability. Longitudinally, their aerodynamic centre lies close to the centre of gravity, so their yaw moment arm is short. Their net contribution to C_{n_β} is therefore small and uncertain and at the aft centre of gravity it can become destabilising. The directional stability requirement is consequently met by the dedicated fins, with the joiner contribution treated as a non-conservative bonus rather than a sized source of stiffness.

The first candidate was a single large fin. The largest single fin that fits beneath the height cap provides too little side area at the available moment arm: its C_{n_β} falls short of the requirement. The configuration was therefore changed to a twin vertical tail, in which two fins supply the required total side area within the same height limit, and their outboard, canted placement keeps them in cleaner flow away from the fuselage wake. Because directional stiffness scales with the product of fin side area and moment arm, the twin arrangement lifts C_{n_β} above the requirement while respecting the height constraint. Table 6.2 compares the relevant derivatives before and after the change.

Table 6.2: Lateral stability derivatives for the single- and twin-tail configurations at cruise

Derivative	Single tail	Twin tail
C_{n_β}	+0.070	+0.152
C_{Y_β}	−1.93	−2.23
C_{n_r}	−0.060	−0.103
C_{l_β}	−0.104	−0.124
Total fin area S_v	2.32 [m ²]	3.77 [m ²]
Fin height (each)	1.6 [m]	1.3 [m]

The comparison shows that having a twin tail significantly increases lateral controllability; C_{n_β} is more than doubled. The twin tail increases the total side area within the height cap, lifting C_{n_β} from below the requirement to comfortably above it and improving the sideforce (C_{Y_β}) and yaw damping (C_{n_r}) derivatives in the same step. Directional stability is thus improved significantly achieving better damping in lateral motions (Subsection 6.2.6).

6.2.4. Control Surface Sizing

The aircraft carries three families of aerodynamic control surfaces: flapelevons (flap + elevator) on the aft wing, flap-erons (flap + ailerons) on the front wing, and a rudder on each of the twin fins. The chord ratio c_f/c , where c_f is the chord length of the surface, of each surface is selected to provide the required control authority at acceptable hinge moments. The resulting geometry is summarised in Table 6.3. The wings carry two folding hinges, so the control surfaces are positioned in a way such that they do not coincide with hinges. Their spanwise location is defined by η .

Table 6.3: Control-surface arrangement.

Surface	Location	Function	c_f/c	Deflection	Span η
Flapelevon	Aft wing	Pitch, flap	0.3	up to 50°	0.77
Inner flaperon	Front wing	Roll, flap, pitch	0.3	up to 90°	0.35
Outer flaperon	Front wing	Roll, flap	0.3	up to 50°	0.77
Rudder (×2)	Vertical tails	Yaw	0.2	up to 30°	full fin

⁰MIL-F-8785C, https://everyspec.com/MIL-SPECS/MIL-SPECS-MIL-F/MIL-F-8785C_5295/, Accessed on 3th of June 2026

In VTOL mode, the inner propellers on the front wing do not have a clear path. To combat this, the inner flaperon is designed to deflect up to 90° so that the thrust is usable fully.

The flaperons and flapelevons are designed to be used simultaneously to have increased dynamic benefits. The large pitching moment that is created by having front wing ahead of centre of gravity is beneficial for pitch authority. This creates two phenomena. Firstly, the available pitch control power C_{m_δ} is increased, causing faster damping of the short period motion with fast and small deflections. Secondly, pitch inputs excite short period cleanly without turning it into phugoid. Therefore, together they make the pitch response of the aircraft better behaved than a single aft elevator.

Each fin carries a rudder, giving the aircraft substantial yaw control authority ($C_{n_{\delta_r}}$). This authority is essential for the Prandtl Plane, because the winglets have minimal control on yaw if they have rudder onto them. This is because their longitudinal moment arm to centre of gravity is too small to have a strong yaw authority. Thus, decent amount of yaw authority is needed both to trim and manoeuvre directionally and to have strong yaw damping against the Dutch roll motion. This has been proven to be achievable by the twin vertical tail

6.2.5. Dynamic Longitudinal Stability

The eigenvalues of the symmetric system corresponding to the short period and phugoid modes with their characteristics are given in Table 6.4.

Table 6.4: Longitudinal eigenmodes at cruise

Mode	Eigenvalue λ [1/s]	ω_n [rad/s]	ζ [-]	Period T [s]
Short Period	$-4.2701, -7.5989$	5.696	1.042	0.4
Phugoid	$-0.0030 \pm 0.2091j$	0.209	0.014	30.0

Short Period

The short period motion is a rapid pitching of the aircraft about the centre of gravity, essentially an angle of attack variation. Its stiffness is set by the pitch stiffness and pitch damping derivatives (C_{M_α} , C_{M_q} , $C_{M_{\dot{\alpha}}}$). A strongly negative C_{M_α} is produced and hence resulting in a well damped, comparatively high frequency short period. Figure 6.4a shows the plotted simulated dynamic response of the aircraft to an elevator input in three states (angle of attack α , pitch angle θ and pitch rate q).

The short period response confirms the rapid, heavily damped character predicted by the eigenvalues in Table 6.4. The pitch rate reacts first, rising to about $+2.3$ [deg/s] and then reversing to roughly -3.3 [deg/s] as the nose pitches back down. The angle of attack and pitch angle follow with a single overshoot of about $+0.8$ [deg] and $+1.0$ [deg] respectively. All three states return to trim within roughly two seconds of the input. There are no sustained oscillations as expected, considering the overdamped real roots ($\zeta > 1$). This behaviour is caused by the strong pitch stiffness and pitch damping derivatives; C_{M_α} and C_{M_q} .

Phugoid

The phugoid is the slow exchange of airspeed and altitude at a constant angle of attack. As is typical of conventional configurations, it is lightly damped and of long period, since its damping derives largely from drag while its frequency is governed by flight speed. Its long period places it well within the bandwidth of the flight controller to suppress, so even a lightly damped phugoid presents no difficulty for the augmented aircraft. Figure 6.4b plots the dynamic response of this motion.

The phugoid response shows the opposite extreme to short period. After the initial transient, the angle of attack remains close to its trim value, oscillating by only about ± 0.2 [deg], whereas the pitch angle swings by roughly ± 2.5 [deg] with a period of about 30 [s], in agreement with Table 6.4. This nearly constant α combined with a large, slow θ is the signature of the phugoid motion. The amplitude decreases only marginally over the 240 [s] window, which reflects the very low damping ratio of $\zeta = 0.014$. The envelope nonetheless decreases rather than grows, so although the mode fails the Level-1 damping floor mentioned in Subsection 6.2.7 it remains genuinely stable and its long period places it well within the bandwidth available to the augmentation system.

6.2.6. Dynamic Lateral Stability

The asymmetric system produces the Dutch roll, the aperiodic roll and the spiral mode. Their characteristics are given in Table 6.5.

Table 6.5: *Lateral eigenmodes at cruise.*

Mode	Eigenvalue λ [1/s]	ω_n [rad/s]	ζ [-]	T or τ [s]
Dutch Roll	$-0.4081 \pm 2.4869j$	2.520	0.162	$T = 2.5$
Aperiodic Roll	-2.85234	–	–	$\tau = 0.35$
Spiral	+0.01266	–	–	$\tau = 78.98$

Dutch Roll

The Dutch roll is the coupled yaw and roll oscillation. Its frequency scales with the directional stiffness that is largely determined by the weathervane stability, $C_{n\beta}$. Large twin vertical tails (Subsection 6.2.3) contribute strongly to this quantity. Furthermore, its damping comes principally from the yaw damping derivative, C_{nr} , which is also governed by the strong stabilizing effect of the vertical tail. The simulated dynamic response of it is visualised in Figure 6.4c with the four states (sideslip angle β , roll angle ϕ , roll rate p and yaw rate r).

The Dutch roll response is the expected coupled roll and yaw oscillation. All four states oscillate together with a period of about 2.5 [s]. The sideslip reaches about -2° while the roll and yaw rates peak near $\pm 4^\circ/\text{s}$ and $\pm 3^\circ/\text{s}$ with each successive cycle clearly smaller. The oscillation has effectively decayed within about 12 [s], consistent with $\zeta = 0.162$ and the strong yaw damping C_{nr} contributed by the twin tail. Two further features are worth noting that are important not to disregard. Firstly, the roll angle settles not to zero, but rather to a small steady bank that then drifts slowly upward. Secondly, the yaw rate settles to a small positive offset. These does not mean necessarily incorrect since they represent the slow, slightly unstable spiral mode appearing once the fast Dutch-roll oscillation has decayed.

Aperiodic Roll

The aperiodic roll is the heavily damped, non-oscillatory decay of roll rate following an aileron input governed by the roll damping derivative C_{lp} . The Prandtl-wing obtains roll damping from both lifting surfaces together with the winglets, giving a large negative C_{lp} and therefore a fast mode with a time constant of 0.35 [s]. The resulting response can be seen in Figure 6.4d.

The response is dominated by the roll rate, which jumps sharply to about $+3^\circ/\text{s}$ immediately after the aileron pulse and then decays almost monotonically to zero within roughly ten seconds. This rapid, non oscillatory decay is governed by the large C_{lp} . The roll angle is the integral of this roll rate, so it climbs quickly and then levels off at a constant bank of about 2° once the roll rate has vanished. The minor oscillations in β and r are residual Dutch roll content excited by the same input, an order of magnitude smaller than the roll motion itself.

Spiral

The spiral is the slow, non-oscillatory divergence or convergence in bank and heading, set by the balance $C_{l\beta}C_{nr} - C_{lr}C_{n\beta}$. The computed mode is unstable, with a positive eigenvalue of $\lambda = +0.0127$. A mildly divergent spiral is the norm rather than the exception as most conventional aircraft have spiral instability. That is because of the directional stiffness $C_{n\beta}$, that drives the divergence. It is typically allowed to slightly outweigh the dihedral effect $C_{l\beta}$ that would restore the bank. Such a slow divergence is harmless in practice, because it can be corrected easily by giving inputs from flight controller that does not affect the handling quality. Figure 6.4e shows the simulated response of the spiral motion.

The spiral response is the only mode that does not return back to trim unlike the other modes. After the initial fast transient dies away, the roll angle, yaw rate and sideslip all grow slowly and monotonically. The roll angle reaching about 17° and the yaw rate about $3^\circ/\text{s}$ by the end of the 200 [s] window. This steady divergence is the unstable spiral, corresponding to the positive eigenvalue in Table 6.5. The growth is exponential but extremely slow, consistent with the long time to double of $T_2 = 54.7$ [s].

6.2.7. Handling Quality Assessment

The flying qualities of the aircraft are evaluated against the criteria of MIL-F-8785C, assuming a Class-II aircraft which is medium weight and has low-to-medium manoeuvrability in a Category B flight phase of which cruise is representative. Although MIL-F-8785C is a military specification, it provides quantitative limits on the dynamic behaviour of an aircraft and it is most widely used reference for this type of assessment. For each dynamic mode it bounds characteristic parameters presented in Table 6.6. Applying these military criteria to an unconventional civil Prandtl-plane configuration is justified by previous work. The same specification was adopted to assess the flying qualities of the IDINTOS PrandtlPlane, a closed-wing aircraft closely comparable to the present layout [47].

Table 6.6: *Handling-quality assessment of the cruise eigenmodes against the Level-1 boundaries of MIL-F-8785C, Category B (cruise).*

Mode	Level-1 criterion	Computed	Status
Short period	$0.30 \leq \zeta \leq 2.0$	$\zeta = 1.042$	Pass
Phugoid	$\zeta \geq 0.04$	$\zeta = 0.014$	Fail
Dutch roll	$\zeta \geq 0.08, \omega_n \geq 0.4$ [rad/s], $\zeta\omega_n \geq 0.15$	$\zeta = 0.162, \omega_n = 2.52$	Pass
Aperiodic Roll	$\tau \leq 1.4$ [s]	$\tau = 0.35$ [s]	Pass
Spiral	Stable or $T_2 \geq 20$ [s]	Unstable $T_2 = 54.7$ [s]	Pass

Overall, the unaugmented airframe is dynamically acceptable at this level. The spiral is mildly unstable, which is the norm rather than the exception for conventional aircraft and remains comfortably within the Level-1 time-to-double limit. Despite the failed criterion on phugoid motion, it is still stated and determined to be stable in Subsection 6.2.5. The Level-1 criterion constrains the dampening of phugoid motion over the regular civil aircraft but since the damping ratio is positive regardless, the stability of the motion is acceptable and safe.

6.3. Designing for Longitudinal Static Stability

Subsection 6.1.1 and Subsection 6.1.2 outlined the theory and methods behind determining the static longitudinal stability and controllability of the aircraft. This section aims to combine these methods to describe how they can be used to determine the corresponding design targets.

However, controllability considerations have unfortunately not been implemented in the design procedure yet. This is due to time constraints. This section thus limits itself to designing for static longitudinal stability. It is expected that the controllability limit is not a leading design driver. However, it is still recommended that this analysis is included in later design iterations.

Designing for longitudinal static stability basically comes down to tweaking the available design parameters such that the CG is always located in front of the neutral point, and that an acceptable value for C_{M_α} is achieved. For a box wing aircraft, the location of the neutral point depends on the following factors:

1. Longitudinal position of aerodynamic centres;
2. Aerodynamic moment coefficients;
3. Distribution of lift between the front and aft wing;
4. Wing planform design.

All these factors can be altered during the design. Thus, the longitudinal location of the neutral point can be influenced. However, some of these design parameters are of more importance to other design considerations, and will thus not be varied to influence stability. These design parameters correspond to the wing planform design and the aerodynamic moment coefficients. The wing planform design follows from considerations regarding the aerodynamic performance of the aircraft, and the aerodynamic moment coefficient follows from the selected airfoil.

The remaining design parameters will be varied to find aircraft configurations that are stable and provide an acceptable value for C_{M_α} for the design condition. It should be noted that only the aerodynamic centre of the front wing may be varied, as the aft wing is placed on top of the vertical tail, whose position is fixed. Lastly, the distribution of total lift between the front and aft wing is primarily dictated by the distribution of total wing area between the front and the aft wings.

A similar approach can be followed to design for the stability of the aircraft in hover with OEI. The parameters that influence this design aspect are shown again below:

1. Number of propellers;
2. Longitudinal and lateral positions of the propellers;
3. Maximum propeller thrust at sea level;
4. MTOW.

The first three parameters in the list are actual design parameters. Here, the MTOW is not a target, but rather flows from the design. However, just as before, not all design parameters can be freely altered to influence the stability of the aircraft. The number of propeller engines is mainly dictated by the Power department. Furthermore, the positions of the propeller engines are heavily constrained by clearance issues. Though, the maximum propeller thrust at sea level is a parameter that can be adjusted more freely. However, altering this parameter would influence engine selection, and consequently the MTOW.

As mentioned before, there are three parameters of interest which have to be investigated, namely: the longitudinal location of the neutral point, the value for C_{M_α} along the corresponding allowable CG-envelope, and the CG-envelope for which the aircraft satisfies the OEI-stability condition. These quantities differ per aircraft configuration, and will thus have to be computed repeatedly. The process of repeatedly assessing the stability of the aircraft is described in Subsection 9.4.3. Figure 6.5 and Figure 6.6 visualise the dependencies of the various stability conditions on the design parameters.

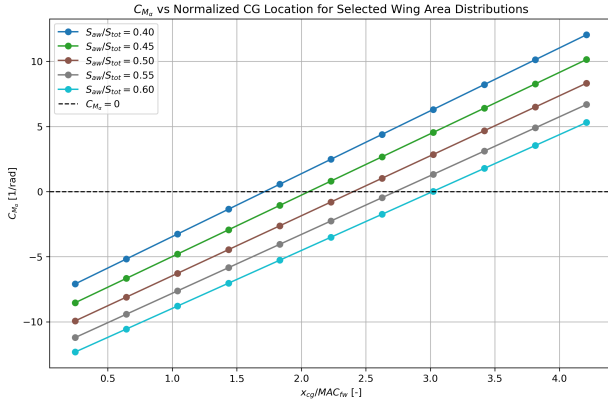


Figure 6.5: C_{M_α} vs. normalised CG-locations for various wing area distributions for a given aircraft configuration and MTOW.

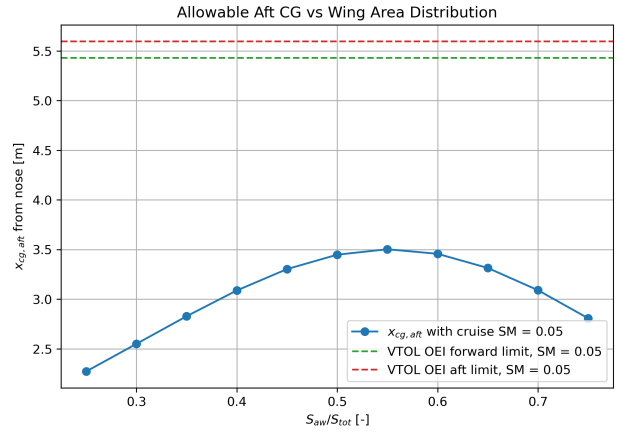


Figure 6.6: Longitudinal location of the allowable most aft CG-location and the allowable CG-envelope for a given aircraft configuration with a variable wing area distribution.

6.4. Performance Diagrams

This section explains the diagrams that assess the aircraft's wing borne performance: the payload-range diagram, the V-n manoeuvre and gust envelope, and climb-descent diagrams.

6.4.1. Payload-Range Diagram

A payload-range diagram was constructed to visualise the dependency of achievable range on the payload mass for the final aircraft configuration. As the aircraft is electric, this trade is the inverse of the conventional fuel case: a fuel aircraft offloads payload to carry more fuel, so range is bought at the cost of payload and the diagram goes down. Here, the battery mass is fixed and does not diminish in flight, so payload cannot be exchanged for stored energy. Instead, shedding payload lowers the aircraft mass, which reduces the cruise power and lets the same fixed energy carry the aircraft further. The relationship is therefore an efficiency effect rather than a capacity trade.

The cruise power follows from steady, level flight, where lift equals weight and the drag is the weight divided by the lift to drag ratio:

$$P_{cruise} = \frac{mgV_C}{(L/D)\eta_{cruise}}, \quad (6.20)$$

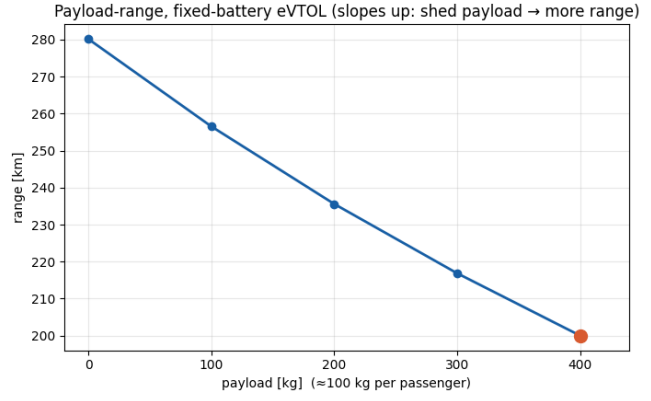
which is linear in mass. The vertical, climb and descent phases are treated as a fixed energy overhead E_{oh} that every flight incurs regardless of range, so only the cruise segment extends as payload is reduced. The cruise range at a given mass is then given by Equation 6.21 [48]

$$R = \frac{V_C}{P_{cruise}(m)} \left(\frac{E_{inst}}{1 + k_{res}} - E_{oh} - E_{aux} \right), \quad (6.21)$$

where E_{inst} is the installed pack energy, k_{res} the reserve fraction and E_{aux} the auxiliary energy. The curve is anchored at the design mission (four passengers, 200 [km]), so it is a trade study about the design point rather than an absolute-range prediction. Sweeping the payload from four passengers to zero with each passenger and baggage combination taken as 100 [kg], gives the figure below. The range increases from the 200 [km] design mission at full payload to approximately 280 [km] at zero payload, an average gain of about 20 [km] per passenger. Unlike a conventional payload-range diagram, there is no maximum fuel plateau, since the energy capacity is fixed by the installed pack rather than limited by a variable fuel load Section 5.1.

Table 6.7: Payload-range points (fixed installed energy, anchored at the design mission).

Pax	Payload [kg]	MTOW [kg]	Range [km]
4	400	1980	200
3	300	1880	217
2	200	1780	236
1	100	1680	257
0	0	1580	280

**Figure 6.7:** Payload-range diagram for the final design (MTOW 1980 kg).

6.4.2. V-n Diagram

A V-n diagram was constructed to define the wing-borne manoeuvre and gust envelope of the aircraft in cruise configuration; it represents the fixed-wing flight envelope. Hover, transition, lift unit failure cases, and rotor jamming cases are outside the scope of this diagram and are assessed separately during detailed structural design. The certification basis is EASA SC-VTOL-01 (small category; the aircraft's MTOM of 1980 [kg], established in Chapter 6, is within the 3175 [kg] limit) [49]. SC-VTOL.2215 requires that critical flight loads be established across the manoeuvre and gust envelope, but does not prescribe the limit load factors; these are adopted from the CS-23 and CS-25.333(b) conventions with which SC-VTOL is intended to be consistent. The positive limit load factor is taken as $n_{\max} = 3.5$ and the negative as $n_{\min} = -1.5$.

The stall boundary was calculated from the lift equation Chapter 4

$$L = \frac{1}{2} \rho V^2 S C_L, \quad (6.22)$$

which gives the load factor as

$$n = \frac{L}{W} = \frac{\frac{1}{2} \rho V^2 S C_{L_{\max}}}{W} = \frac{\frac{1}{2} \rho V^2 C_{L_{\max}}}{W/S} \quad (6.23)$$

The one-g stall speed is therefore

$$V_S = \sqrt{\frac{2(W/S)}{\rho C_{L_{\max}}}}. \quad (6.24)$$

As can be seen from the previous chapters Table 4.4, $C_{L_{\max}} = 1.686$ was used Chapter 10. With the final wing loading and sea-level density, the stall speed is $V_S = 27.1$ [m/s].

The manoeuvring speed was calculated as the speed at which the positive stall curve reaches the positive limit load factor,

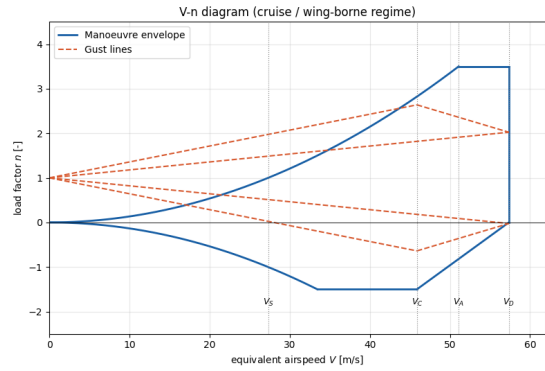
$$V_A = V_S \sqrt{n_{\max}} = 50.7 \text{ [m/s]} \quad (6.25)$$

The cruise speed is set by the design mission at 200 [km/h], i.e. a true airspeed of 55.6 [m/s] at the 3810 [m] cruise altitude. As this envelope is expressed in equivalent airspeed at sea level, the cruise speed and the selected dive speed are

$$V_C = 45.9 \text{ [m/s]}, \quad V_D = 1.25 V_C = 57.4 \text{ [m/s]}. \quad (6.26)$$

The gust envelope was included using design gust velocities of 15 [m/s] at cruise speed and 7.5 [m/s] at dive speed per certifications [49]. The calculated gust load factors remained within the selected manoeuvre load-factor limits. At cruise speed, the gust envelope reached approximately +2.64 and -0.64, while at dive speed it reached approximately +2.02 and -0.02. Therefore, the manoeuvre envelope defines the limiting load factors for the wing borne flight condition.

A Beaufort-8 gale corresponds to a gust velocity of roughly 17-21 [m/s], exceeding the 15 [m/s] certification gust used for the cruise gust line. Evaluating the gust load-factor relation at the cruise speed V_C for this range yields $n \approx +2.9$ to $+3.3$, which remains inside the $+3.5$ structural limit. The aircraft therefore withstands a Beaufort-8 gust at cruise speed without exceeding its design load factor.

**Figure 6.8:** V-n Diagram

6.5. Phase Performances

This section covers the aircraft's turn and climb/descent performance in the wing-borne cruise regime, evaluated at cruise altitude in true airspeed.

6.5.1. Turn Performance

The aircraft turning performance is assessed for the wing-borne cruise regime as a steady, level and coordinated turn. As with the V-n diagram from Subsection 6.4.2, the hover and transition phases are a separate regime: yaw there is produced by differential rotor thrust rather than by banking a lifting wing, so the coordinated-turn equations of motion do not apply here. In contrast to the V-n diagram, which is a sea level structural envelope expressed in equivalent airspeed, the turn envelope is drawn at cruise altitude in true airspeed, since turning performance degrades with altitude as the air thins.

In a level coordinated turn, the vertical component of lift balances weight while the horizontal component supplies the centripetal force,

$$L \cos \phi = W, \quad L \sin \phi = \frac{W}{g} \frac{V^2}{R}, \quad (6.27)$$

where ϕ is the bank angle and R the turn radius. Dividing these gives the load factor directly as a function of bank angle,

$$n = \frac{L}{W} = \frac{1}{\cos \phi}, \quad (6.28)$$

which is the same load factor that defines the vertical axis of the V-n diagram: a banked turn is simply a way of generating the load factors drawn there. The turn radius and turn rate follow from the kinematics,

$$R = \frac{V^2}{g \sqrt{n^2 - 1}}, \quad \omega = \frac{g \sqrt{n^2 - 1}}{V}. \quad (6.29)$$

Two limits bound the achievable load factor. The aerodynamic limit is set by the maximum lift coefficient

$$n_{\text{lift}}(V) = \frac{\frac{1}{2} \rho V^2 C_{L_{\max}} S}{W}, \quad (6.30)$$

and the structural limit is the positive limit load factor $n_{\max} = 3.5$ adopted in the V-n analysis from Subsection 6.4.2. The two are equal at the corner speed,

$$V_A = V_S \sqrt{n_{\max}}, \quad (6.31)$$

which is the lowest speed at which the full structural load factor can be reached and therefore gives the tightest, fastest turn the structure permits. Below V_A the turn is lift-limited; above it, structurally limited. The maximum lift coefficient is taken as $C_{L_{\max}} = 1.686$ ¹. The induced drag uses a span (Oswald) efficiency factor $e = 1.34$, taken from Section 4.4. This factor measures how close a lifting system comes to the minimum induced drag for its span: for a conventional planar wing it cannot exceed unity, with $e = 1$ corresponding to the ideal elliptical lift distribution. The Prandtl-plane lifting system is non-planar – its two vertically separated wings form a closed box that sheds less induced drag than any single planar wing of the same span – so it surpasses the planar limit, giving $e > 1$.

Evaluated at cruise altitude, the one-g stall speed is $V_S = 32.8$ [m/s] and the corner speed is $V_A = 61.4$ [m/s] which sits just above the cruise speed of 55.6 [m/s]. The practical consequence is that at cruise speed the aircraft is lift limited rather than structurally limited: the maximum attainable load factor is $n = 2.87$, comfortably below the structural limit, corresponding with the bank angle of 70° and a radius of 117 [m]. The tightest structurally permitted turn is at V_A , where the aircraft can pull the full $n_{\max} = 3.5$ at a bank angle of 73.4° , a radius of 114 [m], and a turn rate of $30.7^\circ/\text{s}$. These margins are comfortable for an air taxi mission, whose operational turns are gentle (bank angles well below 30°); the envelope confirms the aircraft is not turn limited in any normal manoeuvre.

6.5.2. Climb-Descent Performance

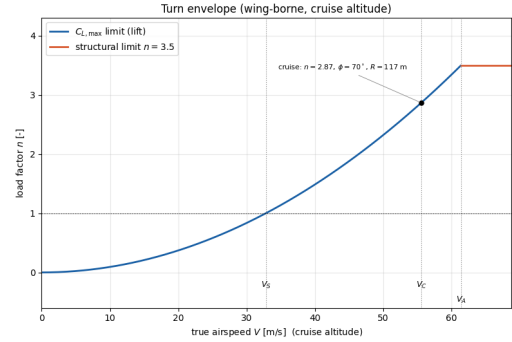


Figure 6.9: Turn Envelope

¹XFLR5, <https://www.flow5.tech/xflr5/xflr5.html>, Accessed on 16th of June 2026

The climb and descent performance is assessed for the wing-borne phase, i.e. after transition when the aircraft is in its cruise configuration. The analysis follows the standard framework [50]: the rate of climb is set by the excess of available over required power.

$$\text{RoC} = V \sin \gamma = \frac{P_a - P_r}{W}, \quad (6.32)$$

and the climb gradient by the excess of thrust over drag, $\sin \gamma = (T - D)/W$. The required power is the full drag power in level flight,

$$P_r = DV = \frac{1}{2} \rho V^3 S (C_{D0} + kC_L^2), \quad C_L = \frac{W}{\frac{1}{2} \rho V^2 S}, \quad (6.33)$$

evaluated at cruise density. The drag polar uses the preliminary $C_{D0} = 0.0205$ and the box wing span efficiency of $e = 1.34$, consistent with the turning analysis Subsection 6.5.1. The induced term is retained: it dominates P_r at low speed and is what gives the required power curve its minimum, without which the best climb speed cannot be located. The required power curve reaches a minimum of 45 [kW] at around 37 [m/s]. The speed for the best rate of climb, V_y , is the speed at which the gap $P_a - P_r$ is widest; because the available-power curve is approximately flat over this range, V_y coincides with the minimum-power speed at $V_y \approx 37$ [m/s].

The design reference mission climbs a shallow, near constant flight path angle. Entering the wing borne climb at around 20 [m/s] and accelerating to the cruise velocity of 55.6 [m/s] the flight path angle to the 3810 [m] altitude is $\gamma = 3.1^\circ$, giving an approximate rate of climb of 1.1 [m/s] at entry to 3 [m/s] at cruise speed as the aircraft accelerates along the fixed gradient. This shallow climb sits well inside the capability envelope Subsection 6.4.2 and is power feasible. The corresponding climb power is obtained from

$$P_{\text{climb}} = W \sqrt{V_{\text{cruise}}^2 + v_y^2} \left(\frac{D}{L} \cos \gamma + \sin \gamma \right), \quad (6.34)$$

in line with the mission-profile model of Chapter 5. A clearance check also confirms the climb clears the highest obstacle peak of 3048 [m] for the Nice-Courchevel route.

The descent is a mirror case: a shallow wing-borne glide on the same gradient family down to the start of the transition, decelerating towards the 5 [m/s] hand over to the rotor borne landing. On the glideslope the required power is below the cruise power so the descent is power limited only by the need to manage energy rather than generate power, all before the transition and vertical landing process.

6.6. Discussion

The aim of this chapter was to define and establish the methodology that is used to assess the stability of the Prandtl Plane in both configurations. Subsection 6.1.1 outlined a workable methodology to evaluate the aircraft's static longitudinal stability. This method makes use of established semi-empirical methods that can be easily incorporated into a preliminary design. However, this also indicates the area of improvement: to better model and predict the stability behaviour of the aircraft in winged flight, the empirically derived variable values need to be replaced by analytically derived values. Additionally, the controllability limit of the aircraft needs to be determined finalise the determination of the feasible CG-envelope of the aircraft in winged flight.

Furthermore, Subsection 6.1.2 outlined the method to evaluate the stability of the aircraft in its VTOL-configuration. Again, the described method can be confidently used in the preliminary design. However, the condition regarding the moment equilibrium around the z-axis needs to be incorporated to improve model fidelity.

The dynamic stability analysis outlined in Section 6.2 contains calculating all of the aircraft stability derivatives using semi-empirical DATCOM methods by USAF. At this stage, this level of fidelity is found to be sufficient. However, in the future these derivatives and coefficients can be determined by conducting a CFD-analysis and even with a flight test.

Lastly, Section 6.4 and Section 6.5 describe the performance of the aircraft at the current stage of the design. The methods outlined in these sections need to be executed for each iteration of the aircraft design.

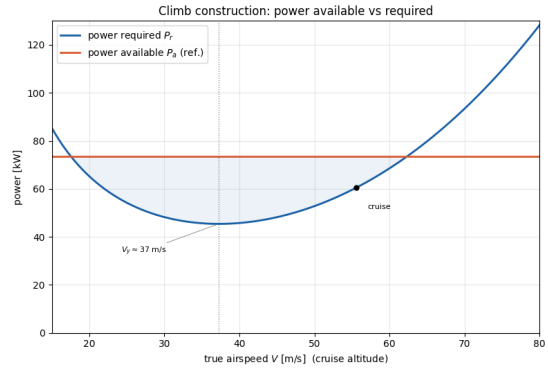


Figure 6.10: Climb performance of the aircraft in winged flight.

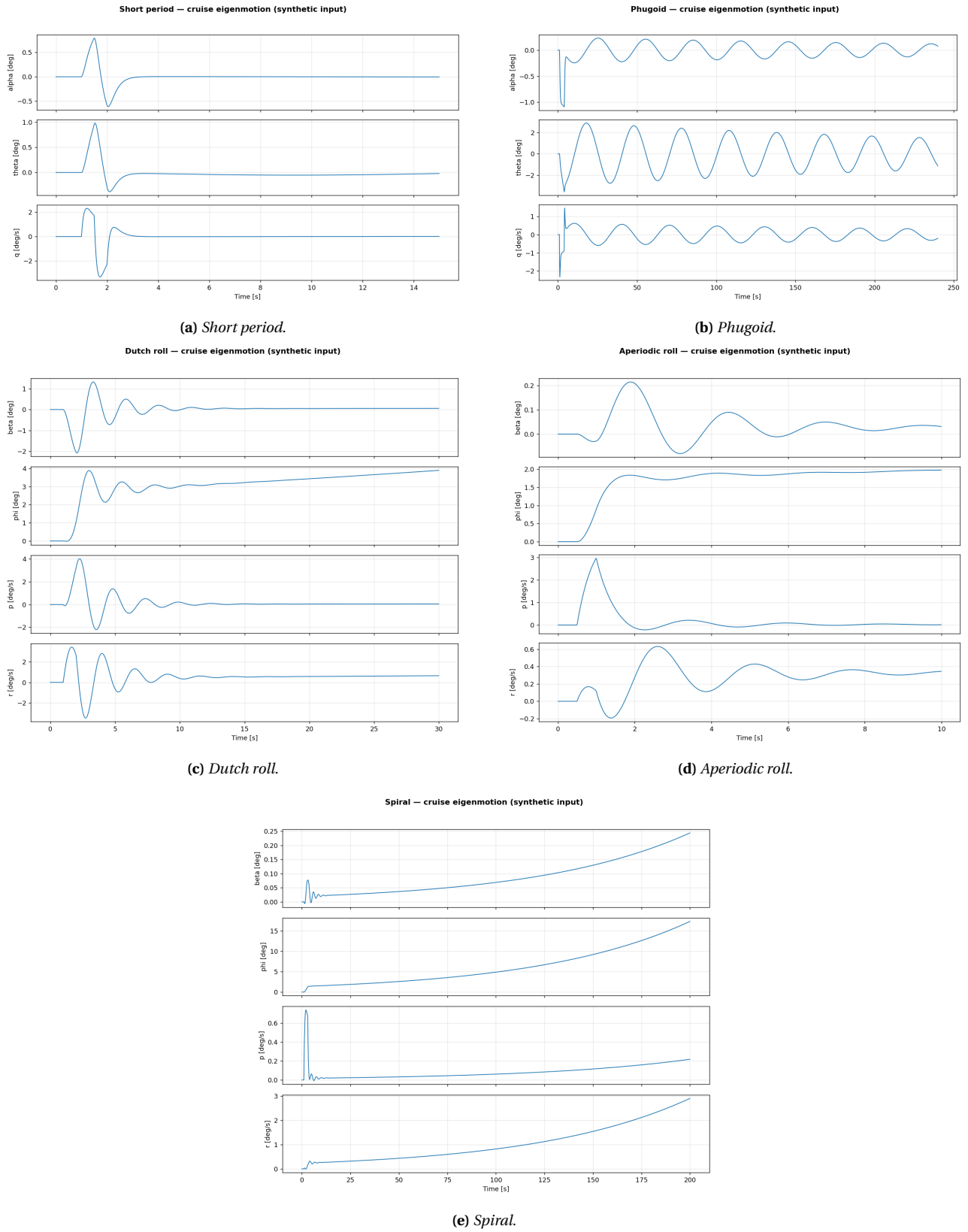


Figure 6.4: Simulated responses for eigenmotions in cruise

7 Autonomous Control

This chapter presents the proposed autonomous-control architecture for Vold-e. Section 7.1 provides an overview of the selected onboard sensors and communication components, including their purpose, locations, and redundancy. Section 7.2 then describes the software architecture and control functions that enable autonomous operation of the aircraft. Finally, Section 7.3 discusses the limitations, risks, and areas for further development of the proposed autonomy concept.

7.1. Components

The aircraft consists of multiple sensors. Some of them are present in every type of modern aircraft and the rest are crucial for the desired autonomous control.

Table 7.1: Overview of the avionics, sensor and communication components used in the aircraft.

Component	Function	Location	Redundancy
<i>Navigation Sensors</i>			
IMU Sensoror STIM300 ¹	High rate IMU for the inner loop rate controller.	Avionics bay, near the centre of gravity.	Multiple voting IMUs, EKF cross checks against GNSS and air data.
GNSS u-blox ZED-F9P ²	Position and ground velocity by triangulation, fused with IMU and barometer.	Two upper fuselage antennas ≥ 1 m apart, receivers in the avionics bay.	Dual receivers, falls back to IMU and VINS on loss or spoofing.
Air Data Set Aerosonic ³	Pitot and static tube give air-speed and altitude, AoA vane gives angle of attack.	Pitot and static ports on nose and fuselage, AoA vane on the nose side.	Heated redundant ports and dual AoA sensors, fallback to GNSS ground speed.
<i>Perceptron Sensors</i>			
LIDAR Phoenix LiDAR Systems ⁴	Dense range (1550 nautical miles) to obstacles and ground distance on landing.	Forward unit on the nose, downward unit on the lower fuselage.	Single point failure per phase, reverts to radar in cruise or barometer in landing.
Radar Echodyne EchoFlight ⁵	All weather, longer range detection of non cooperative obstacles.	Forward facing nose, co aligned with the forward LiDAR.	Redundant pair with the forward LiDAR, only one effective in poor weather.
External Cameras Kappa Electronics ⁶	Situational awareness, vertiport identification and VINS input.	Four units around the fuselage: left, right, rear and downward.	Not catastrophic, loss only limits GNSS denied vertiport operation.
Interior Camera	Cabin or payload bay monitoring and remote oversight.	Cabin ceiling, wide angle view.	Not flight critical, single unit acceptable.
<i>Flight Computer</i>			
Flight Computer Collins FMC-4212 ⁷	Hosts the control laws, EKF and mission management, commands the actuators.	Avionics bay, three units physically separated.	Triple redundant active voting, faulty channel voted out.
<i>Communications</i>			
Communications	Command and control, telemetry and traffic awareness links. Used for mobile network	Terminals in the avionics bay, antennas on the fuselage.	Multiple independent links, not flight critical.

¹Safran Group, <https://www.safran-group.com/sites/default/files/2022-05/safran-sensing-technologies-product-brief-stim300.pdf>, Accessed on 15th of June 2026

²Ublox, https://content.u-blox.com/sites/default/files/ZED-F9P_ProductSummary_UBX-17005151.pdf, Accessed on 15th of June 2026

³Aerosonic, <https://www.aerosonic.com/products-1>, Accessed on 15th of June 2026

⁴Phoenix LiDAR Systems, <https://www.phoenixlidar.com/wp-content/uploads/2019/11/PLS-Ranger-LR-Spec-Sheet.pdf>, Accessed on 15th of June 2026

⁵Echoflight, https://cdn.prod.website-files.com/69148cb0d4c3689c365788f0/69d6b1b31e686a271dd38366_TS-EchoFlight-Echodyne_6AP.pdf, Accessed on 15th of June 2026

⁶Kappa Electronics, <https://www.kappa-optonics.com/en/aviation/uav-uas/>, Accessed on 15th of June 2026

⁷Collins Aerospace, https://prd-sci02-cdn.rtx.com/collinsaerospace/-/media/ca/product-assets/files/public/products/product-brochures/computing/processing-mission-computers/fmc-4212_ds.pdf?rev=152c72082b994dffa6f1006c0f829bf0&rid=56c45a5d1c97408aa54725b4ecea641c&hash=28C2C09E0B0B716499244E6E70D836BE, Accessed on 15th of June 2026

⁸SkyTrac, <https://www.skytrac.ca/products/broadband-uav-satcom/ims350/#capabilities>, Accessed on 15th of June 2026

Component	Function	Location	Redundancy
SATCOM SkyTrac IMS-350 ⁸	Beyond line of sight satellite link via Iridium Certus LEO.	Terminal in the avionics bay, antenna on the upper fuselage.	Single terminal, reverts to cellular and RF.
Transponder	Mode S and ADS-B for cooperative detect and avoid.	Avionics bay, diversity antennas.	Not flight critical, reverts to radar and LiDAR.

7.2. Software

This section describes the software architecture that enables autonomous operation of Vold-e. It explains how sensor data are processed into flight-control commands and how the system maintains safe operation through redundancy, remote supervision, and emergency recovery functions.

7.2.1. Autonomy Architecture

The aircraft has ability to operate autonomously with a remote pilot that can monitor and intervene at any time. This allows human oversight onto the autonomous system which has been ensured by the components selected in Section 7.1, such as having transponder to communicate with ATC. The autonomy of the vehicle is structured as a closed "sense-estimate-decide-act" loop, shown in Figure 7.1. Each stage is described below and the functions map directly onto the components selected in Section 7.1.

7.2.2. The Sense–Estimate–Decide–Act Loop

Sense: The aircraft state and its surroundings are measured by dedicated sensor hardware. The navigation sensors (IMU, GNSS and air probes) measure the aircraft's attitude, position and airspeed. Perception sensors (LiDAR, radar and cameras) measure the surrounding environment. Each is a physical unit producing raw data.

Estimate: The raw measurements are fused into a useable picture of the aircraft and its environment. A state estimator, i.e. an extended Kalman filter, combines the navigation sensors into a single continuous state of position, velocity and attitude; the redundant inertial and GNSS units allow this estimate to be maintained if a single sensor stops working. In parallel, the perception sensors are fused into an obstacle map. Because LiDAR and the cameras may degrade in weather conditions like fog but radar doesn't, the fused map performs well in almost every weather conditions.

Decide: Path determination is handled at two levels. At the strategic level, the route is an input to the aircraft that consists of waypoints and an approach sequence from the flight plan. The mission manager holds this plan, tracks progress through the waypoints and the active flight phase, and handles contingencies such as holding or diverting. At the tactical level, actual trajectory across waypoints are generated by taking the current state, obstacle map and the next waypoint. This trajectory is continuously updated to avoid obstacles and adapt to changeable conditions while the strategic waypoints remain fixed. At this stage, the planning algorithm is defined by its inputs and the clearance constraints it must satisfy.

Act: The flight control inner loop converts the trajectory into commands for the actuators. The loop closes after resulting motion is measured again by sensors.

7.2.3. Flight Computer and Software Hosting

The sensor and actuators are physical components unlike software based functions. The FMC-4000 family is a multicore, DAL-A-certified (DO-178C / DO-254) platform offered for unmanned aircraft systems that provides the signal, image and video processing needed to fuse the perception data¹. The loop handles the main in flight events through the same architecture:

- **Commanded turns and manoeuvres** are produced by the guidance function as trajectory commands by inner loop control laws moving the actuators.
- **Obstacles** appear in the fused obstacle map and cause the guidance function to replan the trajectory around them, with radar providing the detection that remains available in poor weather.
- **Disturbances** such as gusts change the measured aircraft state; the estimator updates the state and the inner-loop control laws correct it through the actuators.

7.2.4. Redundancy and Remote Supervision

The flight computer hosts the entire estimate-decide-act chain so its loss would remove the complete autonomy function rather than degrade a single input. For this reason it is triple-redundant with hot triplex, all three units active and voting. The remote supervisor provides an additional layer of oversight over the SATCOM and cellular links: the operator monitors the aircraft's state and intentions and can command high-level actions such as a reroute, hold or divert. The supervisor can also intervene more directly by commanding the aircraft at the trajectory and attitude level, in effect "hand flying" the outer loop. In all cases the high rate inner stabilization loop remains onboard the flight computer; the link carries only the slower outer loop commands, never the fast stabilization loop, because the satellite-link latency is unsuitable for closing the inner control loop remotely.

¹ Collins Aerospace-RTX, <https://www.rtx.com/news/news-center/2026/02/20/rtxs-collins-aerospace-autonomy-solution-sidkick-flies-ga-asis-yfq-42a-cca-p>, Accessed on 15th of June 2026

7.2.5. Emergency Recovery and Energy Reserve

When the propulsion or energy system has failed entirely, a final automated layer is activated. If the autonomy loop decides that there are no recovery options possible, a ballistic recovery parachute is deployed. This last resort can still be able to fire when all power is lost because, its power source is independent from the main battery. Further details on the system can be found in Section 5.4.

This chain is also backed by a small energy reserve to ensure the deployment occurs within the parachute's structural envelope. This reserve is a bus battery sized to power only flight computer, inertial and air data sensors and the recovery system after the loss of the main bus (Chapter 5). Its duty is only to keep the chain alive until the parachute is deployed at the right moment.

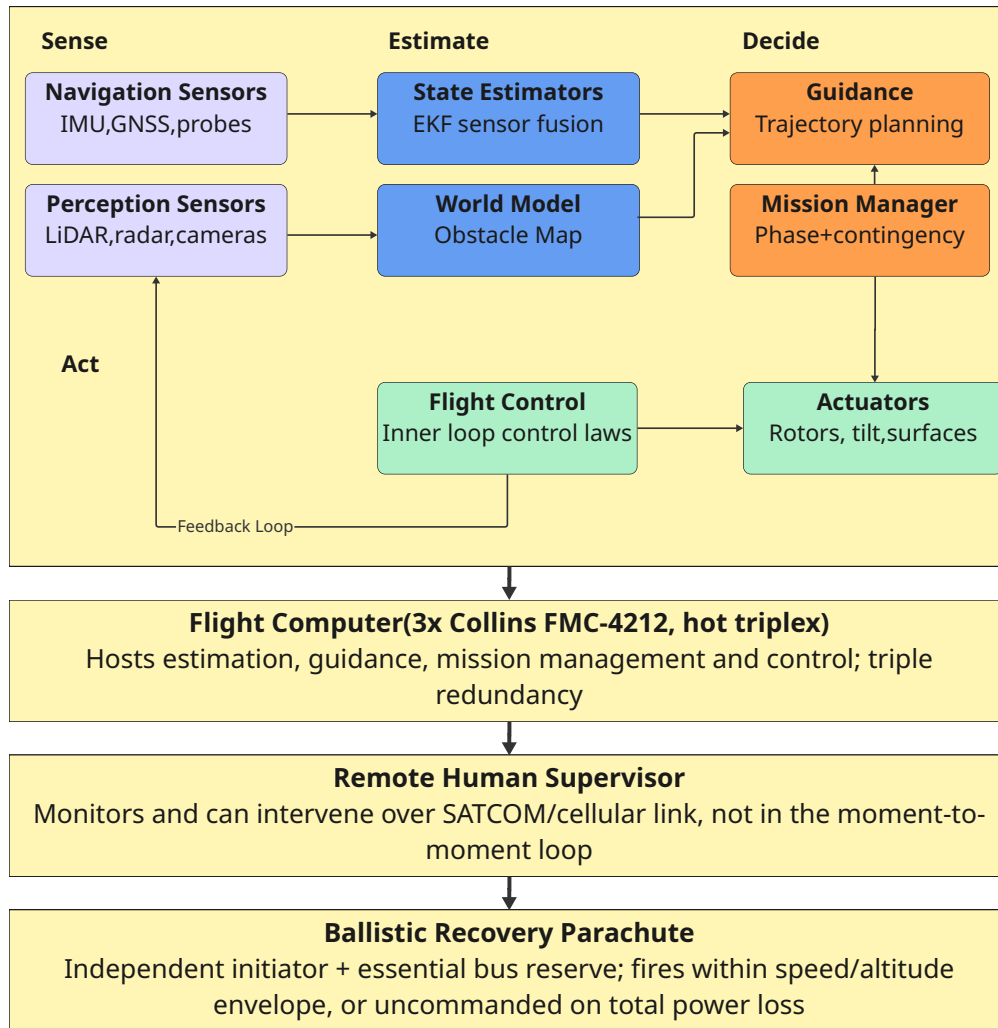


Figure 7.1: Autonomous operation architecture

7.3. Discussion

The autonomy architecture presented in this section is defined at the functional level. The sense-estimate-decide-act loop fixes the flow of data and the redundancy with each function being mapped to chosen components, but the underlying algorithms such as the EKF tuning or the trajectory planner are specified by their interfaces rather than being designed in detail. These are the main items for the next phase with the integration of sensors, where placement effects such as rotor slipstream over the air data probes, electromagnetic interference and field-of-view blockage still need to be assessed, if an actual physical model is built.

There are also two assumptions which carry risk. The first of which, the all weather sensing function of the LiDAR+Radar depends on radar working effectively when the LiDAR and cameras degrade. Second, the concept depends on the certification of full autonomy with remote supervision under the evolving EASA SC-VTOL enhanced category, which is the dominant external risk and tracked in Section 12.1. The redundant computing, layered supervision and ballistic parachute are sized to meet the reliability target, but the quantitative reliability in Subsection 12.2.1 remains preliminary until detailed testing is performed.

8 Structures

This chapter provides an overview of the methods and models used to determine the mass of the structure of the aircraft. The main components sized include the wings, fuselage, hinges, tails and landing gear. A variety of approaches were used for the different components, but they are all physics-based. This chapter describes the physical basis of the MTOW convergence loop.

First, Section 8.1 estimates the aircraft mass moments of inertia from the converged component mass breakdown. Section 8.2 evaluates the loads and hinge reactions acting on the folding wing structure throughout the transformation. Next, Section 8.3 sizes the wing, winglet, and folding-wing primary structures through an iterative physics-based approach. Section 8.4 presents the design, sizing, material selection, and mass budget of the folding hinges and actuators. Section 8.5 estimates the structural mass of the canted tail panels from their governing bending loads. Section 8.6 compares and sizes alternative landing-gear architectures using an energy-based optimisation approach. Section 8.7 evaluates the structural and systems mass associated with cabin pressurisation. Section 8.8 estimates the fuselage skin thickness and mass from the critical bending load cases. Finally, Section 8.9 summarises the principal limitations of the structural analyses and identifies areas for further development.

8.1. Mass Moments of Inertia Calculation

The mass moments of inertia (MMOI) of the aircraft are estimated from the converged Class II mass breakdown. Each major component is represented by a simple geometric shape that is assigned its component mass, placed at its centre of gravity location and combined about the aircraft centre of gravity by using the parallel axis theorem.

8.1.1. Geometric Simplifications

The fuselage is modelled as a thin walled cylindrical shell, since the bulk of its structural mass resides in the outer skin. The actual fuselage is not a uniform tube but tapers towards the nose and tail. In order to capture this, the cylinder length is scaled by an effective length factor that represents the constant diameter mid-section that carries most of the structure, rather than the full scale length. It has a value of 2/3. The rest of the components are approximated as shown below. Each component contributes its standard closed form inertia value evaluated about its own centre of gravity.

- The front and rear wings as rectangular prisms with uniform spanwise mass distribution.
- The winglets are modelled as parallelograms.
- The twin vertical tail panels as flat plates rotated about the longitudinal axis by an angle.
- The rotor pods (motors, hubs and tilt hinges) as point masses at the six rotor stations.
- The propellers as thin discs.
- The battery and payload as solid rectangular boxes that have uniform density.
- The landing gear as slender rods along the skid rails.
- The miscellaneous systems (wiring, ECS, etc.) as a solid cylinder filling the fuselage, scaled by the same fuselage length factor.

8.1.2. General Formulation

The total inertia matrix about the aircraft centre of gravity is obtained by summing the local matrix of every component together with its transport term using the parallel axis theorem.

$$\mathbf{I} = \sum_i [\mathbf{I}_{i,\text{local}} + m_i ((\mathbf{d}_i \cdot \mathbf{d}_i) \mathbf{E}_3 - \mathbf{d}_i \otimes \mathbf{d}_i)], \quad \mathbf{d}_i = \mathbf{r}_i - \mathbf{r}_{\text{cg}}, \quad (8.1)$$

where m_i is the component mass, $\mathbf{d}_i$ its offset from the aircraft centre of gravity, $\mathbf{E}_3$ the 3×3 identity matrix and $\otimes$ the outer product. The diagonal entries of $\mathbf{I}$ give the moments of inertia about the roll, pitch and yaw axes (I_{xx} , I_{yy} , I_{zz}), while the non diagonal entries give the products of inertia, reported in the flight-dynamics convention $I_{ab} = \sum_i m_i d_{a,i} d_{b,i}$. Because the mass layout is laterally symmetric, the products I_{xy} and I_{yz} vanish, leaving I_{xz} as the only non zero product of inertia.

The Prandtl wing is a novel but complex configuration which requires an in depth structural analysis due to the presence of an extra wing and a winglet. Furthermore, the folding mechanism fundamentally changes the way loads are applied on the wing, which requires an in depth load analysis of the forces and moments at each configuration.

8.2. Load Analysis on Wing and Hinge Structures

The unique folding mechanism of the aircraft creates a significant degree of complexity when designing the wing and hinges, as the forces and moments on the wing structures vary greatly depending on the angle of the wing with respect to the main body. Furthermore, their orientations also change drastically as the wing folds and unfolds.

8.2.1. Definition of the Problem and Coordinate Systems

The aim of this model is to provide an overview of the forces and moments on a given wing at any given hinge angle θ , as well as the corresponding reaction forces and moments on the hinge. Due to the complexity and 3D nature of the problem, a set of coordinate systems was necessary as shown in Figure 8.1 to study how the forces change with respect to each individual component. For this study 3 coordinate systems were defined. All of their origins are with respect to the wing's quarter-chord length.

1. The body axis b is defined as a right hand coordinate system, with the x-axis pointing towards the aircraft nose, the y-axis pointing downwards and the z-axis pointing towards the winglet.
2. The hinge axis h is defined as the transformation $\mathbb{T}_z(-\phi_h)\big|_b, \mathbb{T}_y(\theta_h)\big|_b$ from the body axis, where ϕ_h and θ_h are the two angles defining the hinge orientation. In order to achieve the desired wing rotation the values $\theta_h = 45^\circ$, $\phi_h \approx 35.3^\circ$. The x axis represents the hinge's axis with respect to the body.
3. The wing axis w is defined as the transformation $\mathbb{T}_{h_x}(\theta)\big|_b$ where θ $^\circ$ is the angle by which the wing rotates around the hinge. At $\theta = 0^\circ$, the wing is fully unfolded and at $\theta = 120^\circ$ the wing is fully folded. It is important to note that theta being larger than 90° is because the wing must rotate a larger distance than if it were rotating relative to a single axis. Figure 8.1 shows this rotation in more detail.

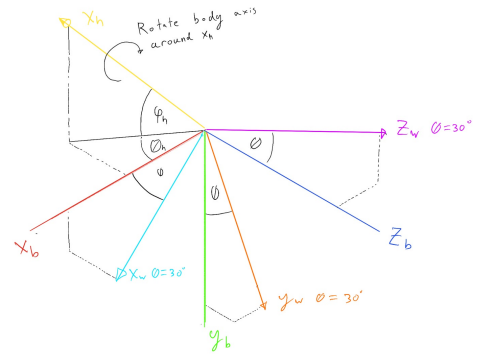


Figure 8.1: Rotation of body axis to wing axis around hinge x-axis ($\theta \approx 30^\circ$)

8.2.2. Important Assumptions & Effects of the Winglet Hinge

This section displays the assumptions taken for the loads and load analysis. Table 8.1 gives a general overview and the important and impactful assumptions. The code given to the assumptions is as follows: ASSU-WH-XX. Where ASSU stands for assumption, WH, for wing and hinge model, and XX the corresponding number of the assumption.

Table 8.1: Assumptions used for the wing loading and hinge model

Assumption	Description
ASSU-WH-01	The wing's elliptical lift distribution is assumed to have a similar shape for all angles of θ , and is scaled by $\left(\frac{V_{cruise}}{V}\right)^2$.
ASSU-WH-02	The centre of gravity and aerodynamic centre of the airfoil are assumed to be located at the quarter-chord point.
ASSU-WH-03	The mass of the propellers is modelled as a point mass acting at their respective centre-of-gravity locations.
ASSU-WH-04	The propeller thrust is modelled as a point force acting at the position of the respective propeller, and is assumed to have no effect on the lift distribution.
ASSU-WH-05	The wing lift and weight distributions are modelled as two-dimensional distributed loads acting along the quarter-chord line.
ASSU-WH-06	The winglet hinge is assumed to carry only reaction forces and no reaction moments, making the structure statically determinate.
ASSU-WH-07	The wing hinge is assumed to be a clamped support from the model perspective. The recorded moment around the hinge rotation axis is used to size the actuator.
ASSU-WH-08	The wing lift is assumed to be negligible for hinge angles above 30 degrees, as both the aircraft speed and the angle of the wing relative to that speed do not meet the minimum stall conditions.

In order to understand the effects of the winglet hinge on the structural rigidity of the wing, it must be understood that, in order for the entire structure to be foldable, the wing and winglet hinge origins must lie on a line perpendicular to the body x axis. More importantly, the hinges must both be facing the same direction.

Furthermore, the hinge at the winglet restricts translational moment along its rotation axis, and restricts any rotational movement that isn't around its rotational axis. This means that it is a structure with 3 degrees of freedom, as any forces along the winglet hinge's y-z axis and any moment around its x-axis will not be met with a reaction from

the hinge. The wing hinge has 2 fewer degrees of freedom, as it is attached to the fuselage via a structure that restricts translation along the y - z axis.

This means that when analysing forces and reactions along a plane perpendicular to both hinge rotational axes, only the wing hinge experiences reaction forces, while neither hinge is able to carry the moments in this orientation. However, when analysing the forces parallel to the hinge axes, both hinges produce reaction forces and moments. This problem is equivalent to analysing the forces on a beam clamped on both sides, which requires details about the beam geometry to solve. As this problem lies beyond the current scope of this model, the conservative assumption will be made that the hinge at the wing carries all the forces and moments in this configuration. The hinges at the winglets still need to carry the winglet weight, as well as loads imposed by the winglets such as lift. These forces will be assumed to translate directly towards the wing and will be analysed as such.

8.2.3. Discretization & Definition of Forces and Moments

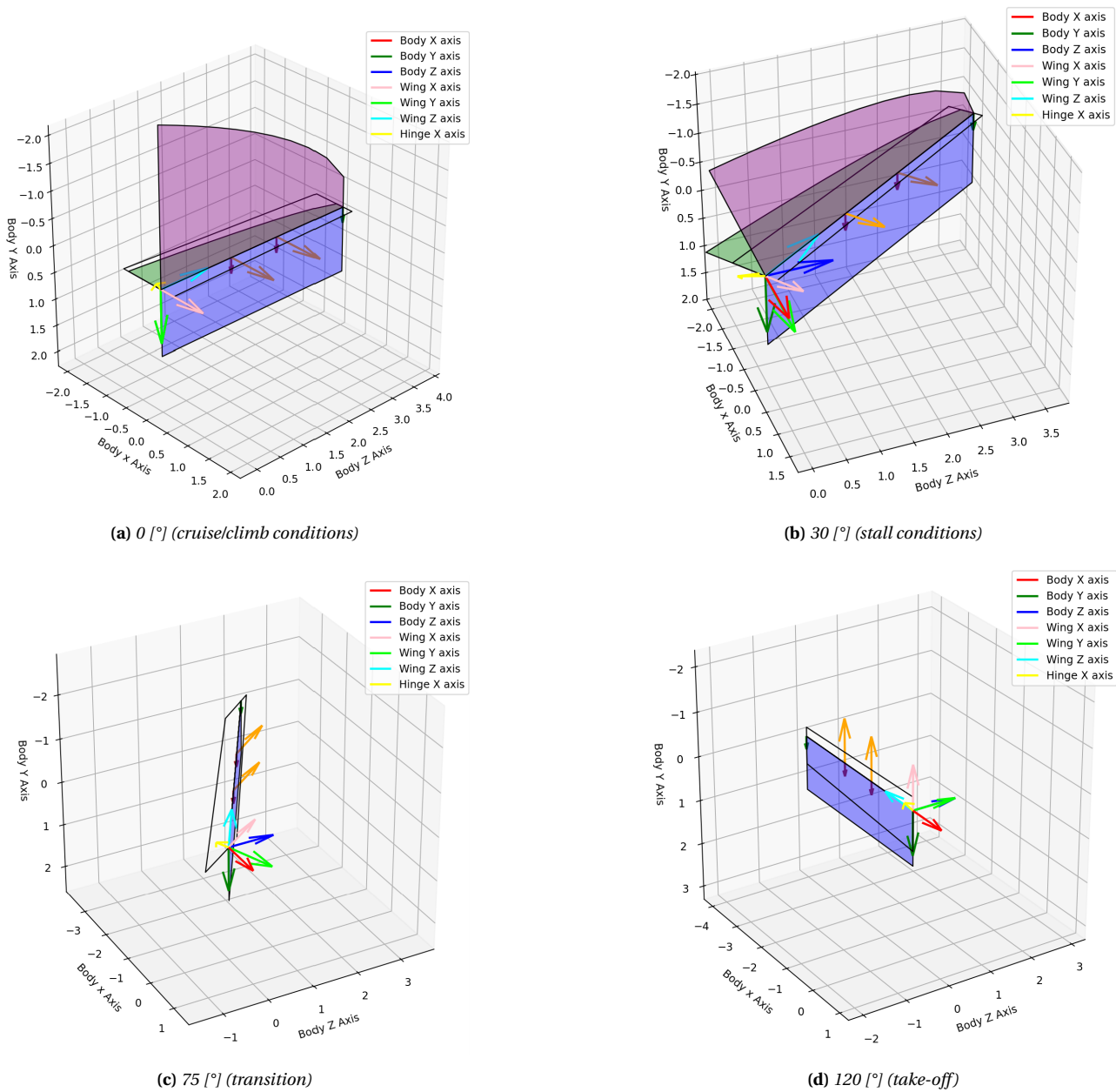
A general overview of the forces imposed on the wing is given below in Table 8.2. The forces are modelled as either point loads, which act on a single position, or distributed loads, which are defined with a starting point and direction vector representing the line they act through. Describing the orientation of these forces is critical to the model; as the wing rotates around the hinge some forces will rotate with it, while others will maintain their orientation. As such, the orientation and position is described in xyz coordinates and the axis they are respective to is stated. A better understanding of the loads on the wing, as well as the different axes, can be seen from Figure 8.2 with its behaviours explained in Table 8.3.

Table 8.2: Overview of forces imposed on the wing

Force	Type	Position	Relative to	Orientation	Relative to	Distribution
Wing weight	Distributed	Starting at the quarter point of the root chord and travelling with respect to the quarter point of the tip chord	Wing	Positive y direction	Body	Evenly distributed along the quarter-chord line, pointing in the positive y direction
Wing lift	Distributed	Starting at the quarter point of the root chord and travelling with respect to the quarter point of the tip chord	Wing	Negative y direction	Wing	Elliptical distribution, as described later
Wing drag	Distributed	Starting at the quarter point of the root chord and travelling with respect to the quarter point of the tip chord	Wing	Negative x direction	Wing	Elliptical distribution
Engine weight	Point	Located at the position of each engine	Wing	Positive y direction	Body	Weight of an individual engine and propeller
Propeller thrust	Point	Located at the position of each engine	Wing	Positive x direction	Wing	Thrust of an individual engine and propeller
Winglet reaction force due to weight	Point	Located at the winglet hinge position	Wing	Positive y direction	Body	Winglet weight
Winglet reaction force due to lift and drag	Point	Located at the winglet hinge position	Wing	Along the x - z plane	Wing	Winglet lift and drag

Table 8.3: Behaviour of the forces acting on the wing at various hinge angles

Force	Color	Observed behaviour
Lifting force	Purple distributed load	Highest at 0 degrees during cruise, still active near stall conditions, inactive for high hinge angles during folding.
Drag force	Green distributed load	Standard at 0 degrees, greatly increases near stall, inactive for high hinge angles during folding.
Propeller weights	Dark purple point loads	Always acting directly downwards.
Propeller thrusts	Orange point loads	Always acting directly forwards on the wing.
Wing weight	Blue distributed load	Always acting directly downwards.

**Figure 8.2:** Overview of the forces and positions of the wing at various angles

To solve the structure, the forces are all taken with respect to the wing axes. Then, distributed loads are discretized into a set of point loads representing the distribution across its span. All the forces are described with corresponding magnitude and position vectors. Then, the position vectors are snapped to a grid with a given spacing in order to avoid errors with force additions and to ease with integration later on. For a set of forces acting parallel to a given axis,

the shear distribution can be assessed along either of the 2 remaining perpendicular axes. Performing this analysis for each the axes yields a total of 6 shear diagrams for the wing structure.

The corresponding reaction forces and moments are calculated straightforwardly, since it is assumed only the wing hinge carries them, and are added to complete the diagrams.

8.2.4. Results and Discussion

Figure 8.3 displays the distribution of reaction forces on the wing hinge for a set of configurations. The transition from take-off to flight is illustrated by the set of yellow, gray and blue lines. As the plane is taking off, the reaction forces point in the positive y direction to counteract the thrusters. As the plane unfolds, more and more of that reaction force vector begins to transfer to the x and z axes as the thrust direction changes. As the yellow lines turn gray, the incident velocity reaches the stall speed and the planform begins to experience lift. As the gray lines turn blue, they begin to cluster together. This illustrates how the sudden presence of drag on the planform calls for a large reaction magnitude in the negative x direction, as well as how the hinge rotates the wing axis less and less as the θ approaches 0. Finally, the red arrow illustrates the reactions during dive, which point almost completely down due to the thrust output being lowered significantly from take-off.

Reaction forces on the hinge w.r.t the body axis [KN]

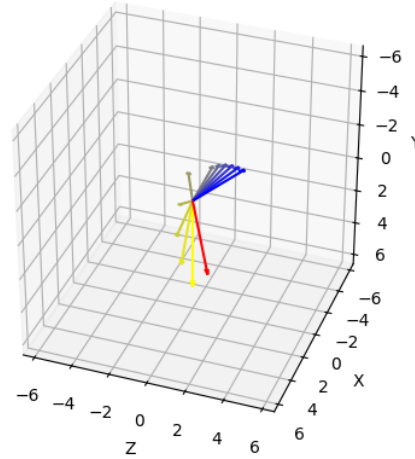


Figure 8.3: Visualization of the reaction forces on the wing hinge for the forward wing

Reaction moments on the hinge w.r.t the body axis [KNm]

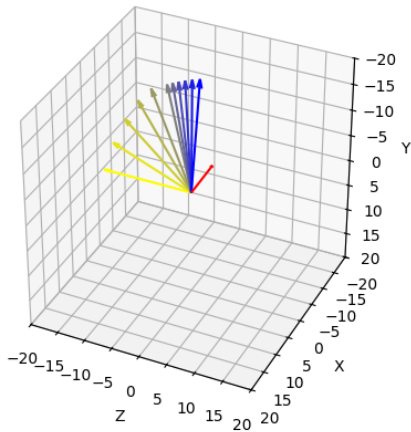
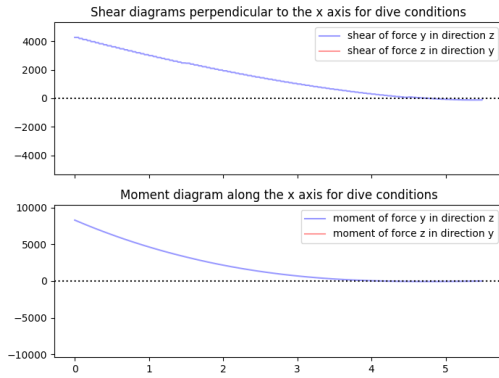


Figure 8.4: Visualization of the reaction moments on the wing hinge for the forward wing

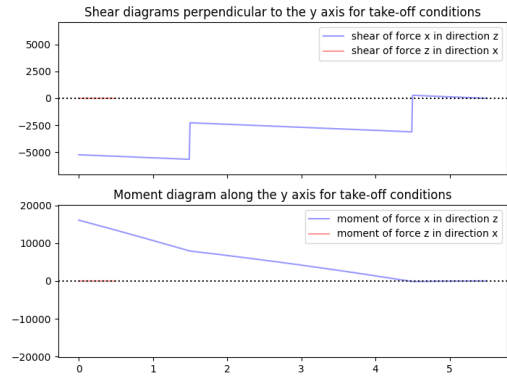
Figure 8.4 displays the distribution of reaction moments on the wing hinge. Note that the perspective of the graph has changed with respect to Figure 8.3 in order to better illustrate the way the moments change. However, the colouring of the lines remains the same, with yellow being take-off and transitioning to blue, which is climb speed, and finally red is unfolded with dive speed. At the beginning of take-off, the moment is purely in the negative z direction due to the thrust forces pointing in the negative y direction. As the plane unfolds, the moment begins to transition, gaining a negative x moment. The introduction of lift is more present in this graph, as the gray lines take a sharp turn and begin to accumulate similarly to the reaction forces. The final reaction moment has a negative y component, countering the thrust, and a negative x component, countering the lift. As the thrust is greatly reduced during cruise and dive, the y-component mostly disappears, leaving x component as shown by the red arrow.

As the hinge is positioned at the quarter chord line and the wings are assumed to be designed in a way that there are no moments around this line, moment analysis with respect to the wing axis can be decomposed into two effects: the y-moment, driven mainly by the propellers, and the x-moment, driven mainly by the wing lift. Figure 8.5b illustrates the effect of the propellers on the wing as it takes off. The discontinuities at positions $x = 1.5$ and $x = 4.5$ are due to the influence of the propeller thrust, which drives the moment. As a result, it can be shown that the wing experiences a significant reaction moment (≈ 170000 [Nm]) around the y axis. This is much larger than for conventional aircraft, and must be accounted for with extra structural components. Figure 8.5a follows the moment diagram for a standard

aircraft, with the effects of lift creating a moment around the x-axis of the wing. The effects of the winglet reactions and thruster weights are negligible for this condition, and can barely be seen on the graph.



(a) Moment diagrams for dive conditions w.r.t the wing axis



(b) Moment diagrams for take-off conditions w.r.t the wing axis

In conclusion, the transition loads are unintuitive and need a dedicated tool, but the reactions curve around the hinge in a way that allows the hinge structure to be placed to carry them. The main model limitations are the onset angle of lift and transition drag, both hard to model linearly and dependent on detailed geometry. Directions stay broadly correct and a higher load factor covers the magnitude error during hinge sizing. Take-off propeller loads are likely overestimated, so the true actuator moment may be lower. This is covered by a high safety factor on the actuators.

8.3. Wing Mass Estimation Strategy

This section performs an estimation on the five main components of one side of the wing structure: the fixed wings and folding wings for front and rear, as well as their respective winglet. The forces on each structure are analysed, and the primary load carriers are sized. Finally, the new mass of each component is used to perform the process again until the masses converge to a value.

8.3.1. Wing and Winglet Geometry

Only the left side is analysed as the wing is symmetric. Each wing is split by the folding mechanism into a fixed section and a folding section. With the connecting winglet, these form the five components in Figure 8.6. The planform is reconstructed from aerodynamics geometry of Chapter 3. The hinge sits at y_{hinge} from the root, splitting the planform; y_{hinge} is set mainly by ground space constraints. Winglets are idealised as parallelograms whose adjacent sides are the front and rear tip chords, so their geometry follows from taper and wing gap.

8.3.2. Assumptions and Software Diagram

Some assumptions were made when sizing the wing components shown in Table 8.4. The same code is used as in Subsection 8.2.2 however this time WW stands for wing and winglet models.

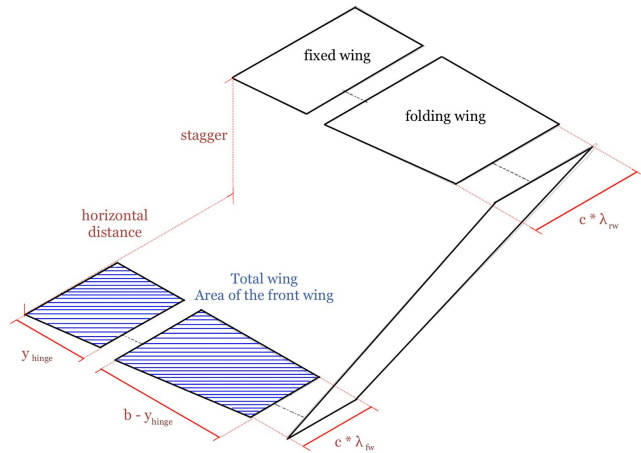


Figure 8.6: 3D visualization of the 5 wing components for the left side of the plane

Table 8.4: General assumptions used for the wing and winglet models

Assumption	Description
ASSU-WW-01	The wing is assumed to be a trapezoidal planform.
ASSU-WW-02	The winglet weight is assumed to be negligible when sizing the winglet structure. However, it still imposes reaction forces on the winglet hinges.

Continued on next page

Table 8.4 (Continued): General assumptions used for the wing and winglet models

Assumption	Description
ASSU-WW-03	The maximum loads experienced by the left and right wing structures are symmetric, and the structures can be assumed to be identical along the plane of symmetry (fuselage center).
ASSU-WW-04	The wing lift distributions can be modelled as elliptical with a non-zero lifting force at the tips.
ASSU-WW-05	The elliptical distribution for a given wing has the same shape for every configuration, and is scaled by V/V_{cruise} where V is the velocity incident to the wing at that given configuration.
ASSU-WW-06	The aerodynamic centre is assumed to be at the quarter-chord line.
ASSU-WW-07	The lift distribution along the winglet can be modelled using information from the lift at the tips of the front and rear wings, and is assumed to vary linearly with the winglet span.
ASSU-WW-08	The structures are designed in such a way that the center of gravity lies at the quarter-chord line.

Figure 8.7 shows the sizing methodology (functions in blue, variables in yellow). The loop uses assumed starting masses, and miscellaneous loads are imported from other departments.

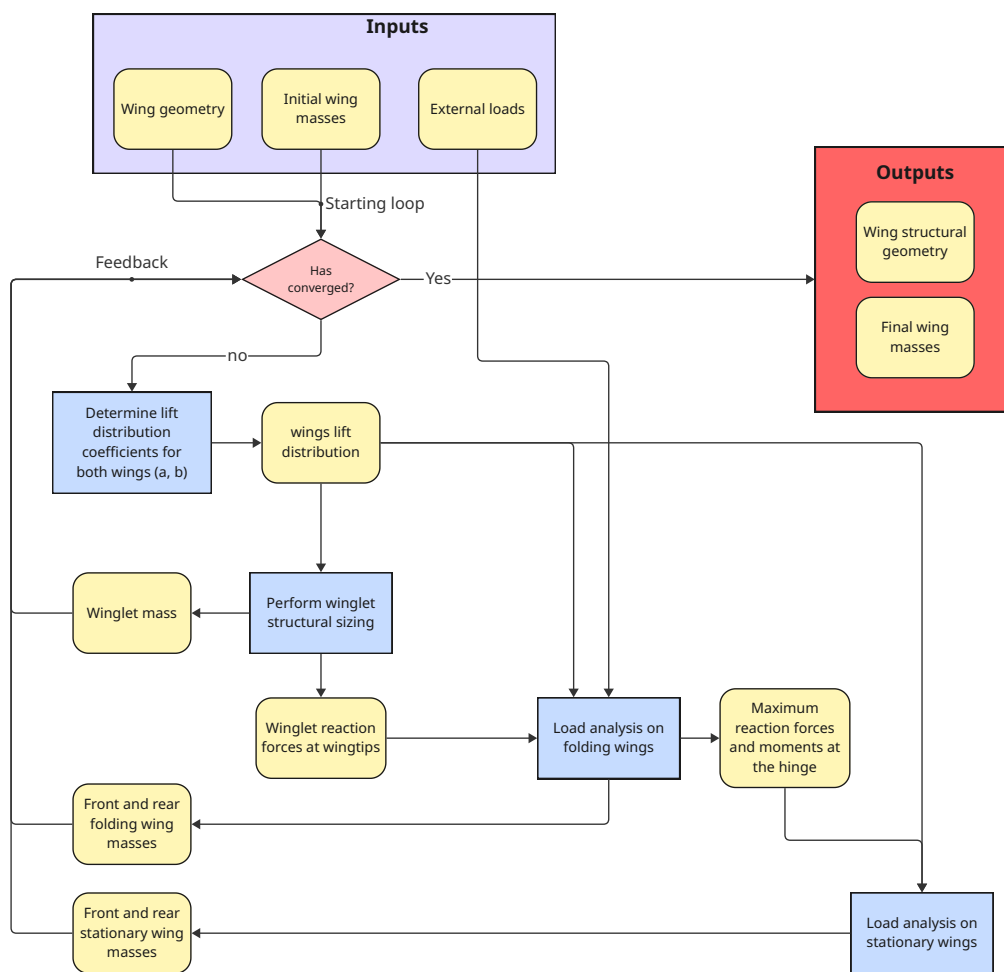


Figure 8.7: Software diagram for the wing sizing

8.3.3. Lift Distribution Estimation Across Wing

Due to the Prandtl design of the wing, the winglets and wings experience lift and drag for various configurations and flight speeds. Using the aforementioned assumptions, an estimate for the lift across the components can be estimated. Figure 8.8 gives a visual overview of how the lift varies with each component. The distributions across the wings remain elliptical. However, the lift at their tips is non-zero. In order for the lift distribution to remain continuous along the wings, thus achieving the desired effect of the prandtl wing, the lift at the front must change signs with respect to the lift at the back. Because of this, when analysing the lift at the winglet, at the front it equals the lift at the front wingtip, and as it travels backwards it changes sign and equals the negative of the lift at the back wingtip.

The elliptical distribution is parametrised by Equation 8.2.

$$y = f(x) = \beta \sqrt{-(x - \alpha)} \left(\frac{V_{cruise}}{V} \right)^2, 0 \leq x \leq b \quad (8.2)$$

where α and β are constants to be found. By observing how the wings are supposed to perform during cruise $\left(\frac{V_{cruise}}{V} \right) = 1$ it is possible to derive a and b from computed parameters and given constraints. The lift at the wingtip and the lift at the root are assumed to always have the same ratio $\frac{L_{root}}{L_{tip}}$, no matter the value of v , as the entire distribution scales evenly. Thus, by equating Equation 8.2 to this ratio by evaluating it at both points, alpha can be found.

$$\frac{f(b)}{f(0)} = \frac{\sqrt{-(b - \alpha)}}{\sqrt{\alpha}} = \frac{L_{root}}{L_{tip}} \quad (8.3)$$

The wing must also carry a share of aircraft weight in cruise, assumed proportional to wing area. Relating this to A_{wing} , A_{total} and MTOW and equating to the spanwise integral gives Equation 8.4, and solved for β .

$$\int_0^L b \sqrt{-(x - a)} \left(\frac{V_{cruise}}{V} \right)^2 dx = W_c = \frac{A_{wing}}{A_{total}} * MTOW * g \quad (8.4)$$

Performing these steps for the forward and rear wings yields their distributions. The lifts at the respective wingtips can then be used to calculate the lift distribution along the winglet, giving the lift for the entire wing structure.

8.3.4. Winglet Structural Sizing

Only lift is considered for the winglet, since its weight is distributed and negligible. The coordinate system is right-handed with x up the winglet and y along the lift (Figure 8.9). The two failure modes are spar bending and stringer buckling. Spars are idealised as I-beams pinned at each end, following the winglet lift distribution (Figure 8.9 and Figure 8.10). Pin reactions follow from force balance, giving the shear and moment diagrams; the peak moment and spar geometry give the peak stress via Equation 8.5:

$$\sigma_{peak} = \frac{M_{peak}}{I_{yy1} \cdot L_{max}} \quad (8.5)$$

where $I_{yy_{tot}}$ is the combined moment of inertia of every spar around the winglet y -axis and L_{max} is the maximum distance in the z -direction from the beam's neutral axis.

As for buckling, the stringers have a U-beam cross section with two flanges extending outwards at the U-tips. This profile was chosen based on aircraft structure principles.[51] Equation 8.6 gives the maximum force an individual stringer can carry before buckling, and Equation 8.7 relates the force experienced by an individual stringer to the winglet geometry and change in moment.

$$P_{crit} = \frac{\pi E I_{st}}{a^2} \quad (8.6)$$

$$P = \frac{\Delta M Q_{st}}{I_{tot}} \quad (8.7)$$

Where E is the material's young modulus, I_{st} is the moment of inertia of an individual stringer, I_{tot} is the total moment

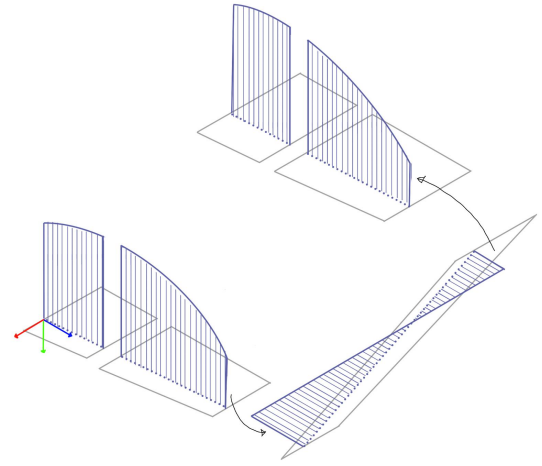


Figure 8.8: Lift distribution across components

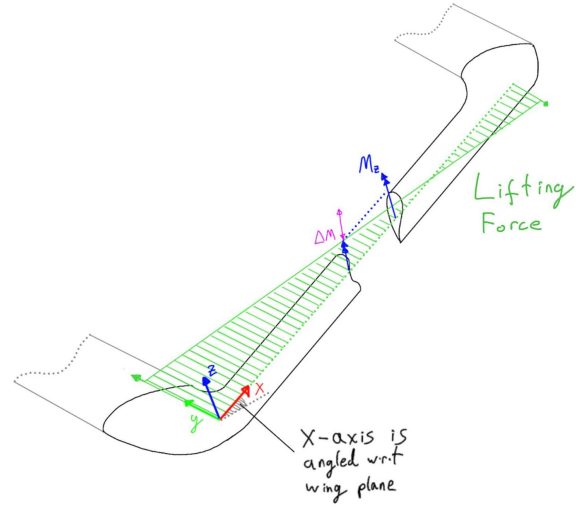


Figure 8.9: Visualization of the coordinate system and loads on the winglet

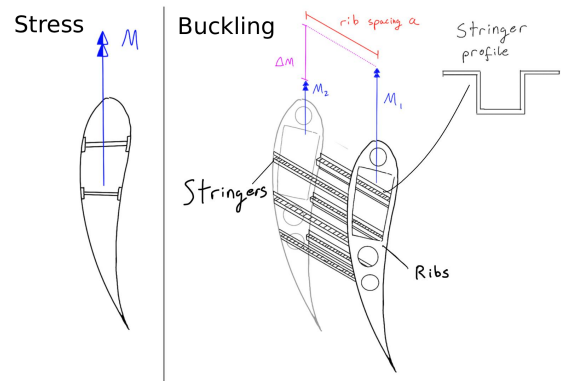


Figure 8.10: Loading conditions imposed on the winglet structure

of inertia for the winglet cross section (stringers and spars), a is the rib spacing, ΔM is the change in moment along the rib spacing and Q_{st} is the first moment of area for an individual stringer. All moments of inertia are taken along the winglet x-axis.

8.3.5. Folding Wing Structural Sizing

Folding-wing loads come from the Section 8.2 tool; the critical wing x-axis moment diagram is plotted and the structure sized as in Subsection 8.3.4. Take-off adds a +x force, because the six thrusters must support the full aircraft weight. Since the spar I-beams are not designed for this direction, a leading-edge spar is added with its section rotated 90 [°] to carry it.

Figure 8.11 shows the folding-wing layout and its hinge connection. A quarter chord spar is rotated to the hinge angle for predictable load transfer, so its rotation must be included when computing I_{xx} ; the leading edge spar's inertia enters I_{tot} for the buckling. Because the leading-edge spar section is asymmetric in I_{yy} , its neutral axis is found by standard methods [51] rather than assumed at the CG; the other spars' smaller contributions are also included. These steps are run for both folding wings. The non-folding components are then sized by treating them as cantilevers clamped at the fuselage, loaded by the folding-component hinge reaction (point load), distributed lift and weight; force/moment balance gives the moment diagram and hence the mass. The take-off thruster moment also loads the wing-box in torsion, with peak shear stress from Equation 8.8:

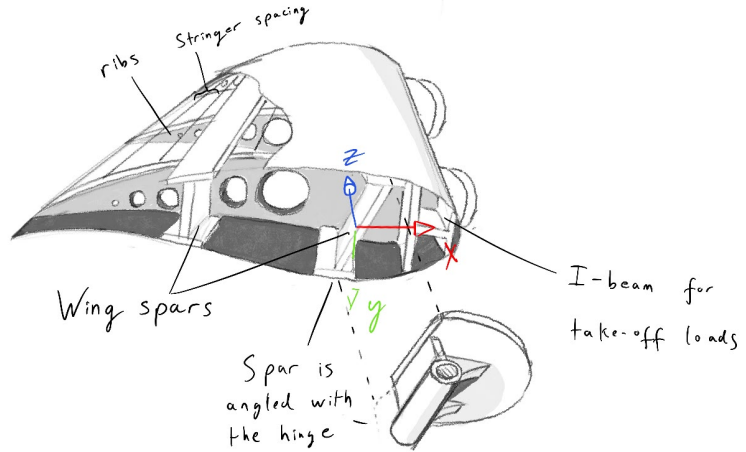


Figure 8.11: Visualization of the main folding wing elements to be sized

$$\tau = \frac{M_{max}}{2A_m t} \quad (8.8)$$

where M_{max} is the maximum moment experienced by the beam, A_m is the mean enclosed area of the cross section [51] and t is the thickness of the analysed section. Calculating τ for each section in the wing-box identifies the region where the highest shear stress is experienced. By comparing the highest shear stress with the maximum acceptable shear stress of the design material, it is possible to assess whether the structure will resist the torsional load.

With these calculations, a first value for the weights of each of the wing components is obtained. However, these calculations must be run in an iterative loop with the output masses feeding back as inputs until the masses converge to their respective values. Once this is achieved, the wings will have been sized for all relevant loads by physical means. It is expected that the mass of the wings is higher than originally expected due to the presence of winglets, buckling analysis and unconventional forces imposed by the eVTOL nature of the project.

8.4. Hinge and Actuator Mechanism: Design and Sizing

This section focuses on the mechanism required to fold and unfold the wings. A more detailed analysis is completed of the actuators which are required and a full overview of how the hinge is designed.

8.4.1. Folding Mechanism Layout

After determining the total loads on the hinge in Subsection 8.2.4 the design process of the mechanism can be performed. For the actuation of the wings, 4 electro-mechanical actuators will be used. Their placement is shown in Figure 8.12. The reason for having one actuator in the root of every wing instead of using only one per side, for moving the whole structure, is that the loads on the hinge will damage the hinge, since the resultant moment from both wings will be concentrated in only one point. By having two actuators per side this load is distributed and also the folding mechanism becomes more robust since it is no longer a single point failure. Each actuator can provide enough force to move the whole structure individually, but that would be done only in a case of failure of the other one.

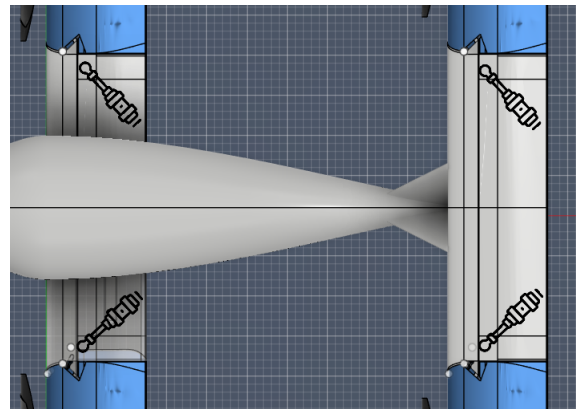
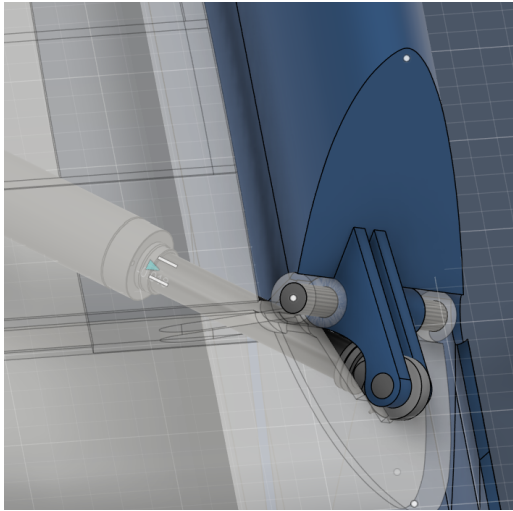


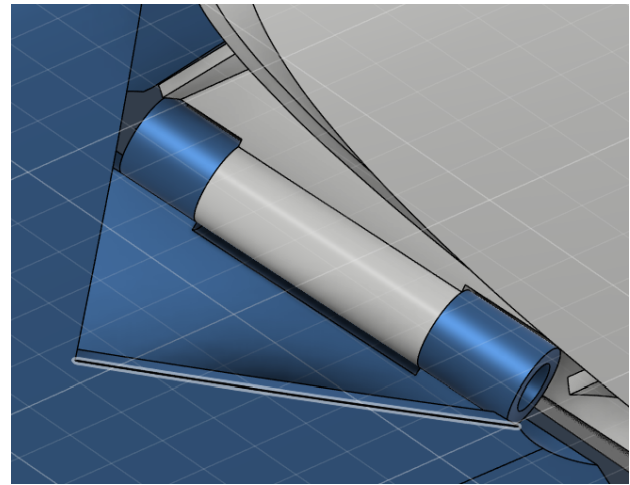
Figure 8.12: Actuators placement

The next step of the design process is to fit the actuators inside the wing box and connect them to the hinge. In order to do that the hinges in the root have an additional arm in comparison to the ones on the tips as shown in

Figure 8.13. This allows them to connect to the actuator. The other end of the actuator is connected to the rear spar of the wing box. In order for these links to work without any conflicts, the axes of rotation at the two ends of the actuator and the one around the wing folds need to be parallel.



(a) Root hinge mechanism.

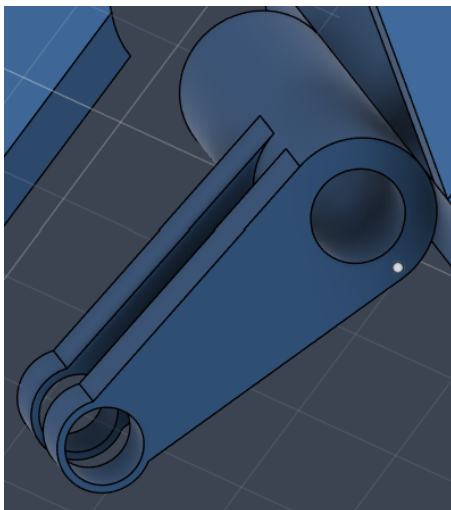


(b) Tip hinge mechanism.

Figure 8.13: Hinge mechanisms for folding mechanism.

8.4.2. Hinge Geometry

The hinge geometry was inspired from the hinge on the F4F-4 Sto-Wing . The changes that were implemented were the inversion of the folding direction and a slight adjustment in the angles of the hinge axes. Additionally the inner hinge was equipped with a moment arm that connects to the actuator as shown in Figure 8.13a. The design of this moment arm was done iteratively by testing it for the most extreme load cases determined in Subsection 8.2.4 by using a 3D model in Fusion 360 and running a FEM simulation recreating the desired load in magnitude and direction. The resultant stress distribution was used to reinforce the moment arm on the weaker regions. In Figure 8.14 the initial and final hinge designs are shown. Due to high tension force during cruise, due to the generated lift and the short resultant moment arm of the hinge in that position the hinge was reinforced in a more triangular shape as shown in Figure 8.14b. Another improvement is the small arc which is needed to ensure the actuator does not go through the hinge during the whole range of motion.



(a) Root hinge initial design.



(b) Root hinge final design.

Figure 8.14: Root hinge mechanism before and after iteration.

The sizing of the two holes was done by using the results from Subsection 8.2.4. The first mode of failure that was considered is the shear on the pins. The relation between the shear force and the cross sectional area is shown in Equation 8.9

¹Northrop Grumman F4F Sto-Wing, <https://www.history.navy.mil/content/dam/nhhc/research/histories/naval-aviation/nava1-aircraft/pdfs/f4f-4.pdf>, Accessed on 15th of June 2026

$$F_{shear} = \frac{A}{\sigma_{pin}} \quad (8.9)$$

The shear force F is caused by the resultant force from the wing loads in direction perpendicular to the pin axis. In addition, the moments perpendicular to the pin axis also create shear force at a distance $\frac{t}{2}$ from the centre of the hinge. This is the point where the two elements of the hinge connect. In Figure 8.15 the moment and force contributing to the shear are shown. In order to simplify the calculation of the resultant shear force, an assumption could be made, that the force F and the resultant force, caused by the moment, are aligned. This means that the shear force in the point where the two parts of the hinge connect will be:

$$F_{shear} = F + \frac{M}{t} \quad (8.10)$$

Using this equation and the loads calculated in the previous section, a diameter of 2 [cm] is calculated for the pin. This is in the condition that the load factor is 5 and the safety margin from the yield stress is 1.5. Next the outer diameter has to be determined. To do that the hinge will be tested in tension. The area of the cross section is shown in Figure 8.16 and can be calculated using Equation 8.11 as well as for calculating the stress.

$$A = (d_1 - d_2)t \quad \sigma_{hinge} = \frac{F}{A} \quad (8.11)$$

Using both equations in Equation 8.11, a formula for calculating the outer diameter is given in Equation 8.12:

$$d_1 = d_2 + \frac{F}{\sigma_{hinge}t} \quad (8.12)$$

The final outer diameter, calculated using the values for the loads from the previous section, is 2.5 [cm]. The same procedure is done for the second hole that is used for the connection with the actuator. The final dimensions of the hinge are shown in Figure 8.17

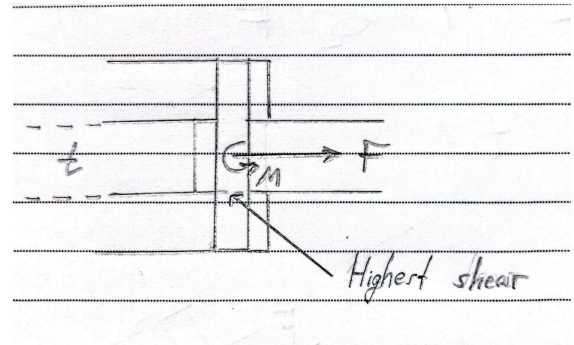


Figure 8.15: Sketch of the force and moment contributing to the shear in the pin

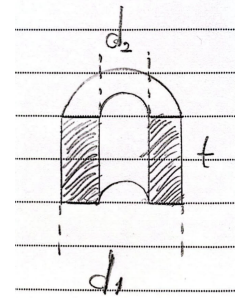


Figure 8.16: Sketch of the cross section of the hinge

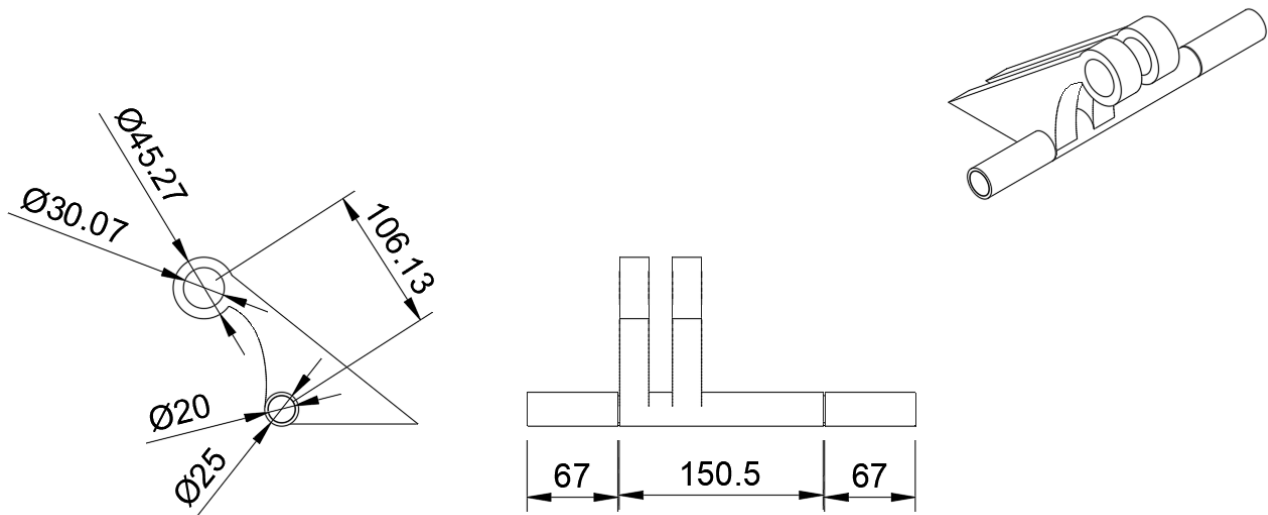


Figure 8.17: Technical drawing of the final hinge design.

8.4.3. Material Selection

The mechanism of the hinge consists of the two pins and the two parts of the barrel. For these parts it is crucial to pick materials that resist corrosion, have high yield strength and resist fatigue.

For the pin, stainless steel is used because of its superior corrosion resistance and high yield stress. The grade used

for this design is 17-4PH. However, the last FEM iterations, shown in Section 10.2, suggest that some yield in the pin might occur at the highest load-case. One of the suggested improvements is the use of different material, but that will be analysed in the future development of the design.

For the body of the hinge, two materials are considered: stainless steel (17-4PH) and titanium (Ti-6Al-4V). They both have superior corrosion resistance and high yield stress, between 760 and 1310 [MPa] depending on the heat treatment for 17-4PH and between 880 and 1100 [MPa] for Ti-6Al-4V. Both materials were tested with the structure from Figure 8.14 using the simulation module in Fusion 360. None of them experiences permanent yield. This means the titanium alloy is chosen due to its lower density.

8.4.4. Actuator Sizing

The actuators connected to the hinges must be able to provide enough force to counter the moment around their axes. Since the moment arm is rotating, the resultant force, perpendicular to the moment arm, changes with the hinge angle Figure 8.18. The moment generated by the actuator is given in Equation 8.13.

$$M = F_{actuator} l \cos \alpha \quad (8.13)$$

The range of angles for α is measured to be between $-62.6[^\circ]$ and $56.7[^\circ]$. From the results in Subsection 8.2.4 the highest value for the moment around the hinge axis is calculated to be 3282[Nm], during the fully unfolded position, where the angle of the hinge is $56.7[^\circ]$. This means that the force, that the actuator needs to provide can be calculated using Equation 8.13 and is equal to 56.4[kN].

After a market research the actuator Ewellix SRSA-U-3910 [52] is picked. The travel of the actuator is 200 [mm] and the maximum dynamic load it could move is 129 [kN]. This is enough to be able to move the folding mechanism in the highest load case and even fold the wings with only one actuator operative per side.

8.4.5. Mass Budget

In Table 8.5 the final masses of the components of the folding mechanism are listed.

Table 8.5: Mass budget table for folding mechanism components

Type	Mass [kg]	Quantity [-]	Total mass [kg]
Inner hinge	2.420	4	9.680
Outer hinge	0.936	4	3.744
Actuator	42.4	4	169.6
Total			183.024

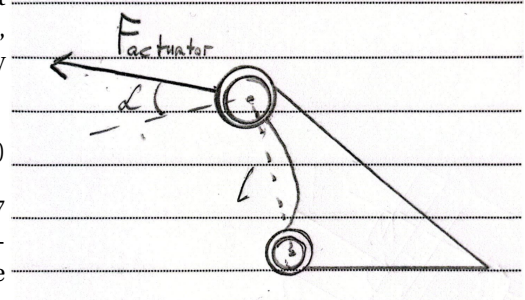


Figure 8.18: Sketch of the actuator force direction with respect to the moment arm of the hinge

8.5. Tail Mass Estimation

The empennage is two separate canted surfaces, so each is treated as an independent cantilever fixed at its root and sized from first principles (unlike the Class I regression masses): spar caps from the ultimate root bending moment via Euler-Bernoulli theory, a minimum-gauge skin (for manufacturability reasons), then a fixed inflation for ribs and fittings. The single-panel result is doubled.

8.5.1. Panel Geometry and Design Loads

The panel is modelled as a tapered trapezoidal cantilever of planform area S_t and aspect ratio A_t . Applying the definition of aspect ratio to the single panel fixes its span. The surface is canted upward from the horizontal by the dihedral angle Λ . The effective moment arm,

$$\ell_{panel} = \sqrt{A_t S_t}, \quad b_{half} = \ell_{panel} \cos \Lambda \quad (8.14)$$

For a trapezoid with taper ratio $\lambda = c_{tip}/c_{root}$ the area is $S_t = \frac{1}{2} c_{root}(1 + \lambda) \ell_{panel}$ which inverts to give the root chord, structural depth and effective depth of the spar:

$$c_{root} = \frac{2 S_t}{(1 + \lambda) \ell_{panel}}, \quad h_{spar} = (t/c) c_{root}, \quad h_{eff} = h_{spar} \cos \Lambda \quad (8.15)$$

Two independent load cases size the root. The first arises from the Prandtl-plane layout where the rear wing carries a fraction f_{rear} of the total lift that is reacted through the tail structure. At the limit load factor n_w the total vertical force is shared equally by the two panels which gives the following equations.

$$F_{vert} = \frac{f_{rear} n_w m g}{2}, \quad M_{rear} = b_{half} F_{vert} \quad (8.16)$$

²Titanium, <https://www.solitaire-overseas.com/blog/what-is-the-density-of-titanium/>, Accessed on 6th of June 2026

The second case requires the tail to withstand its own air load at maximum deflection at dive speed, $V_{\text{dive}} = k_{\text{dive}} V_{\text{cruise}}$. The root moment is then found using dynamic pressure and aerodynamic force:

$$q_{\text{dive}} = \frac{1}{2} \rho_0 (k_{\text{dive}} V_{\text{cruise}})^2, \quad F_{\text{aero}} = q_{\text{dive}} S_t C_{N,\text{max}}, \quad M_{\text{aero}} = \frac{F_{\text{aero}} \ell_{\text{panel}}}{2} \quad (8.17)$$

8.5.2. Ultimate Moment and Spar Sizing

The two limit moments are summed and scaled to the ultimate level by the safety factor η_{SF} (FAR/CS 25.303 [53]).

$$M_{\text{root}} = \eta_{\text{SF}} (M_{\text{rear}} + M_{\text{aero}}) \quad (8.18)$$

which the spar must carry as a force couple. Idealising the spar as two caps separated by the effective depth h_{eff} , the cap force is $M_{\text{root}}/h_{\text{eff}}$ and sizing each cap so that the CFRP compression allowable σ_{allow} is just reached gives a cap area of $M_{\text{root}}/(\sigma_{\text{allow}} h_{\text{eff}})$. Multiplying by the cantilever length yields a representative cap material volume that is given below.

$$V_{\text{caps}} = \frac{M_{\text{root}} \ell_{\text{panel}}}{\sigma_{\text{allow}} h_{\text{eff}}} \quad (8.19)$$

Adding a shear web inflates this volume. For this analysis, it is assumed to be by 40 %, and applying the CFRP density gives the spar volume and consecutively the spar mass.

$$V_{\text{spar}} = 1.4 V_{\text{caps}}, \quad m_{\text{spar}} = \rho_{\text{CFRP}} V_{\text{spar}} \quad (8.20)$$

The skin is governed by the minimum manufacturable CFRP gauge t_{min} rather than by stress and covers both surfaces of the panel. Finally two independent panels give the total mass:

$$m_{\text{skin}} = 2 S_t t_{\text{min}} \rho_{\text{CFRP}}, \quad m_{\text{panel}} = \frac{m_{\text{spar}} + m_{\text{skin}}}{0.75}, \quad m_{\text{tail}} = 2 m_{\text{panel}} \quad (8.21)$$

Where 0.75 was a safety factor applied to the mass to account for non computed components such as ribs and fittings which do not carry the aforementioned load.

8.6. Landing Gear Mass Estimation

In early sizing the landing gear was infeasible it could not meet the mass budget while staying elastic at the standard drop height. Because the gear sits in the MTOW convergence loop, any mass error propagates into the converged MTOW and can hide a constraint rather than resolve it, so it is sized in detail here.

The landing gear is therefore sized based on the CS/FAR Part 27 [54] reserve energy drop while staying elastic at the limit drop constraint. Five architectures are each sized for minimum mass under a common set of drop, peak-deceleration, stroke and footprint constraints using sequential least-squares programming (SLSQP), the same optimiser as the outer MTOW loop. This section derives each model and lists its assumptions and constants.

8.6.1. Overview and Gear Architecture

The gear is a skid type: two ground rails joined by two cross members, each bending down at two corners to give four legs, with an energy absorber at each leg skid hinge. The two cross tubes, four legs and two rails form a fixed load bearing skeleton common to all concepts; only the four absorbers differ between architectures. Capacities and masses are total values one member is sized, multiplied by the count, then the shared structure is added. The five architectures are:

Bent metal cross tube: ductile aluminium tube forming a plastic hinge. Not reusable.

Metal leaf spring: ductile steel leaf/bow spring. Reusable.

Composite leaf: CFRP leaf that is elastic to brittle failure then has fracture. Not reusable.

Crushable hybrid: stiff steel leaf (limit) + honeycomb crush core (reserve). Not reusable.

Elastomeric block: hysteretic rubber block mounts. Reusable.

8.6.2. Modelling Assumptions

The assumptions made while sizing the landing gear are listed below in Table 8.6. The code used was similar as described in Subsection 8.2.2 but the code used this time is LG for landing gear.

Table 8.6: Landing gear sizing modelling assumptions.

Code	Assumption	Description
------	------------	-------------

ASSU-LG-01	Quasi-static energy balance	The gear absorbs the impact kinetic energy plus the work done by aircraft weight over the stroke.
ASSU-LG-02	Symmetric, level drop	Landing is level with symmetric load distribution, so the effective drop mass is shared equally across the gear.
ASSU-LG-03	Parallel members share one stroke	Parallel members deflect equally. Stiffnesses, absorbed energies and reaction forces add, but stroke length (d_y , $d_{\max}$) does not.
ASSU-LG-04	Tip loaded Euler-Bernoulli cantilever	All bending members are tip loaded cantilevers with $k = 3EI/L^3$.
ASSU-LG-05	Plastic-hinge rotation capacity	Estimated from ultimate strain ε_u and an assumed plastic hinge length.
ASSU-LG-06	No lift credit	Lift is neglected ($L = 0$), so the gear reacts the full aircraft weight, conservatively interpreting CS/FAR-27.725(b) [54].
ASSU-LG-07	Geometric mass estimates	Tubes, rails and skeleton members are hollow structural members ($m = \rho A \ell$). Absorber masses use simple physically scaled relations.
ASSU-LG-08	Gear self weight feedback	Gear mass is updated via the outer MTOW convergence loop rather than a separate inner iteration, consistent with the wing and tail models.

8.6.3. Drop Conditions and Energy Targets

Firstly, the effective drop mass from CS/FAR-27.725(b) is considered [53]. In this model, lift credit is not used so, the landing gear is assumed to react the full aircraft weight. This is a conservative estimate, which makes the landing gear able to be effective in many flight situations.

$$W_e = W \frac{h + (1 - L)d}{h + d}, \quad W = m_{\text{MTOW}}g, \quad M_{\text{eff}} = \frac{W_e}{g} \quad (8.22)$$

Here, $h = H_L$ is the limit free-drop height and $d = D_{\text{EST}}$ is the assumed landing gear deflection used to estimate the effective drop mass. The lift fraction is set to zero ($L = 0$) With this conservative assumption, the effective weight becomes equal to the aircraft weight $W_e = W$. Therefore, the effective drop mass is equal to the maximum take-off mass $M_{\text{eff}} = m_{\text{MTOW}}$.

This means that the landing gear is sized as if it must absorb the full drop energy of the aircraft without any lift assistance. Next, sink speeds and energy targets are considered. The impact speed is obtained from the free-drop height. The limit sink speed is based on the limit drop height H_L , while the reserve sink speed is based on a drop height that is 1.5 times larger [53].

$$v_L = \sqrt{2gH_L}, \quad v_R = \sqrt{2g(1.5H_L)} = \sqrt{1.5}v_L \quad (8.23)$$

For a given stroke δ , the total energy absorbed by the landing gear is the sum of the kinetic energy at impact, and the work done by the effective aircraft weight during the stroke.

$$E_{\text{limit}}(\delta) = \frac{1}{2}M_{\text{eff}}v_L^2 + M_{\text{eff}}g\delta, \quad E_{\text{reserve}}(\delta) = \frac{1}{2}M_{\text{eff}}v_R^2 + M_{\text{eff}}g\delta \quad (8.24)$$

8.6.4. Energy-Absorber Concept Models

Each concept is modelled for one quantity then scaled to the full gear. All bending concepts are tip loaded cantilevers of length L and rigidity EI ; the stiffness factor 3 comes from the cantilever boundary condition. The root moment peaks first, so a plastic hinge forms there with rotation capacity from ductility over a fixed fraction hinge length. The common relations are in Table 8.7.

Table 8.7: Common bending framework governing relations.

Quantity	Expression
Stiffness	$k = F/\delta = 3EI/L^3$
Root moment	$M = FL$
Yield moment	$M_y = \sigma_y Z$
Plastic moment	$M_p = \sigma_y Z_p$
Yield force	$F_y = M_y/L$
Yield deflection	$d_y = F_y/k$
Elastic energy	$U_e = \frac{1}{2}kd_y^2$
Hinge rotation	$\theta = (\varepsilon_u/c)\ell_h$
Plastic energy	$U_p = M_p\theta n_{\text{hinge}}$
Maximum stroke	$d_{\max} = d_y + \theta L$
Maximum force	$F_{\max} = \max(F_y, M_p/L)$

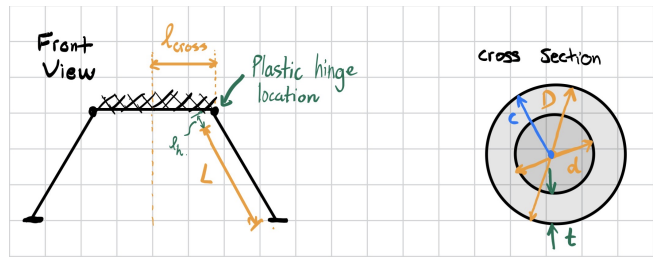


Figure 8.19: Annotated dimensioning of landing gear

Here, Z and Z_p are the elastic and plastic section moduli, ε_u is the ultimate strain, c is the distance from the neutral axis to the outer fibre, and ℓ_h is the assumed plastic hinge length.

Bent metal cross tube. The bent cross tube is modelled as a hollow circular tube with outer diameter D , wall thickness t , inner diameter $d = D - 2t$ and cantilever length L ; relations in Table 8.8. Here, η_{tube} is the hinge length factor, representing the fraction of the tube diameter over which plastic rotation develops. Two hinges are assumed since bending can occur on both sides of the cross tube, and ℓ_{cross} is the under-fuselage span of the tube. This concept is expected to yield before absorbing the full limit drop energy elastically, so it is treated mainly as a plastic energy-absorbing concept rather than a fully reusable elastic spring.

Metal leaf spring. The metal leaf spring is a curved rectangular beam (width b , thickness t , projected length L); relations in Table 8.9. Here, η_{leaf} is the hinge-length factor and η_{dev} is the developed-length factor for curvature. Sized to stay elastic at the limit drop, so it is reusable. An image of the cross section and dimensions can be seen in Figure 8.20.

Table 8.8: Bent metal cross tube governing relations.

Quantity	Expression
Inner diameter	$d = D - 2t$
Area	$A = (\pi/4)(D^2 - d^2)$
Second moment of area	$I = (\pi/64)(D^4 - d^4)$
Elastic section modulus	$Z = I/(D/2)$
Plastic section modulus	$Z_p = (D^3 - d^3)/6$
Extreme fibre distance	$c = D/2$
Plastic hinge length	$\ell_h = \eta_{\text{tube}} D$
Number of hinges	$n_{\text{hinge}} = 2$
Member mass	$m_1 = \rho A(2L + \ell_{\text{cross}})$

Table 8.9: Metal leaf spring governing relations.

Quantity	Expression
Area	$A = bt$
Second moment of area	$I = bt^3/12$
Elastic section modulus	$Z = bt^2/6$
Plastic section modulus	$Z_p = bt^2/4$
Extreme-fibre distance	$c = t/2$
Plastic hinge length	$\ell_h = \eta_{\text{leaf}} t$
Number of hinges	$n_{\text{hinge}} = 2$
Material mass	$m_1 = \rho A(\eta_{\text{dev}} L)$

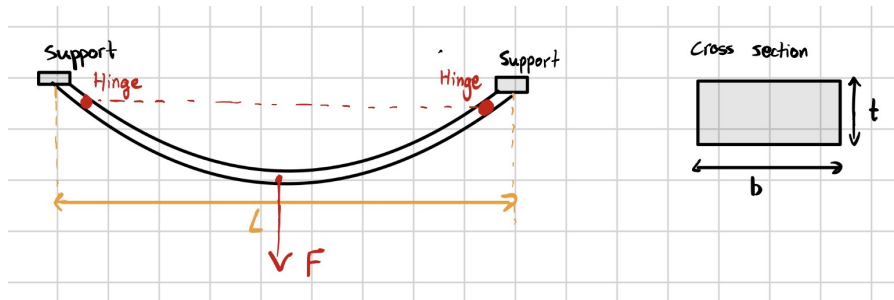


Figure 8.20: Assumed load case on metal leaf spring

Composite leaf. The composite leaf has the same geometry as the metal leaf (identical stiffness), but is elastic to brittle failure, after which energy is absorbed by delamination and crushing; relations in Table 8.10. Here, G_c is the fracture toughness, and η_{delam} and η_{crush} are the delamination and post-failure crush fractions. Not reusable, and can become stroke-limited at higher mass due to its lower stiffness. This figure is depicted in Figure 8.21b.

Crushable hybrid. The crushable hybrid is a stiff steel leaf carrying the limit load elastically plus a honeycomb core crushing the reserve energy; relations in Table 8.11. The design variables are the steel leaf thickness t_{leaf} , the honeycomb crush area A_c and the honeycomb crush stroke s_c , with fixed leaf width $b = b_{\text{leaf}}$ and length $L = L_{\text{leaf}}$. Here, η_{stroke} is the usable-stroke efficiency and ρ_{hc} the core density. This is akin to the appearance in Figure 8.21b.

Table 8.10: Composite leaf governing relations.

Quantity	Expression
Failure force	$F_{\text{fail}} = (\sigma_y Z)/L$
Failure deflection	$d_{\text{fail}} = F_{\text{fail}}/k$
Elastic energy (pre-failure)	$U_e = 0.5 k d_{\text{fail}}^2$
Delamination energy	$U_p = G_c (b \eta_{\text{delam}} L)$
Maximum force	$F_{\text{max}} = F_{\text{fail}}$
Maximum stroke	$d_{\text{max}} = d_{\text{fail}} + \eta_{\text{crush}} L$

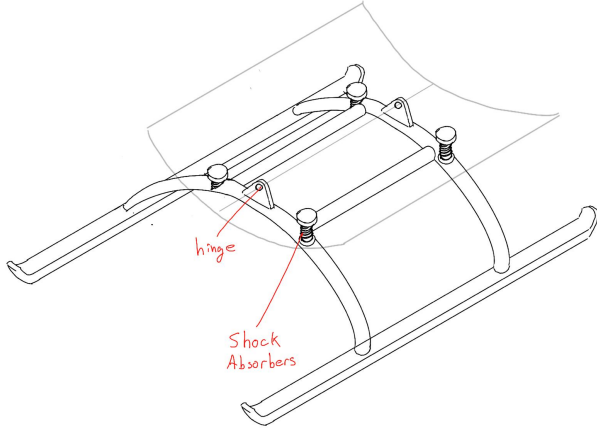
Table 8.11: Crushable hybrid governing relations.

Quantity	Expression
Yield force (leaf)	$F_y = (\sigma_y Z)/L$
Yield deflection (leaf)	$d_y = F_y/k$
Elastic energy (leaf)	$U_e = 0.5 k d_y^2$
Honeycomb plateau stress	$\sigma_c = \sigma_{\text{hc}}$
Crush force	$F_{\text{cr}} = \sigma_c A_c$
Honeycomb energy	$U_p = F_{\text{cr}} s_c \eta_{\text{stroke}}$
Maximum force	$F_{\text{max}} = \max(F_y, F_{\text{cr}})$
Maximum stroke	$d_{\text{max}} = d_y + s_c$
Total member mass	$m_1 = \rho A(\eta_{\text{dev}} L) + \rho_{\text{hc}} A_c s_c$

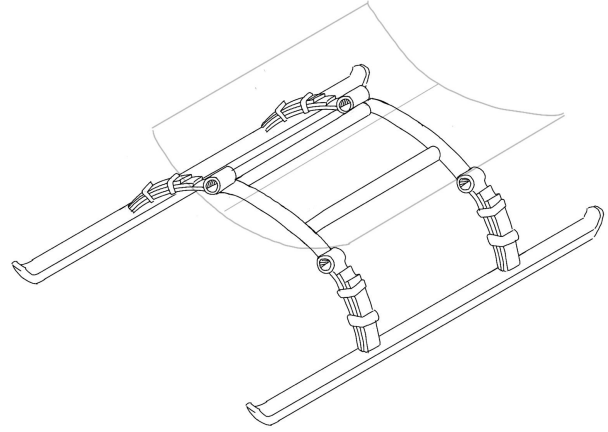
Elastomeric block. The elastomeric block is modelled as a rubber energy absorber with stiffness k_b , maximum compression d_m and hysteretic loss fraction λ , where only part of the stored elastic energy is dissipated as heat during unloading; relations in Table 8.12. Here, m_{mount} is the bracket/housing mass and c_{load} is the rubber/support mass per unit force. Reusable, dissipating energy by internal damping. This configuration is illustrated in Figure 8.21a.

Table 8.12: Elastomeric block governing relations.

Quantity	Expression
Stored elastic energy	$U_e = 0.5 k_b d_m^2$
Dissipated energy	$U_p = \lambda U_e$
Maximum force	$F_{\text{max}} = k_b d_m$
Yield / maximum stroke	$d_y = d_{\text{max}} = d_m$
Block mass	$m_1 = m_{\text{mount}} + c_{\text{load}} F_{\text{max}}$



(a) Elastomeric block landing gear



(b) Metal leaf landing gear

Figure 8.21: Landing gear absorber concepts.

As can be seen in Figure 8.22 the important metric is the energy that can be absorbed by each landing gear, which must be compared per configuration instance. This is an example of one instance.

8.6.5. Whole Gear Assembly and Footprint

A single energy absorbing member is the building block where the real gear uses n of them deflecting in parallel. Since they share the same stroke, their stiffnesses, stored energies and reaction forces simply add while the deflection at yield (d_y) and the maximum deflection (d_{max}) are unchanged as shown in Equation 8.25.

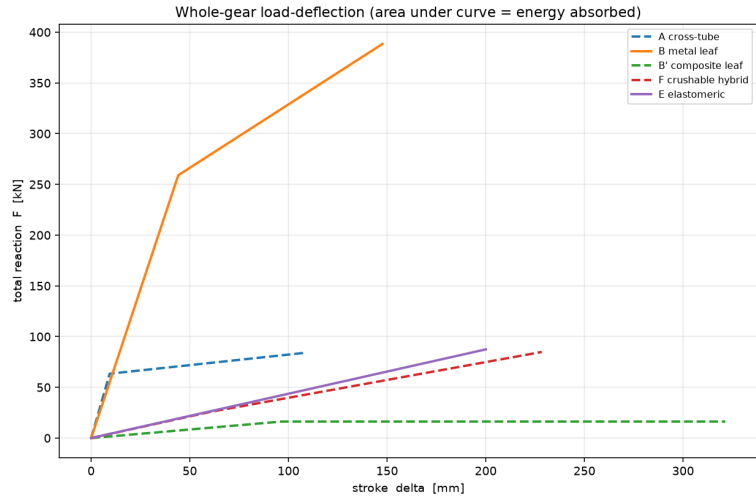


Figure 8.22: Assumed force vs deflection curves per landing gear

$$K = nk, \quad U_e^{\text{gear}} = nU_e, \quad U_p^{\text{gear}} = nU_p, \quad F = nF_{\text{max}} \quad (8.25)$$

On top of the n members, every architecture also carries the same supporting structure as follows. Two longitudinal skid rails and a shared skeleton of two transverse cross tubes plus four legs (one per hinge). The total gear mass is therefore the n members plus the two rails plus the skeleton.

$$m_{\text{gear}} = n m_1 + 2 m_{\text{rail}} + m_{\text{skel}}, \quad (8.26)$$

where m_1 is the mass of one member. Both the rails and the skeleton are hollow aluminium tubes (density $\rho_{\text{Al}} = 2810 \text{ [kg/m}^3]$ ¹) so their masses follow directly from the annular cross section and the tube length:

$$m_{\text{rail}} = \rho_{\text{Al}} \frac{\pi}{4} (D_{\text{rail}}^2 - d_{\text{rail}}^2) L_{\text{rail}}, \quad L_{\text{rail}} = 0.65 L_{\text{FUS}}, \quad m_{\text{skel}} = \rho_{\text{Al}} \frac{\pi}{4} (D_{\text{frame}}^2 - d_{\text{frame}}^2) L_{\text{skel}}. \quad (8.27)$$

Rails are 80/72 [mm] (4 [mm] wall) and the skeleton tubes are 60/54 [mm] (3 [mm] wall). Rail length is 65% of the fuselage length with skeleton length is two 0.60 [m] cross tubes plus the four 0.45 [m] legs, giving the total length of it as: $L_{\text{skel}} = 2(0.60) + 4(0.45) = 3.0 \text{ [m]}$.

¹Click Metal, https://www.clickmetal.co.uk/blog/post/the-science-behind-aluminiums-ultra-low-weight?srsltid=AfmB0OrS_1Hg5ECItTwDfJ5Kgen4ulJ5yMh-JGaYD7iSNNQ_XKLaxko1, Accessed on 23rd of June 2026

This skeleton is deliberately added to every architecture, even though the bending members already span transversely inside their own m_1 . This small doubled effect is kept on purpose as a conservative margin.

Finally, the lateral track is the fuselage width plus a per-side splay:

$$\text{track} = 2.0 \text{ [m]} + 2 L_{\text{splay}}, \quad L_{\text{splay}} = \begin{cases} L & \text{(bending arches, leaf/tube length)} \\ 0.45 \text{ [m]} & \text{(elastomeric, leg length).} \end{cases} \quad (8.28)$$

8.6.6. Sizing Optimisation

Each architecture is sized by choosing its design vector $\mathbf{x}$ to make the whole gear as light as possible. Five constraints are set and must all be satisfied (written so each is ≥ 0 at feasibility):

$$\min_{\mathbf{x} \in [\mathbf{x}_l, \mathbf{x}_h]} m_{\text{gear}}(\mathbf{x}) \quad (8.29)$$

This is subject to the constraints in Table 8.13.

In words, the gear must (1) stay elastic through a limit drop, (2) absorb the reserve drop using elastic plus plastic energy, (3) keep the peak reaction below 20 g, (4) fit the 0.40 [m] stroke, and (5) keep its track within 6 [m]. SLSQP is run from an initial estimate plus 48 random restarts (kept because some designs sit in a small feasible space a single start can miss); the lightest feasible optimum is kept. To stay valid at any take-off mass, the upper bounds are calibrated at a 700 [kg] reference and grown with mass (Equation 8.30) with exponent $s_i \approx 1$ for energy-carrying variables and 0 for stroke, thickness and dimensionless variables so the stroke envelope and load factor cap are preserved.

$$x_{hi,i} \leftarrow x_{hi,i} \cdot \max\left(1, (M_{\text{eff}}/700 \text{ kg})^{s_i}\right) \quad (8.30)$$

8.6.7. Architecture Trade-off and Selection

Every feasible architecture is scored on seven criteria. Three are obtained directly from the sizing results that are the specific energy absorption (energy stored per unit gear mass), the ultimate load factor (the force reaction quantified in g), and the gear's own mass:

$$\text{SEA} = \frac{U_e^{\text{gear}} + U_p^{\text{gear}}}{m_{\text{gear}}}, \quad n_{\text{pk}} = \frac{F}{M_{\text{eff}} g} \quad (8.31)$$

The fourth criterion is binary whether the gear is reusable, such as if it survives the limit drop without permanent damage. The last three are engineering judgements from 0 to 1 that takes into account how easy it is to tune, risks for certification and cost. Two safety margins are also presented that are elastic and reserve margins, telling you how much extra capacity you have:

$$\text{MS}_e = \frac{U_e^{\text{gear}}}{E_{\text{limit}}} - 1, \quad \text{MS}_R = \frac{U_e^{\text{gear}} + U_p^{\text{gear}}}{E_{\text{reserve}}} - 1 \quad (8.32)$$

Each criterion is normalised across the full set of feasible architectures before scoring. Benefit criteria (where a larger score is better) are scaled as $(v - v_{\min})/(v_{\max} - v_{\min})$, and cost criteria (where a smaller score is better) as $(v_{\max} - v)/(v_{\max} - v_{\min})$. Mass is treated differently, using the ratio $v_{\min}/v$: a design k times heavier than the lightest scores $1/k$ rather than simply ranking last. This prevents a much heavier gear from recovering a win through strong performance on the lower-weighted qualitative criteria.

The criteria are then combined with weights that sum to one, and the architecture that scores the highest wins. All weights and scores are provided in Table 8.14.

8.6.8. Sensitivity Analysis

The trade-off picks one winner at one set of inputs and weights. Since the weights are judgements and several drop/absorber inputs are placeholders, the winner's robustness is checked, sizing every architecture at a representative full landing mass.

Table 8.14: Trade-off criteria per landing-gear architecture. Columns marked † are computed by the sizing (placeholders, to be filled from a sizing run); the remaining columns are fixed design inputs (reusability is a binary architecture trait, the last three are 0–1 engineering judgements).

Architecture	$m_{\text{gear}}^{\dagger}$ [kg]	SEA [†] [J/kg]	$n_{\text{pk}}^{\dagger}$ [g]	Reusable	Tune	Cert. risk	Cost
Weights w_c	0.45	0.18	0.11	0.07	0.06	0.09	0.04
Cross-tube	...	...	...	No	0.3	0.1	0.1
Metal leaf	...	...	...	Yes	0.5	0.2	0.2
Composite leaf	...	...	...	No	0.5	0.8	0.5
Crushable hybrid	...	...	...	No	0.9	0.5	0.5
Elastomeric	...	...	...	Yes	0.4	0.4	0.2

The first check perturbs the seven criterion weights by up to $\pm 30\%$ about their initial values, renormalises them to sum to one and runs the trade off again. Across these perturbed weightings the elastomeric block is the design that is placed first in about 98% of cases, with the metal leaf taking the remaining $\sim 2\%$. The choice is therefore not based on the exact weights because mass dominates with a weight of 0.45. The second, deeper check varies the physical assumptions themselves. Eleven inputs are swept over bounds.

Table 8.15: Design variable ranges.

Parameter	Range
Limit drop height	0.20–0.35 [m]
Impact deflection (sets effective mass)	0.08–0.25 [m]
Rotor lift credit	0–50%
Peak deceleration cap	12–25 g
Stroke envelope	0.30–0.50 [m]
Honeycomb crush stress	1.5–4.0 [MPa]
Honeycomb density	50–120 [kg/m ³]
Usable crush fraction	0.60–0.85
Elastomer mount fixed mass	0.8–1.8 [kg]
Elastomer mass per unit load	0.6–1.5 $\times 10^{-4}$ [kg/N]
Skid rail wall thickness	2–6 [mm]

To keep parameter effects clean and the results repeatable, the sizer's random restarts are set per evaluation. In this way, a given input vector maps to one mass. Three complementary methods are then applied from quick and local to fully global.

Tornado (local): Sweeping each input low to high with the others held nominal (Figure 8.23) ranks the local drivers of the selected gear mass about its 48.2 [kg] nominal. The skid rail mass and the elastomer mass per load coefficient dominate, each producing swings of order 20 [kg], with the limit drop height and elastomer fixed mass close behind ~ 15 [kg]. The honeycomb, crush, stroke envelope and deceleration cap inputs are locally flat because they govern concepts the sizer never selects. Notably, impact deflection registers almost no local swing here yet is significant globally. The sign of an effect that lives off the nominal axes and the reason a local method alone is not trusted.

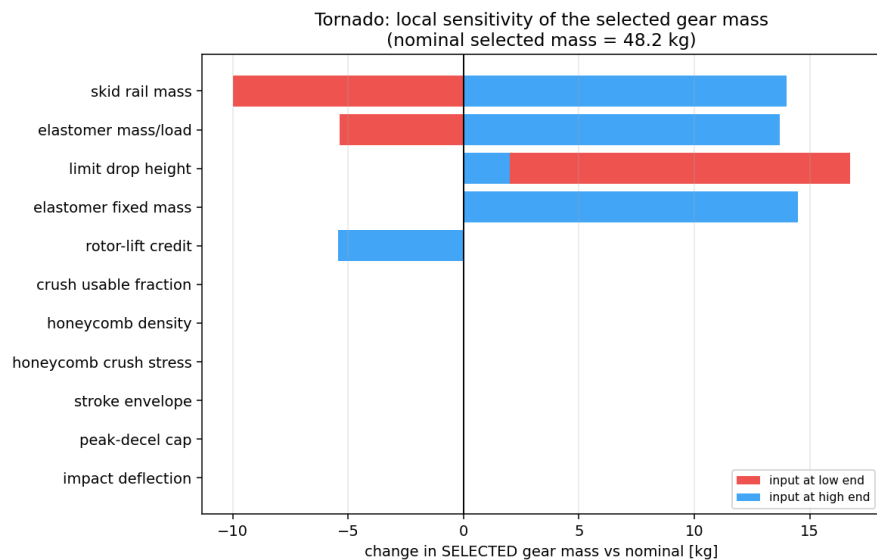


Figure 8.23: Tornado (local one-at-a-time): change in the selected gear mass when each input is moved to its low (red) and high (blue) end about the 48.2 [kg] nominal.

Sobol indices (global, variance-based): A scrambled Sobol' sequence (base $N = 256$) decomposes the variance of the selected mass into a first-order index S_i and a total-order index S_{Ti} that adds its interactions (Figure 8.24),

$$S_i = \frac{\mathbb{E}[f_B(f_{AB_i} - f_A)]}{V}, \quad (8.33)$$

$$S_{Ti} = \frac{\frac{1}{2} \mathbb{E}[(f_A - f_{AB_i})^2]}{V}.$$

with f the selected gear mass, V its variance, f_A, f_B evaluations on two independent sample matrices and f_{AB_i} the B value for input i only. The ranking confirms and extends the analysis of the tornado. The elastomer mass per load coefficient is still a significant contributor, however this it is now observed that the skid rail wall thickness is now the highest driver. and the honeycomb, crush fraction and stroke envelope inputs identically zero.

Monte-Carlo selection robustness (Latin Hypercube): To test this fully, the uncertain space was filled with a 150 sample Latin Hypercube and checked which design the trade off selected in each case (Figure 8.25). The elastomeric block is feasible and selected in 100% of samples. At this landing mass the competing concepts rarely even reach a feasible point aside from the metal leaf, so the elastomeric block is not just the highest scoring design but effectively the only viable one. Notably, this plot changes per MTOW, so this is necessary to include within the convergence loop.

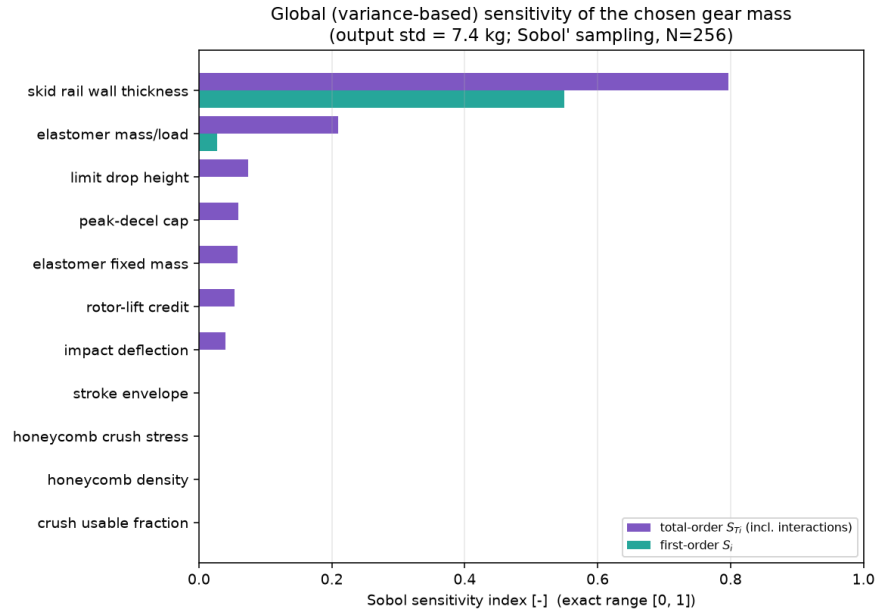


Figure 8.24: Global variance based (Sobol) sensitivity of the selected gear mass, first-order S_i and total order S_{Ti} per input. Inputs that govern non selected concepts contribute nothing.

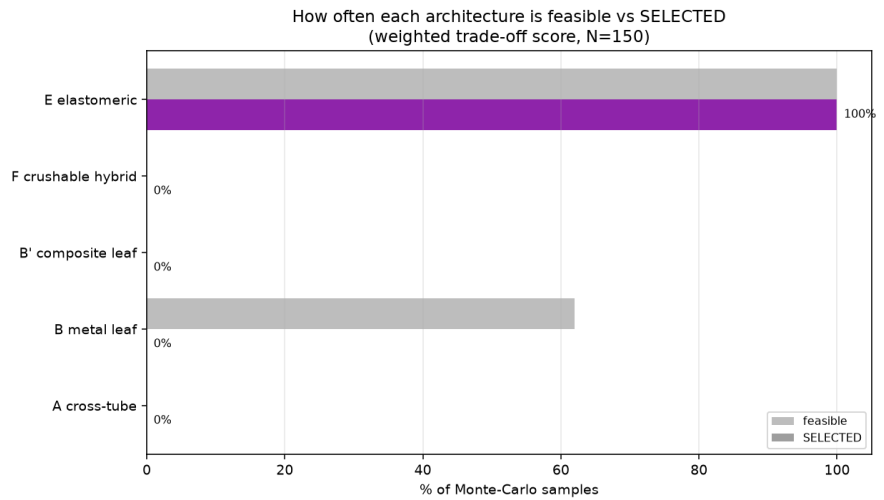


Figure 8.25: Selection robustness under input uncertainty (Latin Hypercube, $N = 150$): fraction of samples in which each design is feasible and in which it is selected.

Findings: The three methods agree, the dominant inputs set the required energy and effective mass (impact deflection, drop height), while absorber-specific inputs barely register because those architectures are rarely selected. The selection is stable, confirming the elastomeric is the best choice for the landing gear.

When integrating the landing gear into the landing_gear_mass(MTOW) code, it forms the effective landing mass from the take off mass, sizes all designs, scores them and returns the winning landing gear's mass. This goes into the outer loop that sums all component masses and iterates until the take off mass converges, which closes the feedback from the gear's own weight. Because mass is a dominant category, the same architecture (the elastomeric block) wins at every mass, thus the loop settles quickly in about 23 iterations.

8.6.9. Final Outcomes

Overall to assess the landing gear, the limit drop height is 0.25 [m], comfortably above the CS-27.725 floor (≥ 0.20 [m]) [54]. An assumed deflection of 0.15 [m] is used only to set the effective mass. No lift credit is taken, so the gear conservatively reacts the full weight. The peak deceleration is capped at 20 g for occupants and airframe and the available stroke envelope is 0.40 [m]. The drop height, assumed deflection, deceleration cap and stroke envelope are all placeholders awaiting firm requirements. The material used throughout the landing gear have the following characteris-

tics: Aluminium: modulus 71.7 [GPa], yield 503 [MPa], density 2810 [kg/m]³, ultimate strain 0.11.² Steel: 200 [GPa], 1200 [MPa], 7850 [kg/m]³, 0.06.³ CFRP: 70 [GPa], 600 [MPa], 1600 [kg/m]³, ultimate strain 0.015, with a fracture toughness of 1500 [J/m]².⁴

In the end the gear comprises of two skid rails, two transverse cross tubes (front and rear) and four energy-absorbing units (one per leg), as seen in Figure 8.21a. The bending architectures use two springs and two knees in both the cross tube and leaf members, giving four plastic hinges in total. The constants used in the sizing model are summarised in Table 8.16.

Table 8.16: Landing gear geometry and model constants.

Category	Parameter	Symbol	Value
<i>Bending model</i>	Cantilever stiffness coefficient	—	3.0
	Plastic hinge length (tube knee)	ℓ_h	1.0 D
	Plastic hinge length (leaf)	ℓ_h	1.5 t
	Developed leaf length factor	η_{dev}	2.0 $\times$ straight length
<i>Geometry</i>	Cross-tube span under fuselage	ℓ_{cross}	0.60 [m]
	Leg length	L	0.45 [m]
	Fuselage width	—	2.0 m
	Footprint cap	—	6.0 [m]
	Rail length	—	65% of fuselage length
<i>Hybrid honeycomb core</i>	Plateau (crush) stress	σ_{hc}	2.5 [MPa]
	Core density	ρ_{hc}	80 [kg/m] ³
	Usable stroke before densification	η_{stroke}	75%
	Hybrid leaf size ($t \times L$)	—	0.08 $\times$ 0.45 [m]
<i>Elastomeric mount</i>	Fixed bracket mass	m_{mount}	1.2 [kg]
	Load-dependent mass coefficient	c_{load}	1.0 $\times 10^{-4}$ [kg/N]
<i>Sizing setup</i>	Bound-calibration reference mass	—	700 [kg]
	SLSQP restarts	—	48

8.6.10. Limitations and Placeholders

Certain limitations that exist throughout the landing gear sizing, include that the bending plastic hinge energy and the composite fracture energy rely on simple capacity estimates. To improve, checking them against detailed FE or test data can be completed. In addition to this, tracks come out around 2.9–3.1 [m] against the 6 [m] cap and mainly guards against wider future designs, the footprint of the landing gear could also be widened to help with the tip-over stability. Finally, all masses are geometric ($m = \rho A \ell$) and exclude fasteners and fittings other than the primary load path so a primary mass fraction factor could be added later.

8.7. Cabin Pressurization

The aircraft cruises at a maximum altitude of 12,500 [ft] (3810 [m]). This is at the supplemental-oxygen limit in FAR 91.211 [55], so two options are possible: carry oxygen equipment or install a pressurized cabin. This section estimates the mass penalty of the pressurized option.

The cabin geometry as shown in Figure 3.3 is used for the sizing of the pressurized cabin. The required skin thickness for hoop-stress is: $t = SF \Delta p r / \sigma_w$, where the safety factor SF is 2. Δp is the cabin pressure differential, r is the cabin radius, and σ_w is the allowable skin stress.

System sizing For a 6,000 [ft] cabin at 12,500 [ft] cruise altitude, the pressure differential is only $\Delta p \approx 18$ [kPa]. With a cabin radius of 0.8 [m], the hoop-stress calculation gives a required skin thickness of about 0.3 [mm]. This is far below the existing 2 [mm] minimum-gauge fuselage skin, so no extra shell thickness is required. The mass penalty therefore comes from the pressurization hardware, seals, and ducting. An estimation for the pressurization mass breakdown can be seen in Table 8.17:

²7075-T6 aluminium, per the ASM/MatWeb datasheet (modulus 71.7 [GPa], yield 503 [MPa], density 2.81 [g/cm]³, elongation 11%): <https://asm.matweb.com/search/SpecificMaterial.asp?bassnum=ma7075t6>.

³Quenched-and-tempered AISI 4340 alloy steel, the conventional aerospace landing-gear steel: modulus 190 [GPa], yield ≈ 1150 [MPa], density 7.8 [g/cm]³ per <https://www.makeitfrom.com/material-properties/Quenched-and-Tempered-4340-Ni-Cr-Mo-Steel>.

⁴Quasi-isotropic carbon-fibre/epoxy laminate values, consistent with published T700-class laminate data ($\rho \approx 1500$ [kg/m]³): <https://www.ncbi.nlm.nih.gov/pmc/articles/PMC9229530/>. The interlaminar fracture toughness (1500 [J/m]²) is in line with mode-II (G_{IIc}) values for toughened systems reported in <https://www.sciencedirect.com/science/article/abs/pii/S0266353897002248>.

Table 8.17: Pressurization mass breakdown for 4 passengers and 1 crew member, 12,500 [ft] cruise altitude, and 6,000 [ft] cabin altitude.

Item	Mass [kg]	Driver
Shell thickness increment	0.0	required 0.3 [mm] $\ll$ 2 [mm] minimum gauge
Door/window seals and doublers	6.2	2 doors and 4 windows
Electric compressor	2.0	minimum practical unit size
Outflow valve, relief valve, and controller	3.6	fixed hardware
Distribution ducting	2.7	4.5 [m] cabin run
Direct pressurization mass	14.5	0.6% MTOW
Battery equivalent of compressor energy	0.7	compressor energy
Effective total	15.2	

Design implications Although the mass penalty is small, pressurization affects the layout. The cabin should remain close to circular, since flat panels would carry pressure in bending and become much heavier. Doors and windows also need seals, local reinforcement, and fatigue-resistant details.

An unpressurized cabin with oxygen masks remains a possible alternative. It would remove most of the 15.2 [kg] pressurization penalty and should be compared again before the configuration is frozen.

8.8. Fuselage Structural Sizing and Mass Estimation

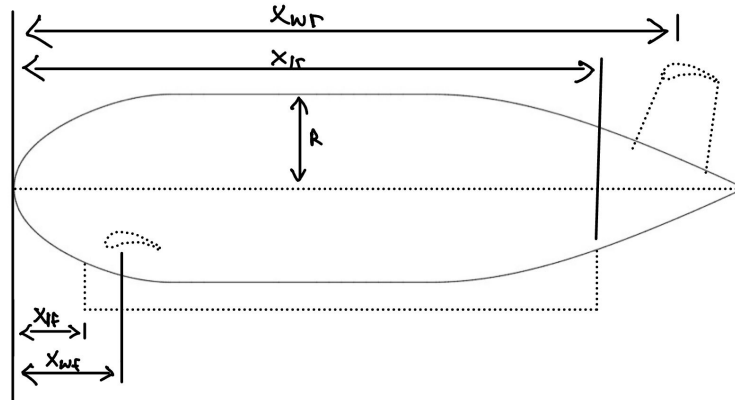
This section presents the preliminary structural sizing of the fuselage. For this the fuselage bending loads are estimated, the governing load cases are identified, and an estimate will be made for the skin thickness and structural mass.

The fuselage is sized using beam idealisation. The external loads are converted to vertical forces acting at the fuselage centreline, after which the support reactions, the shear force distribution, and the bending moment distribution are determined for several critical load cases. The maximum bending moment of the combined load cases is then used for the skin sizing.

8.8.1. Method and Beam Idealisation

The fuselage is modelled as a slender beam along x measured from the nose. The fuselage length is denoted by L_F , and the front and rear wing attachment locations are denoted by x_{wf} and x_{wr} respectively. These wing attachment points carry the main vertical reactions during flight, while during landing the support locations are taken at the landing gear contact points x_{lf} and x_{lr} .

The fuselage cross section is approximated as a thin-walled circular shell with radius R , and local structural details such as frames, stringers, and windows are not sized with this method. Figure 8.26 shows the fuselage idealisation used in the analysis, and has all the relevant dimensions needed for the method.

**Figure 8.26:** Fuselage side view with definition of main dimensions

The fuselage-borne mass is defined as a fraction of the total aircraft mass:

$$m_{fb} = f_{fb} m \quad (8.34)$$

where m is the aircraft mass and f_{fb} is the fraction of the aircraft mass carried by the fuselage. The fuselage-borne mass is then further split into payload mass m_{pl} and remaining fuselage mass m_{unif} , where the remaining fuselage mass is assumed to be uniformly distributed along the fuselage length. The payload mass is assumed to be distributed over the cabin box length L_{box} . For a load factor n_z , the corresponding distributed loads are:

$$w_{unif} = \frac{m_{unif} n_z g}{L_F} \quad (8.35)$$

$$w_{pl} = \frac{m_{pl} n_z g}{L_{box}} \quad (8.36)$$

from which it follows that the total distributed load is:

$$w(x) = \begin{cases} w_{unif} + w_{pl}, & x_{pl} - \frac{L_{box}}{2} \leq x \leq x_{pl} + \frac{L_{box}}{2} \\ w_{unif}, & \text{otherwise} \end{cases} \quad (8.37)$$

To solve the vertical support reactions for every load case the distributed load and its line of action need to be computed, which is done using the following equations:

$$W = \int_0^{L_F} w(x) dx \quad (8.38)$$

$$\bar{x}_w = \frac{1}{W} \int_0^{L_F} x w(x) dx \quad (8.39)$$

8.8.2. Load Cases and Internal Load Diagrams

Three load cases cover the main vertical bending conditions: symmetric manoeuvre, max-tail-load dive, and landing.

1. **Symmetric manoeuvre** - fuselage-borne mass at load factor n_W , reacted by the front and rear wing attachments; loads mainly the fuselage between the wings.
2. **Dive with max tail load** - inertia loads plus a downward tail load at the tail. Dive speed is Equation 8.40; with dynamic pressure and tail dihedral and the vertical tail force (S_t one panel area, C_N normal-force coefficient, Λ dihedral).
3. **Hard landing (VTOL)** - inertia reacted through the gear while wing-borne mass $m_w b = (1 - f_{fb})$ [m] acts at the wing attachments via F_{wf} , F_{wr} , split by rear wing lift fraction f_r .

$$V_D = k_d V_C, \quad F_t = 2q_D S_t C_N \cos \Lambda \quad (8.40)$$

Figure 8.27 shows the free-body diagrams of the three load cases.

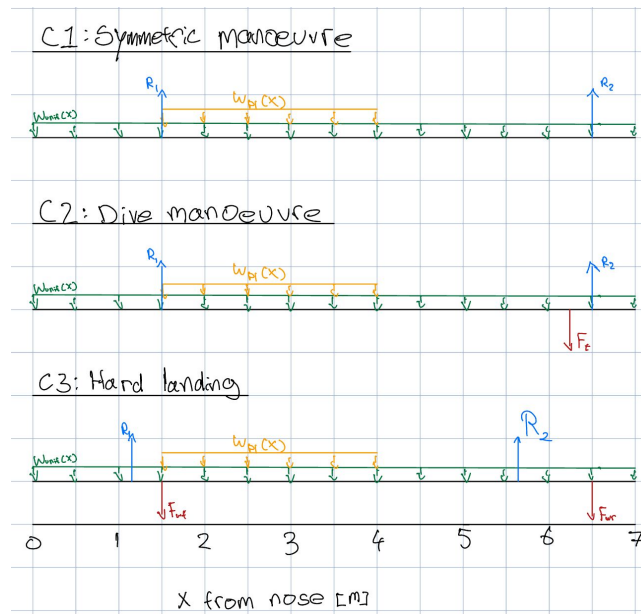


Figure 8.27: FBD's for the 3 load-cases

For each load case, the support reactions are found from the vertical force and moment equilibria, and are denoted by R_1 and R_2 with corresponding locations x_{s1} and x_{s2} . After the reactions are known, the internal shear force and bending moment are calculated along the fuselage. The shear force as a function of multiple known upward-positive forces $F_k^\uparrow$ at locations x_k is defined as:

$$V(x) = \sum_{x_k \leq x} F_k^\uparrow - \int_0^x w(\xi) d\xi, \quad (8.41)$$

where the bending moment is then obtained by integrating the shear force. The moment envelope is obtained by taking the maximum absolute bending moment over all load cases:

$$M_{env}(x) = \max_{c \in \{C1, C2, C3\}} |M_c(x)| \quad (8.42)$$

Applying a final ultimate safety factor j to this maximum bending moment gives us the critical load case which should be designed for and the critical fuselage x location. The moment envelope, with and without the safety factor, and its decomposition over the three load cases can be seen in Figure 8.28:

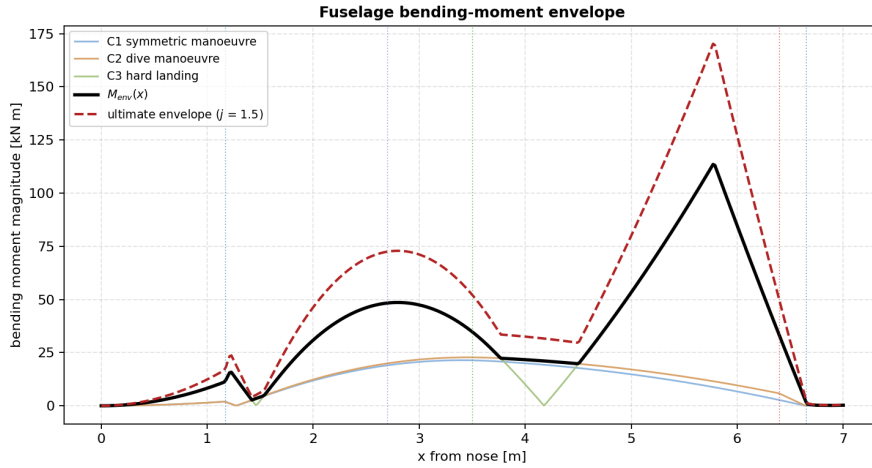


Figure 8.28: Moment envelope for the three load cases

From inspection it can be concluded that the hard landing is the most critical loading case for most of the fuselage length.

8.8.3. Preliminary Skin Sizing

The fuselage skin is sized using the maximum ultimate bending moment M_{ult} obtained in the previous section. The skin is sized using three criteria: yielding, buckling, and minimum gauge. The final thickness is then selected as the maximum thickness required by these criteria.

The first criterion is yielding. The bending stress in the fuselage skin is computed from the ultimate bending moment as:

$$\sigma_b = \frac{M_{ult} R}{I} \quad (8.43)$$

where R is the fuselage radius and I is the second moment of area of the thin-walled shell which is approximated by $I = \pi R^3 t$. The yield criterion requires that the bending stress remains below the allowable yield stress of the material: $\sigma_b \leq \sigma_{y,allow}$. This gives the yield driven skin thickness t_{yield} . Since the material has not been chosen yet, the allowable yield stress is kept as an input at this stage.

The second criterion is buckling. For this a buckling thickness $t_{buckling}$ is computed by requiring that compressive bending stress remains below the critical buckling stress: $\sigma_b \leq \sigma_{cr}$, where the critical buckling stress is computed with:

$$\sigma_{cr} = \frac{E}{\sqrt{3(1-\nu^2)}} \frac{t}{R} \quad (8.44)$$

where E is the Young's modulus, ν is Poisson's ratio, and R is the fuselage radius. Solving the equations for the minimum thickness t that satisfies the inequality gives the buckling driven thickness $t_{buckling}$.

The third criterion is the minimum gauge thickness. This is not a load driven thickness, but a minimum practical skin thickness below which the thickness is not allowed to go. It is included to account for practical structural and manufacturing considerations, and its value is 1 mm.

The yield and buckling criteria are evaluated at every fuselage x -location rather than only at the location of the maximum bending moment. Although the chosen skin thickness doesn't change, this is done to show the variations of the required thicknesses over the fuselage length. The required skin thickness at each x -location is simply obtained by taking the maximum of the 3 thicknesses: $t_{skin}(x) = \max(t_{yield}(x), t_{buckling}(x), t_{gauge})$, and its variation along the fuselage is visually shown in Figure 8.29. This shows that the buckling criterion governs the mid body region, while the nose and tail are governed by the minimum gauge thickness.

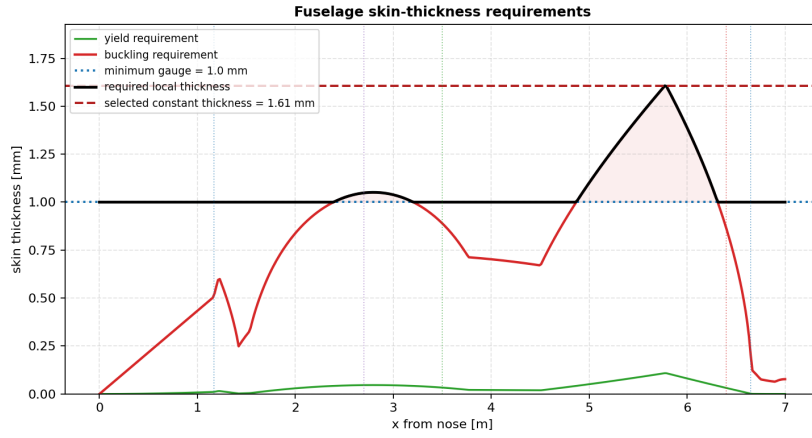


Figure 8.29: Required fuselage skin thickness for every x -location

For the current preliminary design, a constant thickness is selected for the full fuselage length rather than varying the thickness along the length, as it is simpler to manufacture and easier to inspect. The chosen value is the maximum shown in Figure 8.29, which corresponds to a thickness of 1.61 mm.

In a later design iteration, a variable skin thickness could be used to reduce structural mass. This would lead to a more mass efficient fuselage, but requires a more detailed manufacturing assessment.

The fuselage mass is computed from the constant skin thickness:

$$m_{skin} = 2\pi R L_F t_{skin,selected} \rho_{mat} \quad (8.45)$$

where ρ_{mat} is the material density, which will be determined in the next section.

8.8.4. Material Selection and Limitations

After the skin sizing method has been defined, the fuselage material selection is performed. The material affects the yield driven thickness t_{yield} through the allowable stress, and the buckling driven thickness $t_{buckling}$ through the elastic stiffness. It also drives the final fuselage skin mass through the density.

CFRP is selected for the fuselage skin, due to its high specific strength and stiffness. Compared to aluminium, CFRP has a lower density, while still having good strength and stiffness properties. As weight is a crucial driver for the design process, CFRP is considered the best option for the fuselage structure.

The final fuselage skin mass is calculated using Equation 8.45. The current estimate only includes the idealised fuselage skin, and neglects local reinforcements, frames, joints, stringers, and cut outs. These effects should be included in a later detailed structure design.

8.9. Final Remarks and Discussion

The structural analysis provided in this section was more robust than in previous iterations of the design, and represents the transition from empirical estimates to physics-based methods that are closer to the real conditions the aircraft will be under. However, further analysis is required in order to ensure the configuration chosen can truly fulfil the requirements.

- Due to time constraints, fatigue analysis on the structural components was not studied in great detail. This assessment is critical for fulfilling REQ-STK-14, as the effects of time on the structure remain largely unknown
- Failure modes from connection points such as bolts, welds or rivets, are often failure points in aircraft structures that cannot currently be assessed due to the insufficient level of detail in the current design and manufacturing process. In future models involved in detailed subsystem design, the effects of connection points should be accounted for.
- The fuselage modelling could also be improved by considering the wing primary fraction, as only skin is sized and even monocoque fuselage have additional structural elements.
- The landing gear analysis could be further improved by simulating the compression and rebound of the landing gear using the drop conditions, as opposed to a purely energy based approach.
- The fidelity of the loads on the wing could be improved, as currently the lift aerodynamic center is assumed to remain stationary during transition, when in reality this is not the case. Furthermore, a better estimate for the velocity incident to the wing must be found in order to assess exactly when the wings begin to produce lift, as currently it is assumed at a hinge angle of $\theta = 30$ [°]. Finally, the effects of the winglet hinge must be accounted for in order to size it properly.

9 Code Hierarchy and Optimisation

This chapter provides an overview of the structure of the code optimizer. Initially in Section 9.1 the general layout and methodology of the code is provided. In Section 9.2 the convergence method is described in order to give the final MTOW value. Next in Section 9.3 a few adjustments which were made during finalizing the code. Afterwards in Section 9.4 outlines the methodology of the optimizer consisting of the methods to tune certain parameters and receive the final results. Finally all outcomes of the optimizer are provided in Section 9.5. Finally, Section 9.6 discusses the limitations and possible future improvements of the optimisation procedure.

9.1. Sizing, Stability and Optimisation Code

This section describes the Python code structure that sizes the aircraft, evaluates its stability, and tunes selected design variables. The code is structured around a central input file containing all design variables, which is read by the remaining modules. To better visualize the overall structure Figure 9.1 shows how the pieces of the code work together at run time.

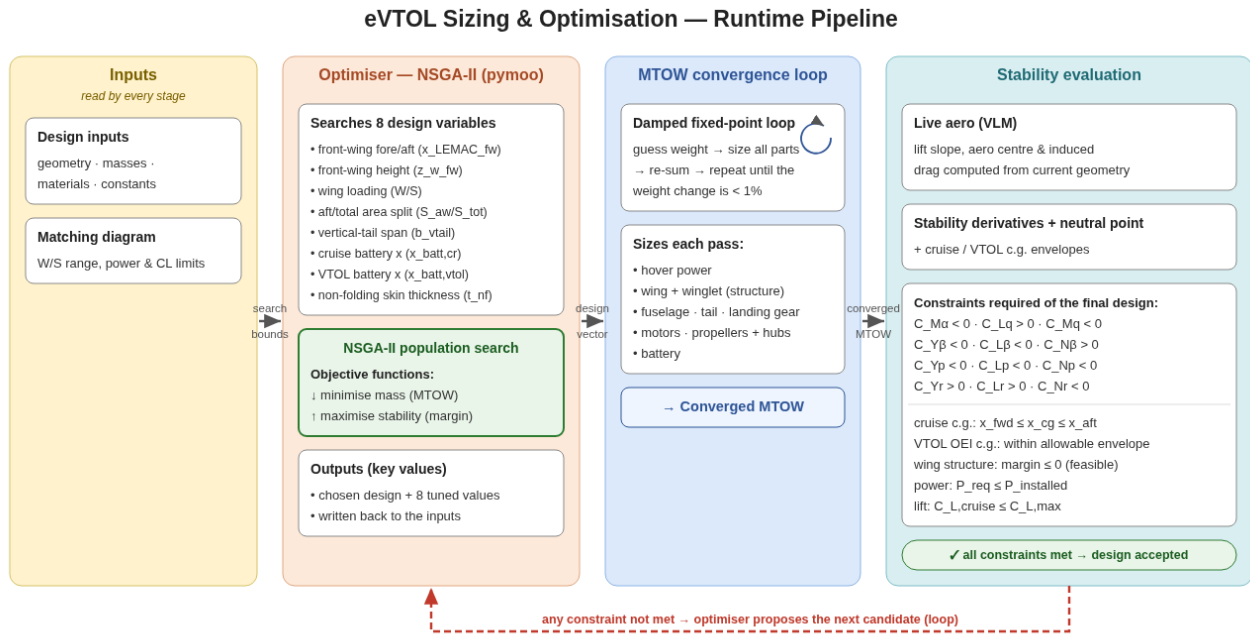


Figure 9.1: Broad Code Overview

Read from left to right, the diagram consists of four stages: Inputs, Optimiser, MTOW convergence loop, and Stability evaluation. The solid black arrows are the forward flow of one design evaluation. The dashed red arrow at the bottom is the feedback that closes the optimisation loop. The left hand column is the data that every other stage reads. The design inputs (geometry, masses, materials and physical constants) and the matching diagram, supply the allowable wing-loading (W/S) range together with the installed power and cruise lift feasibility limits.

The optimiser proposes candidate aircraft. It searches eight design variables: the front wing fore/aft position ($x_{LEMAC_{fw}}$), the front wing height ($z_{w_{fw}}$), the wing loading (W/S), the aft/total wing-area split (S_{aw}/S_{tot}), the vertical tail span ($b_{v_{tail}}$), the cruise and VTOL emergency battery positions, and the non folding wingbox skin thickness (t_{nf}). It pursues two objective functions at once: minimise mass (the converged MTOW) and maximise stability (the worst stability margin). Because these two goals compete, the result is a set of best trade off designs. The optimiser reports the chosen design and its eight tuned values, which are written back into the inputs so the rest of the code uses the tuned aircraft.

For each candidate the aircraft must first be sized. Component masses depend on the take off weight, but the take off weight is the sum of those same masses, so the loop solves this circular dependency by a damped fixed point iteration. It guesses a weight, sizes every part, adds them up, and moves the guess only part way toward the new total, repeating until the weight changes by less than 1%. Each pass sizes the hover power, the wing and winglet structure, the fuselage, tail and landing gear, the motors, propellers and hubs, and the battery, yielding the converged MTOW.

The converged design is then checked. The aerodynamic coefficients (lift curve slope, aerodynamic centre and induced drag) are recomputed live from the current geometry using the vortex lattice method, the stability derivatives, neutral point and centre of gravity (c.g.) envelopes are evaluated, and the design is tested against every requirement it must satisfy.

The codebase is built around a single parameter sheet `parameters.py` that acts as the main source for the rest of the sheets. Every geometry value, mass, material property and physical constant is defined there, and every other module imports its values from it. This keeps any given number from being written in two places and drifting out of sync.

The remaining code is grouped by engineering task:

- `parameters.py` contains all geometry, masses, materials and constants.
- `class_II_sizing` involves mass and energy sizing.
 - `mtow_sizing.py` houses the Maximum Take-Off Weight (MTOW) convergence loop.
 - `mass_components.py` contains one function per component mass (fuselage, wing, tail, motors, etc.), including the detailed structural wing sizing.
 - `energy.py` is responsible for mission energy and battery mass.
 - `MMOI.py`: the mass/position build-up that produces the centre of gravity (c.g.) and the mass moments of inertia.
- `final_characteristics`: aerodynamics, stability and optimisation.
 - `optimiser.py`: the NSGA-II optimiser.
 - `stability_eval.py`: the shared stability-evaluation core and the single list of stability requirements.
 - `stability_sensitivity.py`: the standalone sensitivity study.
 - `aero_model.py`: the Vortex Lattice aerodynamics.
 - `matching_diagram.py`: wing loading / thrust to weight feasibility bounds.
 - `structures/hinge_loading.py`: structures (wing planform, loads, winglet sizing).
 - `vehicle_dynamics`: the stability derivative model (`aircraft.py`) and the c.g. envelope checks.
- `support_files` and `fusion_geometry.py`: helper relations and CAD derived component masses.

9.2. MTOW Convergence Method

Sizing an aircraft involves a circular dependency. The mass of almost every component (wing, battery, motors, structure) depends on the total aircraft weight, but the total weight is the sum of all those component masses. The code resolves this with a damped fixed-point iteration.

The method works in two layers:

1. One pass (`compute_mtow`). Given a single guess for the MTOW, the function sizes every component once. It first sizes the hover powerplant from the guessed weight, then computes each component mass (fuselage, wing and winglet, landing gear, tail, motors, propellers, hubs, battery, payload, and miscellaneous systems), and finally sums them to produce a new MTOW estimate.
2. Repeat until stable `_solve_converged_mass`. The guess and the new estimate will not agree at first. The loop starts from the middle of an allowed weight band (1000–5500 [kg]) and, on every step, moves the guess only halfway toward the newly computed mass:

$$\text{MTOW}_{\text{next}} = \text{MTOW}_{\text{guess}} + \frac{1}{2}(\text{MTOW}_{\text{computed}} - \text{MTOW}_{\text{guess}}).$$

Taking only half a step prevents the estimate from overshooting and oscillating, and instead lets it settle smoothly. The loop stops once the guess and the computed value agree to within 1% (relative), which happens in roughly twenty iterations.

When the loop converges, the final design is cached in a module-level state. The cache holds the final MTOW and the complete mass breakdown. The rest of the program reads this cache rather than resizing the aircraft again. The converged per-wing planform is also written back into the wing geometry dataclass. This ensures that the aerodynamic, stability, and mass-moment-of-inertia models use the iteratively sized geometry rather than the initial hand-typed seeds.

9.3. Model Adjustments and Improvements

This section details the way in which previous discussed are implemented in code, and relevant changes that were made.

9.3.1. Battery Sizing, Packing and Sliding Installation

The pack is installed in the underfloor bay, 0.25 [m] below the cabin floor, and can translate longitudinally between two stations. In normal operation (taxi, vertical take-off, transition and cruise) it is locked at its forward cruise station, $x_{\text{batt,cr}} = 1.53$ [m] from the nose, which trims the cruise c.g. In vertical flight the c.g. requirement is set by the one-engine-inoperative (OEI) case: with a lift rotor failed, the asymmetric thrust balance demands a markedly more aft c.g., which the heavy (~ 540) [kg] battery provides when slid to its aft *VTOL-OEI station*, $x_{\text{batt,vtol}} = 5.73$ [m]. This brings the folded-configuration c.g. into the controllable VTOL-OEI envelope Subsection 6.1.2.

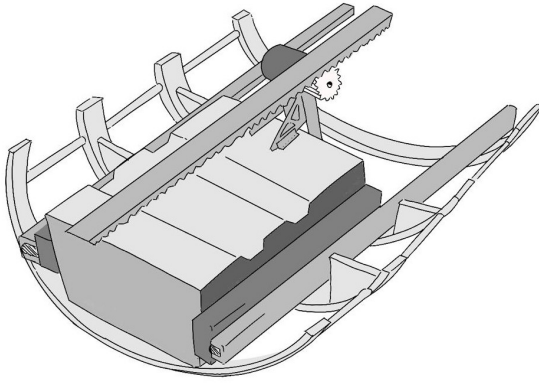


Figure 9.2: Rack and pinion based sliding mechanism

Mechanically, the slab rides on two longitudinal linear guide rails and is driven by a single electromechanical actuator. For this two actuators were considered: a high-reduction ball screw or a rack and pinion. It is also ideal to have mechanical lock pins capture the slab at each end station so the guide rails carry the sustained inertial and crash loads. Because the aft deployment is a vertical-flight/emergency function rather than a continuous control input, a low failure rate and an actuator with high potential stroke speed is desirable. Based on the information given, the rack and pinion is the preferred actuator choice because it has lower mass and is able to move fast.¹ This is critical because the c.g. shift from battery movement is only relevant in the OEI VTOL case, and there it needs to act fast to stabilise the aircraft. Figure 9.2 represents a conceptual sketch of how this system would operate in the underfloor section of the fuselage.

9.3.2. Battery Box Dimensioning for Minimum Sliding Length

The sliding installation of Subsection 9.3.1 only works if the pack can translate far enough aft to move the folded configuration centre of gravity into the controllable VTOL one engine inoperative (OEI) envelope. Because the pack must remain inside the airframe, the rearmost station its centre can reach is set by where the slab's aft face meets the fuselage tail:

$$x_{\text{batt,max}} = L_{\text{fus}} - \frac{1}{2}L_{\text{box}}, \quad (9.1)$$

which increases monotonically as the box is made shorter in the fore-aft (x) direction. The aft c.g. authority available from the battery is therefore maximised by minimising the slide direction length L_{box} for the given pack volume. This drives the box dimensioning.

The box volume is fixed by the cell count and the volumetric packing factor, and using the values determined in Subsection 9.3.1, the volume required by the battery pack is:

$$V_{\text{box}} = \frac{N V_{\text{cell}}}{\eta_{\text{vol}}} = \frac{3995 \times 41.778}{0.72} = 0.232[\text{m}^3] \quad (9.2)$$

As shown in Figure 3.3, the battery area below the cabin floor is equivalent to $0.27 [\text{m}^2]$ with a largest cross section having a length of $0.7 [\text{m}]$. Now the only remaining unknown for the battery is its length. Dividing the initial found volume by the curved area available, the length of the battery ends up being $0.86 [\text{m}]$. When the battery pack is slid to the most aft c.g. position, the diameter of the fuselage is equivalent to $0.8 [\text{m}]$, meaning that the battery pack is able to fit, and hence Vold-e is OEI stable during hover.

9.3.3. Wing Loading as a Search Variable

In the original sizing the wing loading was a single fixed number, $W/S = 760 \text{ N/m}^2$, read off once from an offline matching diagram built on hardcoded inputs together with the qualitative rule that the lightest aircraft sits in the top right corner of the feasible region. To make W/S a genuine design variable the matching diagram was rebuilt as an iterative module that recomputes every constraint from the converged design state and from a single parameter source, with no qualitative point picking. Instead of returning one operating point it exposes the feasible interval of W/S together with the governing power relations, so the optimiser can sweep the wing loading quantitatively. Each candidate W/S sets the reference area through

$$S = \frac{W}{W/S}, \quad (9.3)$$

which propagates into the wing mass and back into the converged W , and is routed in parallel into the stability model so the planform used for the static margins stays consistent with the planform used for the mass. The wing loading is quantised onto a coarse grid limited by bounds so that the weight reconvergence at each distinct W/S can be cached.

9.3.4. Matching Diagram Changes

The matching diagram plots the power loading W/P against the wing loading W/S and overlays the flight conditions the aircraft must satisfy. For this vehicle the main conditions are stall, cruise, rate of climb, and climb gradient; the conventional takeoff and landing run constraints are absent because a vertical takeoff and landing aircraft performs

¹ How to Choose the Best Linear Actuator for your Application, Teknic, Inc. <https://teknic.com/choose-a-linear-actuator/> (accessed 16 June 2026).

these phases on its rotors rather than on a runway. The stall condition contains no power term and so appears as a vertical line that constrains the wing loading,

$$\left(\frac{W}{S}\right)_{\text{stall}} = \frac{1}{2} \rho_0 V_{\text{stall}}^2 C_{L\text{-max}} \quad \left(\frac{W}{P}\right)_{\text{cruise}} = \eta_p \sigma^{3/4} \left[\frac{C_{D0} \frac{1}{2} \rho V^3}{W/S} + \frac{W}{\pi b^2 e \frac{1}{2} \rho V} \right]^{-1} \quad (9.4)$$

$$\left(\frac{W}{P}\right)_{\text{RoC}} = \eta_p \left[c + \frac{\sqrt{2/\rho_0}}{1.345 \left[\left(\frac{b^2}{W} \right) e \right]^{3/4} (W/S)^{1/4} / C_{D0}^{1/4}} \right]^{-1} \quad \left(\frac{W}{P}\right)_{\text{grad}} = \eta_p \left[\sqrt{W/S} \left(\frac{c}{V} + \frac{C_{D0}}{C_L} \right) \sqrt{\frac{2}{\rho_0} \frac{1}{C_L}} \right]^{-1} \quad (9.5)$$

Every power loading is referenced to sea level, because the installed power is fixed by the hover requirement and lapses with density, so the curves can be compared directly against the available power. The feasible region is the area lying below the lowest of the three power curves and to the left of the stall line. No assumption is made about which power condition binds. The boundary is taken as the pointwise minimum of the three.

The feature that makes this diagram look unlike a textbook one is the treatment of the aspect ratio. In a conventional matching diagram the aspect ratio is chosen independently and held constant, so the cruise and climb curves are evaluated at one fixed A and take their familiar shapes. Here the span is not free. The box wing footprint and the folding constraint fix the full span at $b = 13.0$ m, and once the span is fixed the reference area is no longer an independent quantity but is set entirely by the wing loading through $S = W/(W/S)$. The aspect ratio then follows as

$$A = \frac{b^2}{S} = \frac{b^2}{W} \left(\frac{W}{S} \right), \quad (9.6)$$

so for a fixed weight and span the aspect ratio rises linearly with the wing loading. The horizontal axis of the diagram is therefore doing three jobs at once: it is a wing loading axis, an inverse wing area axis, and a proportional aspect ratio axis. A point far to the right is not only a heavily loaded small wing but also a high aspect ratio wing.

This coupling is what reshapes the curves. In the constant aspect ratio diagram the only explicit W/S dependence in the cruise relation is the parasite term growing as $1/(W/S)$ and the induced term growing as W/S . Once $A = A(W/S)$ is substituted, the induced term becomes proportional to $(W/S)/(\pi A e)$ with A itself proportional to W/S , so the leading wing loading dependence in that term cancels and the penalty no longer climbs in the usual way toward high wing loading. The climb relations carry the same effect through their $(Ae)^{3/4}$ and $C_D = C_{D0} + C_L^2/(\pi A e)$ factors, which now stiffen as the wing loading increases. The Oswald factor is taken from the Prandtl box wing relation and depends only on the fixed span and vertical gap, so it stays constant across the sweep and all of the variation in the induced terms comes from $A(W/S)$ alone. The visible result is constraint curves whose slope and curvature differ from the standard forms, and a feasible region whose right hand side is governed almost entirely by the vertical stall line at 765.6 N/m^2 rather than by a power curve cutting across it, because the growth of aspect ratio toward high wing loading keeps the power conditions from becoming the binding limit. This is reflected in the following diagram:

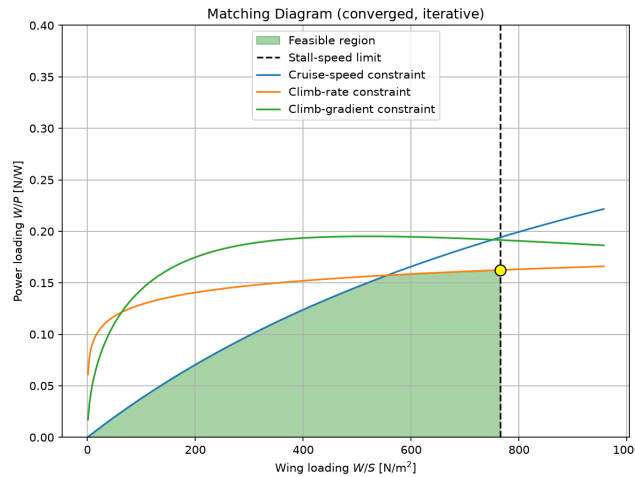


Figure 9.3: Updated 'matching' diagram in the case of non constant aspect ratio

9.3.5. Vortex Lattice Aerodynamics (VLM) Integration

For the optimiser to be meaningful, the aerodynamic coefficients have to react when the wing geometry changes. Early in the project these coefficients were fixed numbers taken from a one off XFLR5 analysis, so they did not respond to changes in wing loading, aspect ratio, taper or sweep. They were replaced with a live Vortex Lattice Method (VLM) model. The model is built on AeroSandbox, which provides the VLM solver and a standard atmosphere (giving the cruise Mach and Reynolds number), together with NeuralFoil, a fast neural network surrogate that supplies the 2-D airfoil section properties the VLM needs. The function used recomputes the front and aft wing lift curve slopes,

aerodynamic centre positions and zero lift moments from the current geometry and overwrites the old fixed values. The results are cached against a rounded description of the geometry, so an identical wing shape is never solved twice. This matters because the optimiser evaluates many candidate designs.

9.4. NSGA-II Optimiser Methodology

An optimisation layer sits above the sizing loop, tasked with selecting a small set of design variables that minimise aircraft mass while satisfying all stability constraints. It operates in two phases: Phase 1 evaluates whether the current design already meets the requirements, and Phase 2 invokes the optimisation algorithm if constraints are not satisfied. The chosen algorithm for this process was using NSGA-II. The motivation behind this was due to having two conflicting objectives (MTOW vs. worst normalised stability margin). Pareto trade-offs finding the best compromises was the necessary course of action as opposed to finding a single weighted optimum. Each evaluation is non-differentiable and without explicit formula's due to for example building VLM aerodynamics and a mass converger, which rules out gradient based methods.

9.4.1. Tuning Parameters and Bounds

Firstly, it was imperative to select the parameters that would be tuned within the design process. The eight optimiser variables were not chosen arbitrarily; they were selected using a dedicated sensitivity study. This study was run on its own prior to starting the optimisation process. The methods employed involved the well established method of performing a tornado analysis and computing sobol indices.

Tornado analysis: This aimed to provide a quick, readable ranking of which input shifts an output the most when swept from its low to its high value (this was used periodically when quick checks were needed, but does not capture all coupling).

Sobol indices (variance based global sensitivity): This is estimated with the Saltelli/Jansen sampling scheme. Rather than perturbing one input at a time, the method samples all inputs simultaneously across their ranges and decomposes the total variance of the output into the fraction contributed by each input. The first-order index measures an input's effect on its own, while the total-order index also captures the variance it produces through interactions with other inputs. Comparing the two reveals not just which inputs matter most, but how much of their influence is coupled to others.

The headline result is a heatmap of the total-order Sobol index for every (input $\times$ output) pair. Read at a glance, it shows which knob drives which requirement (for example, wing dihedral driving the roll stability derivative) and which knobs most strongly drive the MTOW. The variables that both (a) strongly influence the stability margins or the mass and (b) were still physically free to change were then pushed into the optimiser design variables; those with little influence were left fixed to keep the search small and fast.



Figure 9.4: Heatmap of the different design parameters

One important caveat to note from this is that despite some parameters having a strong effect (such as wing dihedral), because of the fact that the optimiser was applied while a configuration was already decided it meant that several influential design inputs had to be fixed or have more stringent bounding. The final design variable set therefore reflects what was genuinely still adjustable at that stage rather than the full set of influential parameters.

Table 9.1: NSGA-II tuned design variables, their search bounds, and role

Variable	Symbol	Bounds [unit]	Explanation
Front-wing LEMAC	$x_{\text{LEMAC},fw}$	[0.00, 1.50] m	Front-wing leading-edge MAC station; shifts the cruise c.g. envelope and the wing mass station (c.g.).
Cruise battery station	$x_{\text{batt},cr}$	[1.50, 3.04] m	Sliding-battery longitudinal position in cruise; balances the cruise c.g.
VTOL battery station	$x_{\text{batt},VTOL}$	[2.74, 6.50] m	Aft emergency battery station; seats the VTOL one-engine-inoperative c.g. in its envelope.
Front-wing height	$z_{w,fw}$	[-0.90, -0.60] m	Vertical position of the front wing (z up); drives the lateral-directional derivatives.
Vertical-tail span	b_{vt}	[1.40, 1.60] m	Sizes the vertical tail (and, coupled, its mass); trades directional stability against MTOW.
Wing loading	W/S	[459.4, 765.6] N/m ²	Sets the wing area $\Rightarrow$ wing mass and reference area; the live MTOW lever. Bounds: matching-diagram stall limit (upper), 0.6 $\times$ (lower).
Aft/total area split	S_{aw}/S_{tot}	[0.45, 0.65]	Front-vs-aft box-wing area share; sets per-wing structural mass and the aero wing split.
Non-folding thickness	t_{nf}	[1.0, 6.0] mm	Non-folding wingbox skin gauge; pure feasibility lever setting the torsional twist checked against the 1° limit.

The optimiser then varies the seven design variables selected, which are each bounded to by requirements of other departments, along with physical constraints as seen in Table 9.1. Note that the non-folding thickness parameter is not considered in the heatmap because it is one level down from MTOW, as it informs the wing mass. This variable was included as later in the design phase it was not clear which wing box thickness would be ideal. Adding this was also valuable as it was another MTOW driver.

9.4.2. Objective Functions

The optimiser is multi-objective in nature, so for a design vector $\mathbf{x}$ it minimises the pair

$$\mathbf{F}(\mathbf{x}) = (\text{MTOW}(\mathbf{x}), g_{\text{worst}}(\mathbf{x})),$$

In other words the first objective is the MTOW and the second is the comfortability at which the design meets stability and cg requirements, as it is not ideal for the margins on these to be near zero (marginal instability could occur or minor component manufacturing differences could be enough to tip really small margins).

Firstly, each stability/c.g. requirement r constrains some quantity $q_r(\mathbf{x})$ to an allowable band $[\ell_r, u_r]$. This is the convention used for the margin, which is a measure of how much slack the design has from the bound.

$$m_r(\mathbf{x}) = \max(\ell_r - q_r(\mathbf{x}), q_r(\mathbf{x}) - u_r), \quad \begin{cases} m_r \leq 0 & \text{requirement met, with } |m_r| \text{ of slack,} \\ m_r > 0 & \text{requirement violated.} \end{cases}$$

When considering stability derivatives as a requirement, it is observed that the raw margins are not comparable as aerodynamic derivatives span several orders of magnitude. For example C_{M_q} is an order of approximately $\sim \mathcal{O}(10^2)$ versus C_{Y_p} with $\sim \mathcal{O}(10^{-2})$. Additionally, c.g. limits are lengths in metres. Each margin is therefore divided by a fixed baseline scale of its magnitude at the unmodified reference design $\mathbf{x}_0$ because this captures the effect of some derivatives being affected in a limited way by the tuning parameters. (This is expressed by the following: $s_r = |m_r(\mathbf{x}_0)|$) This is computed once and cached at the beginning of the optimiser run and every requirement is from there is evaluated with respect to its baseline slack. The objective is then evaluated at the single worst case across all requirements:

$$g_{\text{worst}}(\mathbf{x}) = \max_r \frac{m_r(\mathbf{x})}{s_r}.$$

With this definition $g_{\text{worst}} \leq 0$ means every requirement is satisfied with slack as the maximum design is not above zero. Driving g_{worst} more negative yields a design that is robustly stable where it is actually under tension. This is also why a worst-margin objective is used instead of a summed constraint violation. A total violation metric flattens to zero once a design is feasible, so there would be no tension between the two objective functions any more. This continues to shape the Pareto front inside the feasible region.

In the same vein, it is also necessary to explain why the MTOW and $g_{\text{worst}}(\mathbf{x})$ have objectives that compete. The goals pull in opposite directions. Lowering MTOW means shrinking mass-driving geometry, raising W/S (a smaller wing), trimming the vertical tail, or skewing the area split. However those same moves erode stability. A smaller tail weakens the directional derivatives, a smaller or repositioned wing shifts the neutral point and c.g. envelopes, and the margins m_r tighten. Conversely, buying robustness (a larger tail, a more conservative split) adds structural and

system mass and pushes MTOW back up. Because no single design simultaneously minimises both, the result is not a unique optimum but a Pareto front. Its a set of non dominated compromises trading mass against stability, from which a representative design (the lightest with $g_{\text{worst}} \leq 0$) is selected.

9.4.3. Constraints

A design is accepted only when every constraint satisfies $g_i(\mathbf{x}) \leq 0$, the same sign convention used for the worst case margin in the previous subsection. The sixteen stability and centre of gravity requirements are exactly the requirements r whose normalised margins m_r/s_r were defined there. The objective collapses them into a single worst case g_{worst} , whereas as constraints each requirement is enforced on its own,

$$g_r(\mathbf{x}) = \frac{m_r(\mathbf{x})}{s_r} \leq 0.$$

Three further constraints act only as constraints and never enter the objective: two matching diagram limits and one wing structural limit. This gives nineteen constraints in total. The sixteen scaled requirements divide into twelve derivative sign requirements and four centre of gravity envelope requirements.

Firstly, the signs of static and dynamic stability derivatives were ascertained via standard aircraft dynamics theory. These fix the sign of each static and dynamic stability derivative so that every restoring and damping term acts in the stabilising direction. They constrain signs rather than magnitudes and the robustness objective then drives the binding ones further below zero. The twelve are listed in Table 6.1.

A further four requirements are related to gravity envelope requirements where the centre of gravity must lie inside its allowable longitudinal envelope in both the cruise configuration and the VTOL one engine inoperative (OEI) configuration. Here the constrained quantity q_r is written directly as a difference of stations measured from the nose datum, so the design dependent envelope limit sits inside the term and the bound stays a constant at zero:

$$g_{\text{cg,cr,fwd}} = x_{\text{cg,cr}}^{\text{fwd}} - x_{\text{cg,cr}} \leq 0, \quad g_{\text{cg,cr,aft}} = x_{\text{cg,cr}} - x_{\text{cg,cr}}^{\text{aft}} \leq 0, \quad (9.7)$$

$$g_{\text{cg,VTOL,fwd}} = x_{\text{cg,VTOL}}^{\text{min}} - x_{\text{cg,VTOL}} \leq 0, \quad g_{\text{cg,VTOL,aft}} = x_{\text{cg,VTOL}} - x_{\text{cg,VTOL}}^{\text{max}} \leq 0. \quad (9.8)$$

The cruise limits follow from the static margin and neutral point envelope, while the VTOL limits follow from the OEI control balance envelope. If that OEI envelope cannot be computed, a large penalty replaces its limits so that both VTOL constraints fail rather than the evaluation aborting. The forward VTOL constraint $g_{\text{cg,VTOL,fwd}}$ is the single binding constraint of the problem, which is why the initial population is seeded with the sliding battery biased toward its aft travel limit.

Matching diagram feasibility informs the two further constraints. These constraints carry the matching diagram into the search, already scaled to order one:

$$g_{\text{power}} = \frac{P_{\text{req}}(W/S) - P_{\text{inst}}}{P_{\text{inst}}} \leq 0, \quad g_{C_L} = \frac{C_{L,\text{cruise}}(W/S) - C_{L,\text{max}}^{\text{op}}}{C_{L,\text{max}}^{\text{op}}} \leq 0. \quad (9.9)$$

The first keeps the cruise and climb power demand below the hover sized installed power, and the second keeps the cruise lift coefficient below its operational maximum. Both stay deeply slack for this hover power dominated aircraft, but they keep the matching diagram enforced if a future design approaches the power or stall limits.

Lastly, Wing structural feasibility informs the final constraint. This constraint centres around having no spar bending overstress, and the non-folding wingbox torsional twist should be within its allowable θ_{allow} . It is emitted as a continuous wing margin,

$$g_{\text{wing}} = \max\left(\frac{|\theta_{\text{twist}}|}{\theta_{\text{allow}}} - 1, \begin{cases} 0 & \text{no bending overstress} \\ +1 & \text{bending overstress} \end{cases}\right) \leq 0, \quad (9.10)$$

so the twist term provides a continuous gradient in the wingbox thickness, while a bending overstress forces a positive, infeasible value.

9.5. Optimised Design Results

The multi-objective optimiser converges on a feasible Prandtl plane eVTOL with a maximum take-off mass (MTOW) of 1979.9 [kg], operating at a wing loading of $W/S = 765$ [N/m²] (the upper edge of the matching diagram feasible window [459, 766] [N/m²]) and an aft to total wing area split of $S_{\text{aft}}/S_{\text{tot}} = 0.6$. The converged design satisfies every static longitudinal stability and centre of gravity (c.g.) envelope requirement and is flagged accepted. The discussion below covers the mass distribution, the static and dynamic stability behaviour, and the resulting stability derivatives.

Figure 9.5 gives the converged MTOW breakdown. The battery is the single largest item at 540 [kg] (27.3% of MTOW), followed by the fixed mission payload at 400 [kg] (20.2%); together, energy and payload account for roughly 47.5% of the aircraft mass, which is characteristic of battery electric VTOL platforms. The primary structure totals 438 [kg] (22.1%), dominated by the CFRP box wing (198 [kg]) and the fuselage (127 [kg]); the relatively low structural

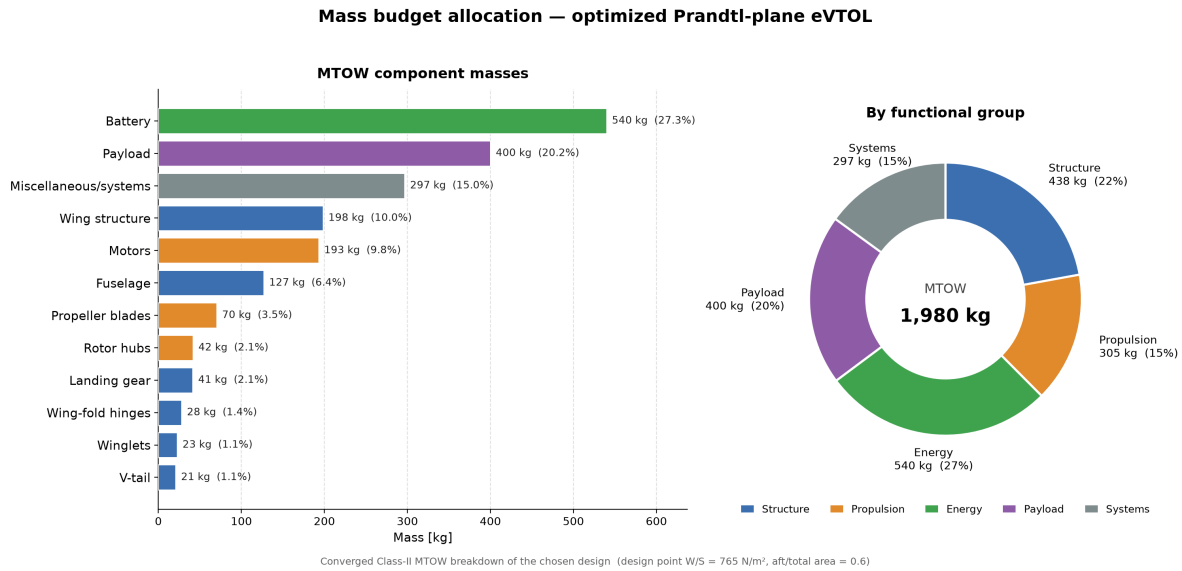


Figure 9.5: Mass budget allocation of the optimised design: per component MTOW contributions (left) and functional-group subtotals (right).

mass fraction reflects the composite, span shared box wing layout. Propulsion (motors, blades and hubs) contributes 305 [kg] (15.4%), with the electric motors the largest single propulsion item at 193 [kg], and the miscellaneous/systems group a further 297 [kg] (15.0%).

The neutral point sits 2.72 [m] aft of the nose, while the cruise c.g. lies at 2.64 [m], giving a static margin of 7.9% of the 1.06 [m] global mean aerodynamic chord, comfortably above the 5% minimum. The 5% static margin aft c.g. limit is at 2.67 [m], so the cruise c.g. sits only about 3 [cm] forward of it. This aft cruise c.g. constraint is the tightest of all sixteen requirement checks, indicating that the optimiser has driven the c.g. aft toward the stability boundary (minimising trim drag and tail size). Figure 9.6 shows the cruise static stability trends. The neutral point moves aft as the rear wing area fraction grows (a), the corresponding aft c.g. limit (b), the pitch stiffness derivative C_{M_α} remaining negative across the whole design envelope (c), and the combined cruise/VTOL-OEI feasible band (d).

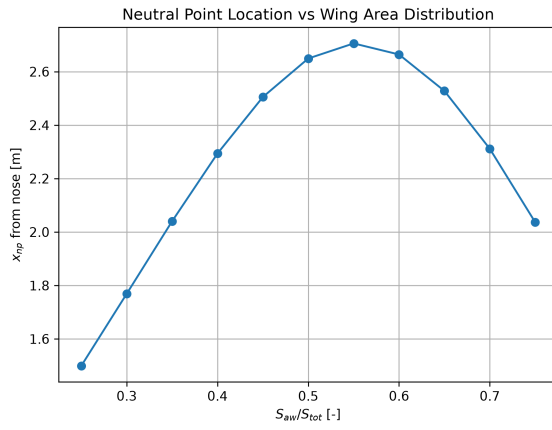
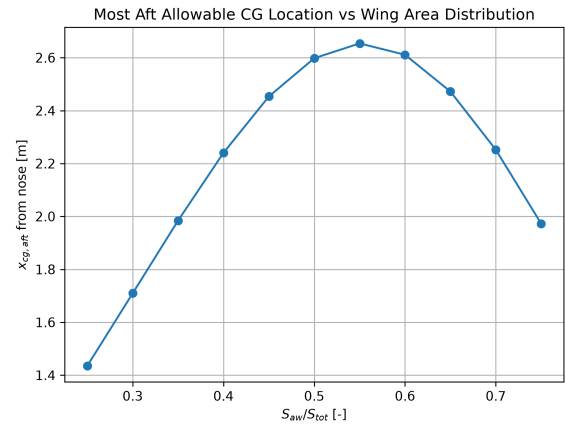
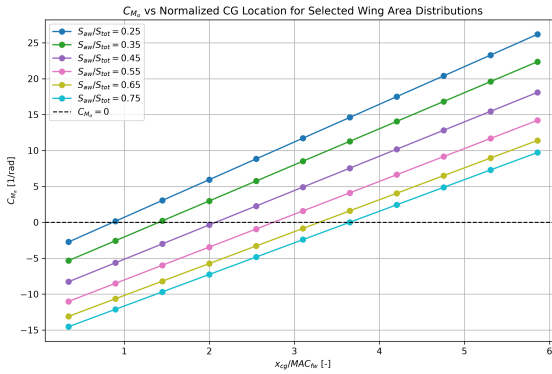
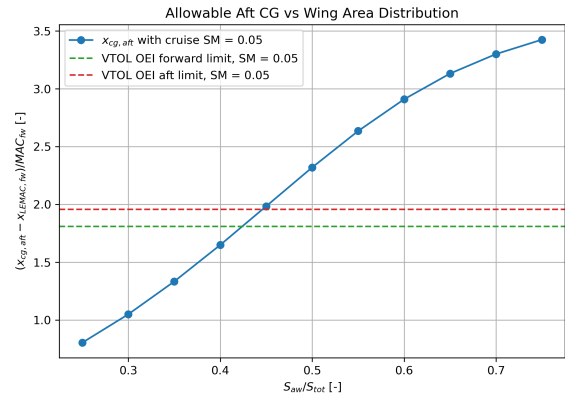
In the VTOL configuration the wings fold and the c.g. shifts aft to 4.53 [m]. A one engine inoperative (OEI) hover trim analysis over the six rotor layout sets an allowable VTOL c.g. band of 4.43–4.85 [m], inside which the folded wing c.g. comfortably lies. Figure 9.7 presents the VTOL OEI envelope, including the aft c.g. limit in both MAC normalised and nose datum coordinates. All twelve derivative sign checks and all four c.g. envelope checks (cruise fore/aft, VTOL fore/aft) pass.

Table 9.2 lists the principal stability and control derivatives of the converged design (per radian). The sign convention follows the standard body axis definitions, with rolling and yawing moment coefficients written C_ℓ and C_n to distinguish the rolling moment from lift C_L . Longitudinally, $C_{M_\alpha} = -6.12$ confirms strong static pitch stability, and the pitch rate derivatives $C_{M_q} = -104.3$ and $C_{L_q} = 13.3$ indicate very heavy pitch damping. Laterally and directionally, the design is stable on every count: $C_{Y_\beta} = -2.18$, a dihedral effect $C_{\ell_\beta} = -0.098$, weathercock stability $C_{n_\beta} = +0.144$ from the V-tail, roll damping $C_{\ell_p} = -0.386$ and yaw damping $C_{n_r} = -0.109$. Control authority is provided by $C_{M_{\delta_e}} = -3.01$ (elevator), $C_{n_{\delta_r}} = -0.041$ (rudder) and $C_{\ell_{\delta_a}} = 0.036$ (aileron). All twelve sign based requirement checks are met.

Table 9.2: Principal stability and control derivatives of the optimised design (body axes, per radian). “Req.” is the sign required for stability; “Met” indicates whether the requirement is satisfied.

Longitudinal				Lateral-directional				Control			
Derivative	Value	Req.	Met	Derivative	Value	Req.	Met	Derivative	Value	Req.	Met
C_{L_α}	6.07	> 0	yes	C_{Y_β}	-2.181	< 0	yes	$C_{M_{\delta_e}}$	-3.014	—	—
C_{M_α}	-6.12	< 0	yes	C_{ℓ_β}	-0.098	< 0	yes	$C_{L_{\delta_e}}$	0.800	—	—
C_{L_q}	13.28	> 0	yes	C_{n_β}	+0.144	> 0	yes	$C_{Y_{\delta_r}}$	0.141	—	—
C_{M_q}	-104.25	< 0	yes	C_{Y_p}	-0.009	< 0	yes	$C_{n_{\delta_r}}$	-0.041	—	—
$C_{M_{\dot{\alpha}}}$	-16.19	< 0	yes	C_{ℓ_p}	-0.386	< 0	yes	$C_{\ell_{\delta_a}}$	0.036	—	—
$C_{L_{\dot{\alpha}}}$	4.30	—	—	C_{n_p}	-0.047	< 0	yes				
				C_{Y_r}	+0.357	> 0	yes				
				C_{ℓ_r}	+0.135	> 0	yes				
				C_{n_r}	-0.109	< 0	yes				

The linearised state space model, built from the cruise mass/inertia properties (Table 9.3) and the derivatives

(a) Neutral point vs. S_{aft}/S_{tot} .(b) Aft c.g. limit (5% SM) vs. S_{aft}/S_{tot} .(c) C_{M_α} over the design envelope.

(d) Combined cruise/VTOL-OEI feasible c.g. band.

Figure 9.6: Cruise static longitudinal stability of the optimised design.

above, yields the eigenmotion characteristics summarised in Table 9.4; the corresponding time responses are shown in Figure 6.4. The short period mode is well damped ($\zeta = 0.89$, $\omega_n = 6.8$ [rad/s]), the Dutch roll mode meets all three handling quality bounds ($\zeta = 0.156$, $\omega_n = 2.21$ [rad/s]), and the roll subsidence mode is fast ($\tau = 0.41$ [s]). The spiral mode is slightly unstable and is marginally divergent (eigenvalue $+0.024$ [s $^{-1}$]) but its 29 [s] doubling time exceeds the 20 [s] requirement and is easily corrected. The only mode that does not meet its target is the phugoid. It is stable but very lightly damped ($\zeta = 0.011$, below the $\zeta \geq 0.04$ threshold), with a slow 29 [s] period. This is a slow, long period oscillation and is readily handled by the pilot or a basic stability augmentation system. Four of the five modes meet the handling quality targets.

Table 9.3: Cruise mass and inertia properties (body axes); VTOL values are for the folded-wing configuration.

Quantity	Cruise	VTOL
Mass [kg]	1979.9	1979.9
I_{xx} [kg m 2]	9087	9087
I_{yy} [kg m 2]	6642	9074
I_{zz} [kg m 2]	13647	16079
I_{xz} [kg m 2]	1937	736

Table 9.4: Dynamic eigenmotion characteristics in cruise. $T_{1/2}$ is time to half amplitude; T_2 time to double (spiral). All but the phugoid meet their handling-quality requirement.

Mode	Eigenvalue [1/s]	ζ	ω_n [rad/s]	Period/ T [s]	Status
Short period	$-6.047 \pm 3.047j$	0.893	6.77	$P = 2.1$	Pass
Phugoid	$-0.002 \pm 0.214j$	0.011	0.214	$P = 29.3$	Fail ($\zeta \geq 0.04$)
Dutch roll	$-0.345 \pm 2.178j$	0.156	2.21	$P = 2.9$	Pass
Spiral	+0.024	—	—	$T_2 = 29.0$	Pass ($T_2 \geq 20$)
Roll subsidence	-2.422	—	—	$\tau = 0.41$	Pass ($\tau \leq 1.4$)

9.5.1. Fuselage Geometry Iteration

The optimised design resulted in a new front wing position. The new leading edge of the MAC of the front wing is located at 0.17 [m] from the nose, while the new vertical front wing position is 0.86 [m] below the fuselage centreline.

As a result, the fuselage geometry was adapted to a low-nose configuration, as this improved geometrical compatibility with the new front wing location. A side-view of the new fuselage is shown in Figure 9.8.

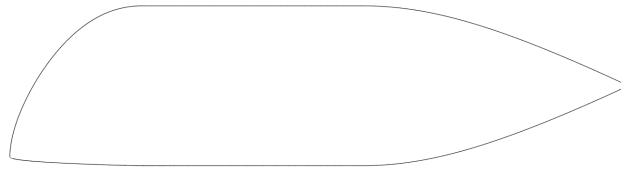
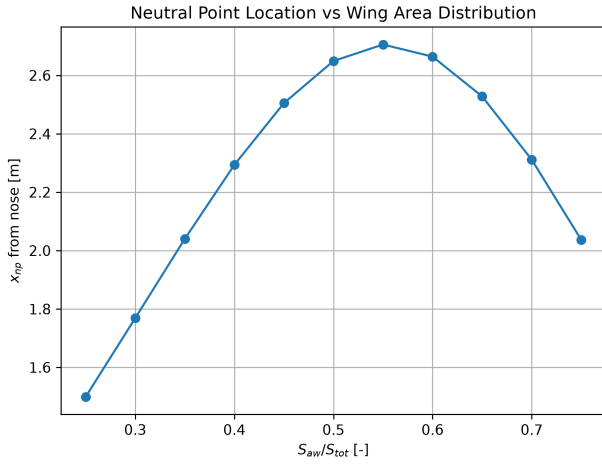
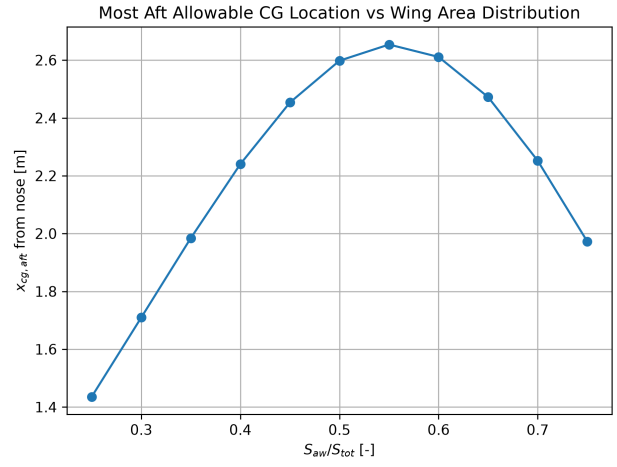
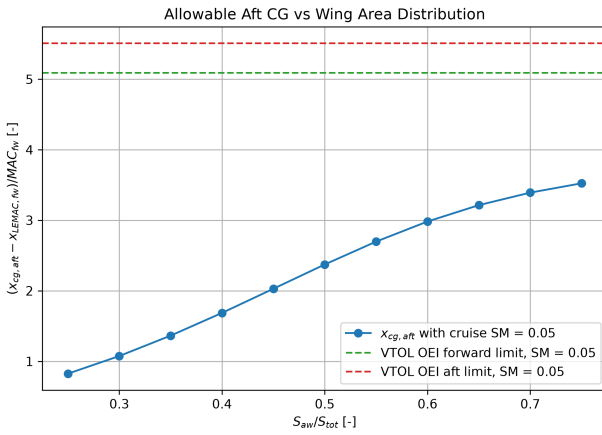


Figure 9.8: *Optimised Fuselage Sideview*

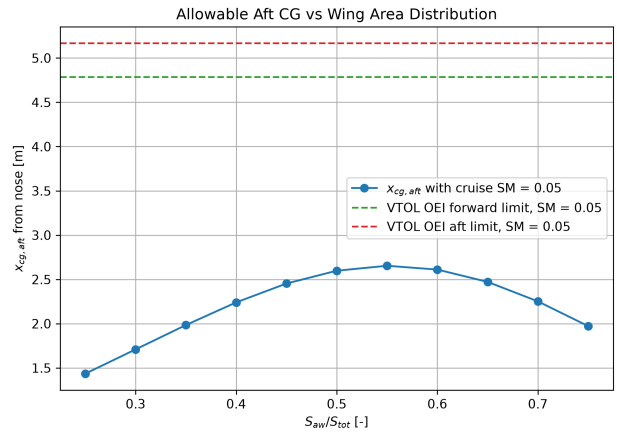
It should be noted that this new fuselage shape should be interpreted as an integration-driven geometry update rather than a separate aerodynamic optimisation. This new fuselage will be used in Chapter 7 in the Ansys analysis to get the final aerodynamic performance characteristics.

9.6. Discussion

In order to further improve the optimiser and associated code, the main improvement would be to eventually run a full optimiser sweep over all relevant parameters depicted in the Sobol analysis in Figure 9.4. To facilitate this, all design methods would need to be implemented within the code. In the current design scope this is not feasible for every constant to be parametrised, but this would be something to pursue in the future. Furthermore, additional objective functions could be added, such as a cost function to check the eVTOL price based on material usage. Lastly, as can be observed in Table 9.2 the C_{mq} derivative has a really high margin, so some penalisation to favour MTOW minimisation in this situation could potentially reduce mass further.

(a) Neutral point vs. S_{aft}/S_{tot} .(b) Aft c.g. limit vs. S_{aft}/S_{tot} .

(c) Aft c.g. limit (MAC-normalised) with VTOL-OEI band.



(d) Aft c.g. limit (nose datum) with VTOL-OEI band.

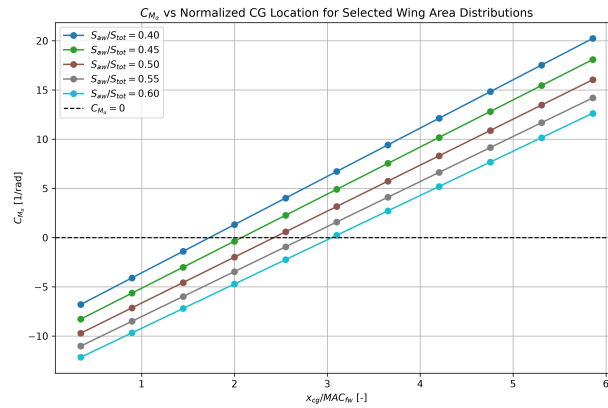
(e) C_{M_α} over the VTOL design envelope.

Figure 9.7: VTOL one-engine-inoperative (OEI) static stability and c.g. envelope.

10 Final Design Feasibility Assessment

This chapter evaluates whether the final Vold-e configuration is technically feasible at the current stage of the design. The assessment focuses on the aerodynamic performance of the integrated aircraft in cruise and on the structural performance of the folding-wing concept. Together, these analyses provide a final check of the design choices developed throughout prior chapters.

Section 10.1 focuses on the aerodynamic performance of the complete aircraft using CFD simulations. It presents the aerodynamic polars, the force balance at the cruise condition, and an analysis of the resulting flow field. The reliability of the CFD results are then assessed through an independence study and finally remaining modelling limitations are discussed. Section 10.2 evaluates the structural performance of the design by presenting the test setup and discussing the obtained FEA results. Finally, Section 10.3 summarises the aerodynamic and structural feasibility results and highlights the main areas requiring further refinement.

10.1. Aerodynamic Performance

The final aerodynamic performance was evaluated using steady RANS CFD simulations of the finalised integrated aircraft geometry in ANSYS Fluent. In contrast to the earlier aerodynamic analyses, which considered individual components or lower-fidelity numerical methods, this final analysis considers the integrated aircraft geometry. This allows the aerodynamic interaction between the main components to be captured more realistically than in the earlier component-level assessments. However, due to time constraints, the CFD simulations were conducted using the geometry available before implementation of the final optimisation described in Chapter 9. Therefore, these results should be interpreted as a detailed aerodynamic assessment of the pre-optimisation integrated design and as an indication of the expected aerodynamic behaviour of the final configuration, rather than as an exact prediction for the fully optimised aircraft.

The main objective of the analysis is to assess the final aerodynamic characteristics of the developed design in cruise configuration. Subsection 10.1.1 first presents the aerodynamic polars obtained from several simulation for a wide range of angle-of-attacks and uses the lift-curve analysis to determine crucial aerodynamic parameters, inter alia, the cruise angle of attack. Using this angle, Subsection 10.1.2 presents the resulting aerodynamic forces, force coefficients, and lift-to-drag ratio at the cruise flight condition. Subsection 10.1.3 examines the corresponding flow field, including the wake development, pressure distribution and skin-friction behaviour. The sensitivity of the results to mesh resolution and turbulence-model selection is assessed in Subsection 10.1.4, while the main assumptions and limitations of the CFD analysis are discussed in Subsection 10.1.5. These subsections focus on the analysis and interpretation of the obtained CFD results, while the descriptions of the geometry preparation, meshing strategy and solver setup are provided in Appendix A.

10.1.1. Aerodynamic Polars for Cruise Vehicle Configuration

The aerodynamic characteristics of the final cruise-vehicle configuration were first evaluated over a range of angles of attack in order to establish the cruise operating condition and assess the lift and drag development of the complete aircraft. Steady-state ANSYS Fluent simulations were performed using numerical setup as described in Appendix A, with the angle of attack varied from $\alpha = -4^\circ$ to $\alpha = 22^\circ$. The resulting lift and drag coefficients are presented in Table 10.1.

Table 10.1: Lift and drag coefficients for different angles of attack for the complete aircraft at cruise conditions. The values correspond to the full aircraft and were obtained by multiplying the half-model symmetry results by two.

α [deg]	C_L	C_D	α [deg]	C_L	C_D
-4	-1.46788e-01	1.76510e-02	14	8.82554e-01	9.42044e-02
-2	-1.00540e-02	1.70934e-02	16	9.43126e-01	1.12920e-01
0	1.23954e-01	1.91767e-02	17	9.70420e-01	1.24766e-01
2.6	2.95551e-01	2.46545e-02	18	9.84808e-01	1.39266e-01
4	3.84189e-01	2.98738e-02	19	9.91660e-01	1.53096e-01
6	5.07205e-01	3.87708e-02	20	9.46880e-01	1.64734e-01
10	7.15680e-01	6.24211e-02	22	8.11400e-01	2.13880e-01

The CFD data were subsequently used to construct the lift curve of the complete aircraft, shown in the Figure 10.1. The lift curve was used to determine the maximum lift coefficient, estimate the zero-lift angle of attack, and identify cruise the angle of attack.

As shown in Figure 10.1, a linear fit was applied only to the approximately linear region between $\alpha = -4^\circ$ and $\alpha = 6^\circ$. The fit was based on

$$C_L = C_{L,0} + a\alpha, \quad (10.1)$$

where $C_{L,0}$ is the lift coefficient at zero angle of attack and a is the lift-curve slope.

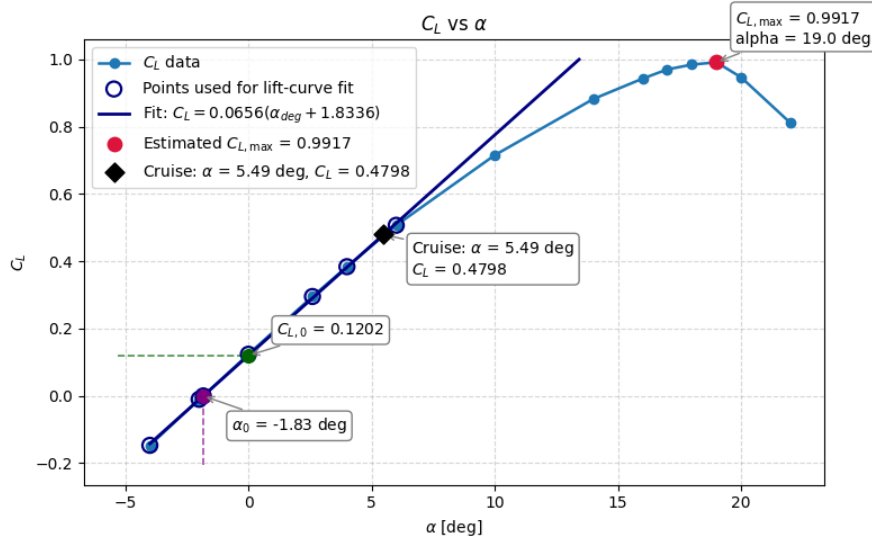


Figure 10.1: Lift curve of the complete aircraft obtained from ANSYS Fluent data, with a linear fit applied to the approximately linear region.

The maximum lift coefficient was taken directly as the highest C_L value obtained from the CFD simulations. The zero-lift angle of attack was estimated from the linear fit and then an additional ANSYS Fluent simulation was performed at this estimated zero-lift angle to obtain the corresponding zero-lift drag coefficient, $C_{D,0}$.

In addition, the lift coefficient required during cruise was determined from the lift-equals-weight condition using the final converged maximum take-off mass of $m_{MTOW} = 1980$ [kg]:

$$C_{L,cruise} = \frac{W}{q_{\infty} S} = \frac{m_{MTOW} g}{\frac{1}{2} \rho_{\infty} V_{\infty}^2 S}. \quad (10.2)$$

Using the cruise flight condition of $V_{\infty} = 55.6$ [m/s], $\rho_{\infty} = 0.836$ [kg/m³], and the reference area $S = 31.33$ [m²], the required cruise lift coefficient is $C_{L,cruise} \approx 0.480$. The corresponding cruise angle of attack was obtained from the linear lift-curve fit. The resulting aerodynamic parameters are provided in Table 10.2.

Table 10.2: Aerodynamic parameters obtained from the lift-curve and drag-polar analyses.

Parameter	Symbol	Value	Unit
Maximum lift coefficient	$C_{L,max}$	0.9917	–
Angle of attack at maximum lift	$\alpha_{C_{L,max}}$	19	[°]
Lift coefficient at zero angle of attack	$C_{L,0}$	0.1202	–
Zero-lift angle of attack	α_0	–1.83	[°]
Zero-lift drag coefficient	$C_{D,0}$	1.702×10^{-2}	–
Required cruise lift coefficient	$C_{L,cruise}$	0.480	–
Cruise angle of attack	α_{cruise}	5.49	[°]

The drag polar, shown in Figure 10.2, was constructed using the CFD data points presented in Table 10.1, together with the additional point evaluated at the estimated zero-lift angle of attack.

As visible in the Figure 10.2, a parabolic drag relation, given in Equation 10.3, was fitted to the discrete C_L and C_D data using a least-squares method. The fit was again applied only to the region between $\alpha = -4$ [°] and $\alpha = 6$ [°].

$$C_D = C_{D,0} + k C_L^2, \quad (10.3)$$

where $C_{D,0}$ represents the zero-lift drag coefficient and k is the induced-drag factor.

The fitted drag-polar factor was used to estimate the effective Oswald efficiency factor of the complete aircraft. The Oswald efficiency factor was therefore estimated using Equation 10.4.

$$k = \frac{1}{\pi A R e}, \quad e = \frac{1}{\pi A R k} = 0.692. \quad (10.4)$$

where e is the Oswald efficiency factor, AR is the aspect ratio, which from design geometry is $AR = \frac{b^2}{S} = 5.394$ and k is the induced drag factor, which using the fitted drag polar value is $k = 0.08522$.

Obtained Oswald efficiency factor should be interpreted as an **effective** Oswald efficiency factor and should not be confused with the ideal induced-drag span-efficiency factor predicted by the Rizzo box-wing relation [19]. Because

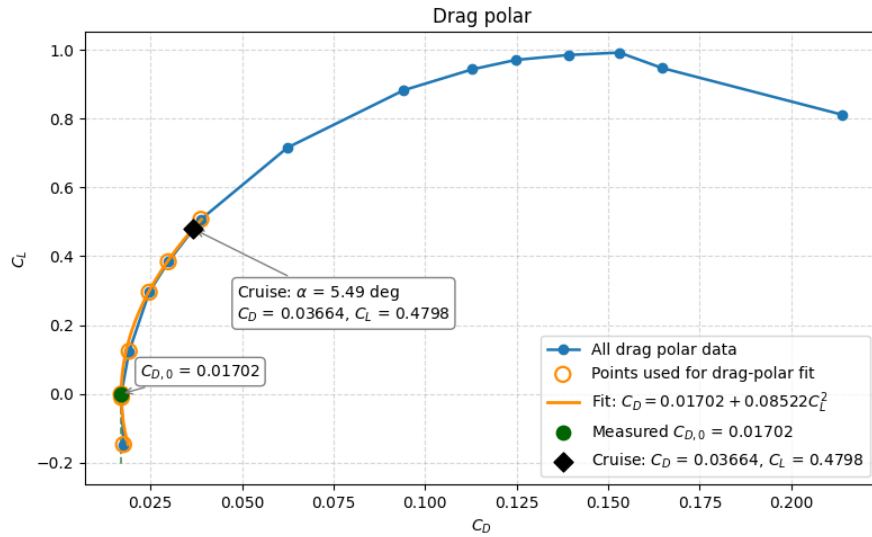


Figure 10.2: Drag polar of the complete aircraft obtained from ANSYS Fluent data, with a parabolic fit applied to the pre-stall region.

the effective factor is extracted from the full-aircraft CFD drag polar, the fitted parameter k includes all lift-dependent drag components. It reflects not only induced drag, but also viscous, pressure, and interference drag contributions captured in the CFD results. This creates a clear distinction: the theoretical Rizzo efficiency ($e = 1.34$) represents the isolated induced-drag benefit of the box-wing layout, while the CFD-derived efficiency ($e = 0.692$) captures the realistic performance of the complete viscous aircraft.

10.1.2. Force Analysis in Symmetric Cruise Flight

After determining the required cruise angle of attack from the lift-curve analysis, an additional ANSYS Fluent simulation was performed at the final cruise conditions. This simulation used the cruise velocity ($V_{\text{cruise}} = 55.6$ [m/s]) and altitude ($h_{\text{cruise}} = 12,500$ [ft]) specified in the mission analysis, together with the determined cruise angle of attack ($\alpha_{\text{cruise}} = 5.49^\circ$), to assess the aerodynamic performance of the integrated aircraft at its design operating point.

Because the CFD simulation was performed using a symmetry plane, only half of the aircraft was included in the computational domain. Therefore, the force and force-coefficient values reported directly by ANSYS Fluent were multiplied by two to obtain the corresponding full-aircraft values. Table 10.3 presents the resulting lift and drag components for the complete aircraft at the cruise condition.

Table 10.3: Force and force coefficient components for the complete aircraft at cruise conditions. The values correspond to the full aircraft and were obtained by multiplying the half-model symmetry results by two.

Zone	Forces [N]			Coefficients [-]		
	Pressure	Viscous	Total	Pressure	Viscous	Total
Drag direction: (0.9954, 0.0957, 0)						
Front wing	-5458.987	104.778	-5354.209	-0.13484	0.00259	-0.13225
Fuselage	5941.204	81.997	6023.201	0.14675	0.00203	0.14878
Rear wing	524.267	112.211	636.478	0.01295	0.00277	0.01572
Winglets	85.881	56.465	142.347	0.00212	0.00139	0.00352
Net	1092.366	355.451	1447.816	0.02698	0.00878	0.03576
Lift direction: (-0.0957, 0.9954, 0)						
Front wing	-62282.446	-1.082	-62283.528	-1.53844	-2.67×10^{-5}	-1.53846
Fuselage	71113.414	4.608	71118.022	1.75657	1.14×10^{-4}	1.75668
Rear wing	8520.935	-2.929	8518.006	0.21048	-7.24×10^{-5}	0.21040
Winglets	1709.360	1.927	1711.286	0.04222	4.76×10^{-5}	0.04227
Net	19061.263	2.523	19063.786	0.47083	6.23×10^{-5}	0.47089

The aerodynamic forces obtained from ANSYS Fluent were used to determine the final lift-to-drag ratio of the aircraft at the cruise condition. The resulting aerodynamic performance parameters are summarised in Table 10.4.

Table 10.4: Total aerodynamic performance of the complete aircraft at cruise conditions. The values correspond to the full aircraft and were obtained by multiplying the half-model symmetry results by two.

Parameter	Symbol	Value	Unit
Drag force	D	1447.816	N
Lift force	L	19063.786	N
Drag coefficient	C_D	0.03576	–
Lift coefficient	C_L	0.47089	–
Lift-to-drag ratio	L/D	13.17	–

The required cruise lift coefficient obtained from the lift-equals-weight condition was $C_{L,\text{cruise}} = 0.4798$, which based on the linear lift-curve fit corresponded to an estimated cruise angle of attack of $\alpha_{\text{cruise}} = 5.49^\circ$. However, as presented in Table 10.4, the additional CFD simulation performed at this angle of attack resulted in $C_L = 0.47089$, which remains 0.0089 below the required value. That difference of 1.9% indicates that the analysed operating point is close to, but not exactly at, the trimmed cruise condition.

The difference arises because the cruise angle of attack was obtained from a least-squares linear fit to a finite number of discrete CFD results and therefore the fitted relation does not necessarily reproduce the result of an individual simulation exactly. An additional small increase in angle of attack would therefore be required to obtain the exact lift-equals-weight condition. However, the remaining discrepancy is small and does not affect the feasibility conclusions of the present assessment. A series of additional simulations around $\alpha = 5.49^\circ$ was therefore not performed, since each integrated-aircraft CFD simulation requires substantial computational resources. Consequently, $\alpha = 5.49^\circ$ is treated as a sufficiently accurate approximation of the trimmed cruise operating point and is also referred to as *cruise condition* in the further analyses.

10.1.3. Cruise Flow Field Analysis

To support the numerical force results and identify the main sources of aerodynamic losses, the surrounding flow field was also analysed at cruise condition using velocity vectors, velocity contours and wake-recovery plots.

The velocity-vector field is first inspected on several spanwise planes to obtain a qualitative overview of the flow around the complete aircraft. The planes are placed at $z = 0$ [m], $z = \pm 3$ [m] and $z = \pm 6.4$ [m], so that both the central fuselage flow and the outer wing/winglet regions are included. The results visualized in Figure 10.3 show that the freestream remains mostly aligned with the aircraft longitudinal direction, while local changes in velocity magnitude appear around the wing leading edges, winglets and fuselage. These regions indicate where the geometry locally accelerates or decelerates the flow. The vector field also shows that the main disturbance remains concentrated close to the aircraft, with the flow becoming more uniform further away from the body.

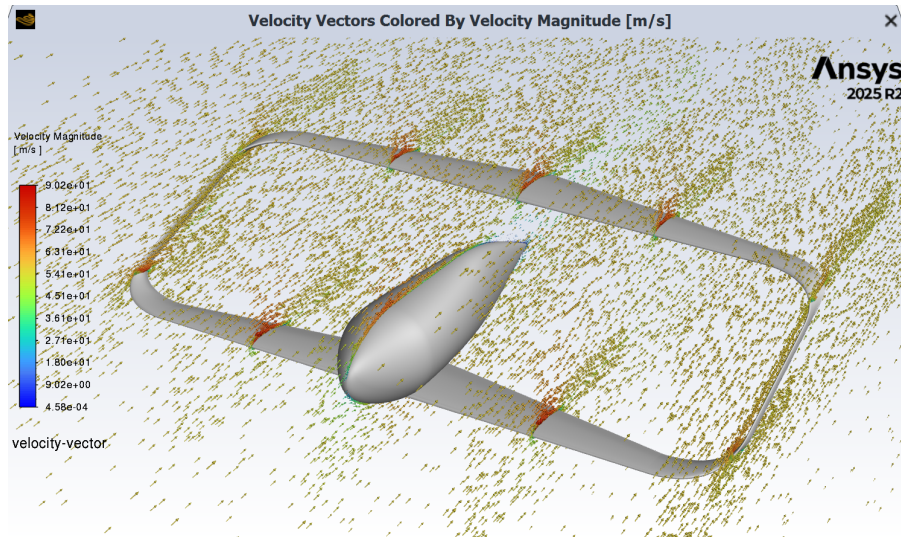


Figure 10.3: Velocity vectors around the complete aircraft in cruise configuration on planes at $z = 0$ m, $z = \pm 3$ m and $z = \pm 6.4$ m.

The velocity contours in the top and side views, shown in Figure 10.4, provide a clearer view of the wake formation behind the aircraft. In the top view (Figure 10.4a), the wake remains narrow and is mainly located behind the fuselage and lifting surfaces. The lower-velocity regions downstream of the aircraft indicate the momentum deficit caused by pressure drag, viscous losses and local flow turning around the geometry. In the side view (Figure 10.4b), the wake behind the fuselage can be seen more clearly. The velocity deficit starts directly behind the aft fuselage and gradually expands downstream, while the surrounding flow remains close to the freestream velocity. This suggests that the strongest wake effects are confined to a limited region behind the aircraft.

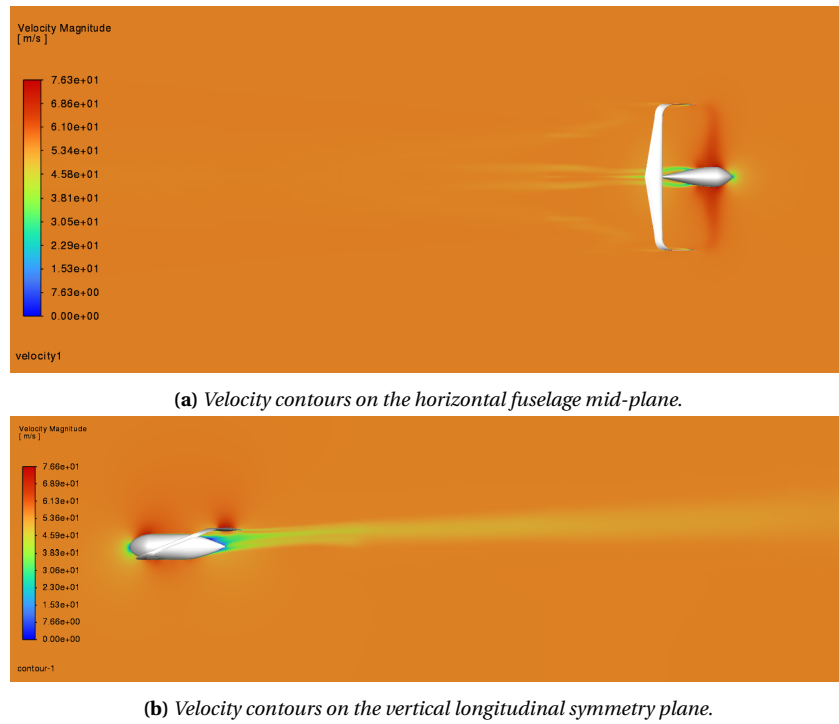


Figure 10.4: Velocity magnitude contours showing the wake formation behind the aircraft in cruise configuration.

To quantify wake recovery, the velocity magnitude was extracted along two lines downstream of the fuselage, as shown in Figure 10.5. One line is horizontal, while the other is inclined at the cruise angle of attack of 5.49° . Both lines start at the aft end of the fuselage and extend to the end of the computational domain. The velocity distribution shows a strong initial deficit immediately behind the fuselage, where the flow has not yet recovered from the body wake. Further downstream, the velocity increases progressively towards the freestream value of 55.6 [m/s]. This confirms that the wake weakens with distance from the aircraft and that the flow gradually recovers as momentum is redistributed from the surrounding higher-speed flow into the wake region.

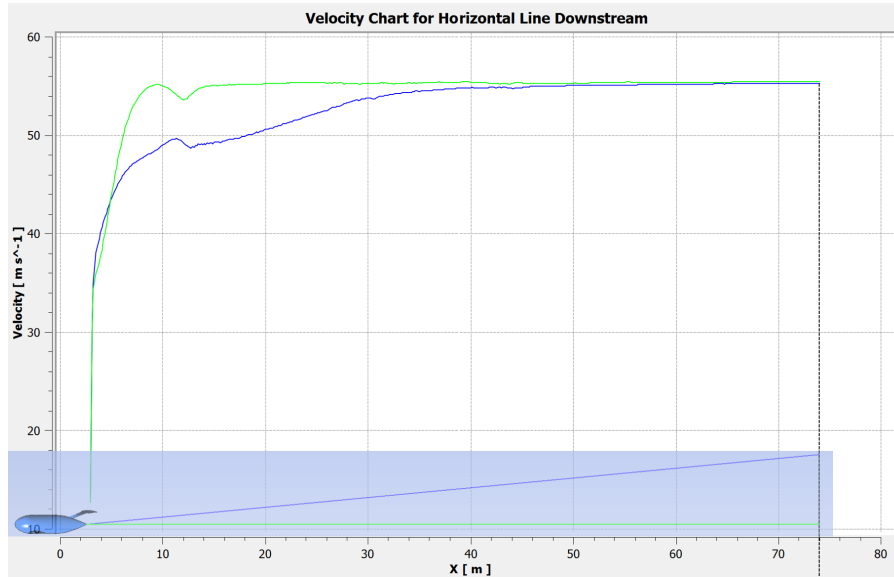


Figure 10.5: Velocity magnitude along horizontal (green) and 5.49° -inclined (blue) lines downstream of the fuselage, showing recovery of the fuselage wake towards the freestream velocity.

The downstream contour planes further illustrate this recovery process. As visualised in Figure 10.6a, at $x = 10$ [m], corresponding to one fuselage length (7[m]) downstream of the aft fuselage end, the wake is still clearly visible as a localized low-velocity region. The disturbance is strongest close to the centre of the wake and is still influenced by the fuselage and lifting-surface interaction. Next figure, Figure 10.6b, shows velocity contours at $x = 30$ [m] ($\approx 4L_{fus}$ downstream the fuselage end), the velocity deficit is significantly weaker and more diffuse. The wake has expanded and mixed with the surrounding flow, reducing the magnitude of the local velocity disturbance. This comparison

confirms that the flow disturbance decreases downstream and that the selected computational domain is sufficiently long to capture the main wake-recovery trend.

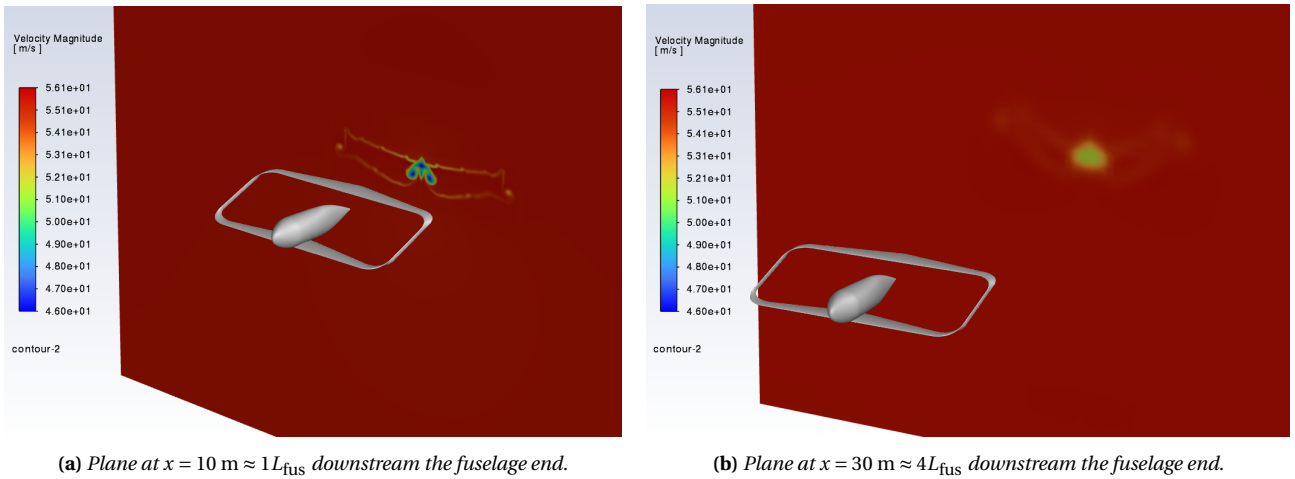


Figure 10.6: Velocity magnitude contours on downstream planes, showing the reduction of the wake disturbance with distance.

The pressure field was then analysed to identify how the lift-producing pressure difference is formed over the aircraft. The pressure-coefficient distributions over the front and rear wings are shown in the Figure 10.7a and Figure 10.7b, respectively. In both plots, the curve with the more negative pressure coefficient corresponds to the upper surface of the wing, while the less negative or locally positive curve corresponds to the lower surface. This is because the upper surface accelerates the flow and therefore produces a lower static pressure. Both wings show a strong suction peak close to the leading edge, followed by a gradual pressure recovery towards the trailing edge. The front wing reaches a slightly stronger suction peak than the rear wing, indicating a higher local loading near the leading edge. The overall separation between the upper and lower surface curves confirms that both lifting surfaces contribute to the positive lift generated by the aircraft.

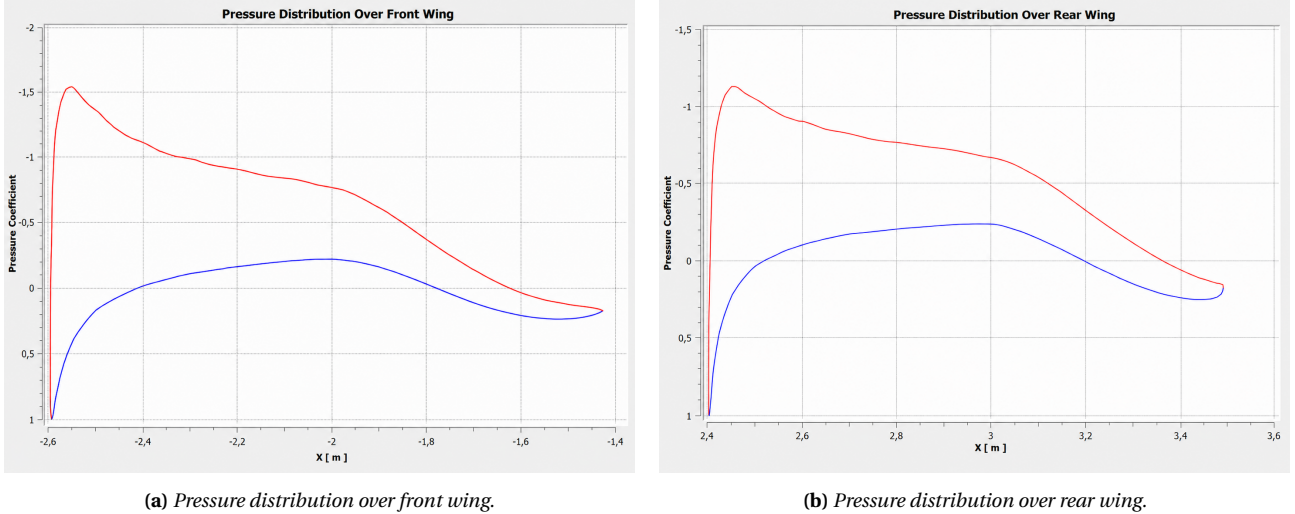


Figure 10.7: Chordwise pressure-coefficient distributions on the front and rear wings in the cruise configuration. The red and blue curves represent the upper and lower wing surfaces, respectively.

The surface pressure contours support the trends observed in the pressure-coefficient plots. The Figure 10.8a shows the upper side of the wings which obtains a lower, more negative pressure, visible as blue regions over large parts of the lifting surfaces. In contrast, Figure 10.8b demonstrates the lower side which shows higher pressure values, indicated by the green, yellow and red regions. This pressure difference between the upper and lower surfaces is the main source of lift. To make the pressure variations over the aircraft more visible, the contour range was limited to -750 [Pa] to $+500 \text{ [Pa]}$.

Finally, Figure 10.9 shows the static pressure contour on the central (XY)-plane ($z = 0 \text{ [m]}$). A high-pressure region is visible at the nose, where the incoming flow decelerates near the stagnation point. Downstream of the nose, the pressure decreases as the flow accelerates around the fuselage. A pressure recovery then occurs towards the aft section, although the wake region behind the fuselage still shows a disturbed pressure field. The same plane also shows the pressure disturbance around the rear wing section, where a low-pressure region forms over the upper side of the

airfoil. Together, these plots show that the aircraft pressure field is dominated by the nose stagnation region, wing suction regions and the wake recovery behind the fuselage.

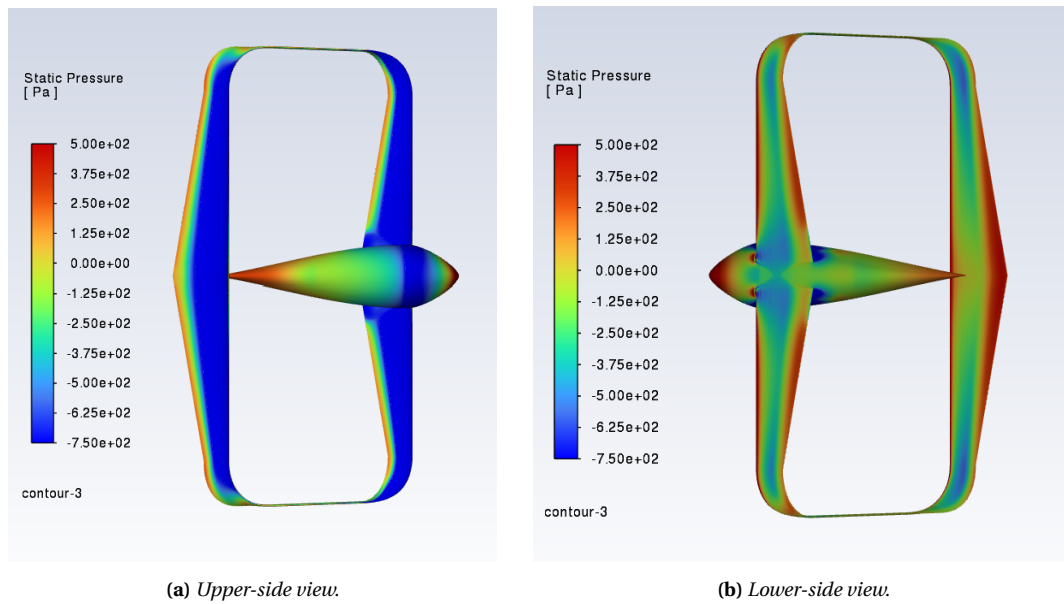


Figure 10.8: Surface static pressure contours over the aircraft, with the range limited to -750 Pa to $+500$ Pa.

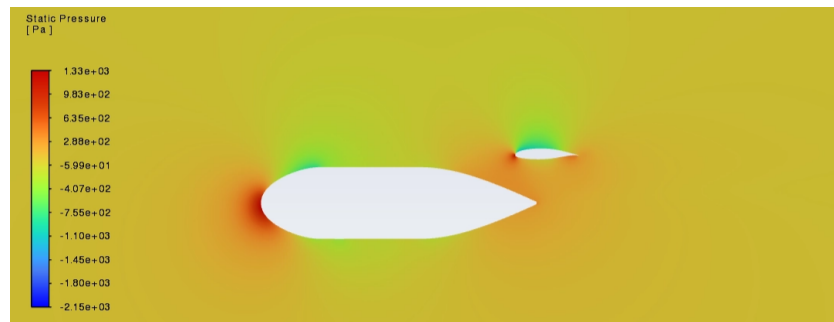


Figure 10.9: Static pressure contour on the central XY -plane at $z = 0$ m.

The final surface quantity inspected is the skin-friction coefficient, shown in Figure 10.10. The skin-friction coefficient indicates the strength of the wall shear stress and is therefore useful for identifying regions where viscous effects are important. Higher values are observed around the wing leading edges, winglets and curved junction regions, where the local velocity gradients near the wall are larger. On the fuselage, the coefficient is generally lower and decreases towards the aft section, which is consistent with boundary-layer growth along the body. The plotted values do not fall below zero, meaning that no reversed wall shear is observed on the aircraft surface. This indicates that the flow remains attached over the visible surfaces and that no major separated region is present at the analysed cruise condition. Very low values may still occur locally in low-shear or near-stagnation regions, but the absence of negative values supports the conclusion that the final configuration does not exhibit large-scale surface flow separation.

10.1.4. Independence Study

A mesh independence study was performed to assess the sensitivity of the predicted aerodynamic coefficients to the spatial discretisation. The mesh used for the final aerodynamic analysis was treated as the medium mesh, while additional coarse and fine meshes were generated by scaling the principal surface, volume and local-refinement sizes. The resulting mesh levels and cell counts are summarised in Table 10.5. The computational domain, boundary conditions, turbulence model, solver settings and inflation first-layer height were kept unchanged to ensure that the observed differences resulted only from mesh resolution.

Table 10.5: Overview of the meshes used for the mesh independence study

Mesh	Refinement factor	Number of cells
Coarse	$1.4h$	9,576,897
Medium (used)	h	15,071,004
Fine	$h/1.4$	27,906,785

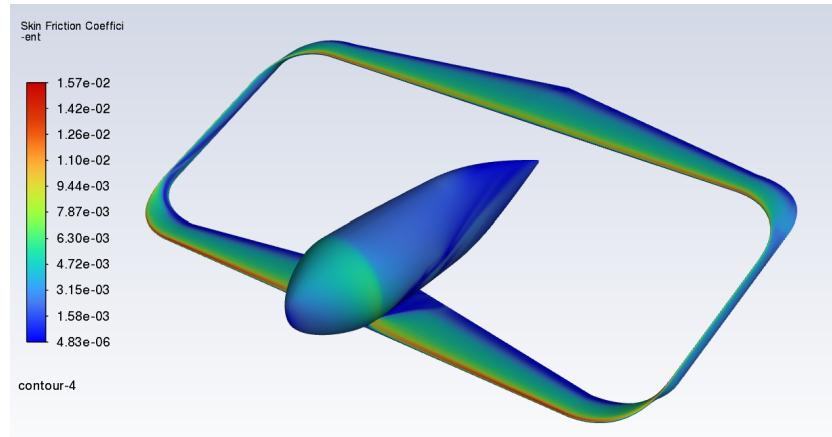


Figure 10.10: Skin-friction coefficient distribution over the complete aircraft in cruise configuration.

The relative differences in the aerodynamic quantities were calculated using the fine mesh as the reference solution, as it provides the closest available approximation to a mesh-converged result. This allows the difference between the medium and fine meshes to be evaluated directly and determines whether the medium mesh provides sufficient accuracy for the final simulations at a lower computational cost. The relative differences in lift coefficient, drag coefficient and lift-to-drag ratio were calculated using Equation 10.5.

$$\Delta C_L = \frac{|C_{L,i} - C_{L,\text{fine}}|}{C_{L,\text{fine}}} \times 100\%, \quad \Delta C_D = \frac{|C_{D,i} - C_{D,\text{fine}}|}{C_{D,\text{fine}}} \times 100\%, \quad \Delta(L/D) = \frac{|(L/D)_i - (L/D)_{\text{fine}}|}{(L/D)_{\text{fine}}} \times 100\%, \quad (10.5)$$

For this study, a difference below approximately 3% between the medium and fine meshes was considered acceptable for the main aerodynamic coefficients and the resulting lift-to-drag ratio. In particular, the drag coefficient was expected to show a difference below approximately 2%, as it is typically the most mesh-sensitive quantity and directly affects the predicted aerodynamic efficiency. The calculated differences, obtained using Equation 10.5, are presented in Table 10.6.

Table 10.6: Aerodynamic results of the mesh independence study. Percentage differences are calculated relative to the fine mesh.

Mesh	C_L	C_D	L/D	ΔC_L	ΔC_D	$\Delta(L/D)$
Coarse	2.3052×10^{-1}	1.7748×10^{-2}	12.989	2.74%	3.49%	0.78%
Medium (used)	2.35447×10^{-1}	1.78812×10^{-2}	13.167	0.66%	2.77%	2.17%
Fine	2.3701×10^{-1}	1.8390×10^{-2}	12.888	0.00%	0.00%	0.00%

The medium mesh differs from the fine mesh by 0.66% in C_L , 2.77% in C_D , resulting in difference of 2.17% in L/D , indicating that the medium mesh provides sufficiently mesh-independent aerodynamic results while requiring lower computational cost than the fine mesh.

Furthermore, a turbulence model study was performed to assess the sensitivity of the predicted aerodynamic coefficients to the selected turbulence model. This comparison is relevant because different RANS turbulence models may predict different flow behaviour, which can affect the overall lift and drag of the integrated aircraft configuration. The SST $k-\omega$ model was compared with the Realizable $k-\epsilon$ model with enhanced wall treatment and the Spalart-Allmaras model. These models were selected as they both are commonly used for aerospace flows and are relatively compatible with the fine near-wall resolution used in this study. The percentage differences were calculated relative to the SST $k-\omega$ results. The results of the comparison are presented in Table 10.7.

Table 10.7: Turbulence model independence study results. Percentage differences are calculated relative to the SST $k-\omega$ model.

Mesh	C_L	C_D	L/D	ΔC_L	ΔC_D	$\Delta(L/D)$
SST $k-\omega$ (used)	2.35447×10^{-1}	1.78812×10^{-2}	13.167	0.00%	0.00%	0.00%
Spalart-Allmaras	2.37250×10^{-1}	1.79530×10^{-2}	13.215	0.77%	0.40%	0.36%
Realizable $k-\epsilon$ with enhanced wall treatment	2.37190×10^{-1}	1.71820×10^{-2}	13.805	0.74%	3.91%	4.84%

The Spalart-Allmaras model produced aerodynamic coefficients close to those obtained using SST $k-\omega$, while the Realizable $k-\epsilon$ model resulted in larger differences, particularly in the predicted lift-to-drag ratio. Overall, the turbulence-model comparison indicates that the predicted aerodynamic performance is not strongly dependent on the selected RANS model for the analysed cruise condition.

10.1.5. CFD Limitations

It should be noted that this CFD analysis is used as a final preliminary verification of the aerodynamic design. The analysis captures the integrated aircraft geometry, but several simplifications were required to obtain a feasible and robust simulation setup. Therefore, the results are considered suitable for final design consistency checks within the scope of this design phase, but not as a fully validated prediction of the final aircraft performance. The main limitations and their expected influence are summarised in Table 10.8.

In addition, the CFD simulations were performed before the final optimisation described in Chapter 9. Consequently, the analysed geometry and associated design parameters may differ slightly from those of the final optimised configuration.

Table 10.8: *Main CFD modelling limitations, expected influence and rationale*

Simplification / limitation	Expected influence	Rationale
Propellers not modelled	Rotor slipstream, rotor-wing interaction and propeller drag are excluded; powered-aircraft lift and drag may differ.	The assessment focused on the clean integrated airframe; rotating propellers would substantially increase computational cost.
V-tail excluded	Directional effects, local tail-fuselage-wing interactions and a small portion of parasite drag are omitted.	Its contribution was secondary for symmetric, longitudinal cruise analysis, while excluding it simplified meshing.
Small external elements omitted	Parasite and junction drag may be underestimated.	Features such as skids and attachments require disproportionate local mesh refinement.
Detailed geometry simplified	Local separation, pressure peaks and small-scale drag contributions may not be represented fully.	Gaps, openings and small sharp edges reduce meshing robustness.
Hollow elements closed	Internal flow paths and cavity-pressure effects are not captured.	Watertight surfaces were required for reliable volume meshing.
Ideal smooth surfaces assumed	Skin-friction drag may be lower than for a manufactured aircraft.	Surface roughness and imperfections were outside the scope of preliminary CFD.
Steady RANS approach	Unsteady flow and strong separation, particularly near stall, may be predicted less accurately; $C_{L,max}$ should be treated cautiously.	It provides a practical accuracy-cost compromise for design-phase analysis.
Turbulence model and wall treatment	Boundary-layer growth, pressure recovery and separation onset depend on model assumptions and near-wall resolution.	The setup is suitable for comparative analysis but retains inherent modelling uncertainty.
Finite domain and numerical discretisation	Domain boundaries, mesh resolution and discretisation may affect force coefficients.	Settings balanced accuracy, convergence robustness and feasible computational cost.

10.2. Structural Performance

In order to analyse and validate the performance of the folding mechanism, its most critical part, the root hinge, was tested using the simulation module in Fusion 360. The type of study performed on the part was static stress. In Subsection 10.2.1 the 3D model of the hinge is meshed. Then in Subsection 10.2.2, the constraints and the applied forces and moments are shown. Finally, in Subsection 10.2.3, the results of these simulations are shown.

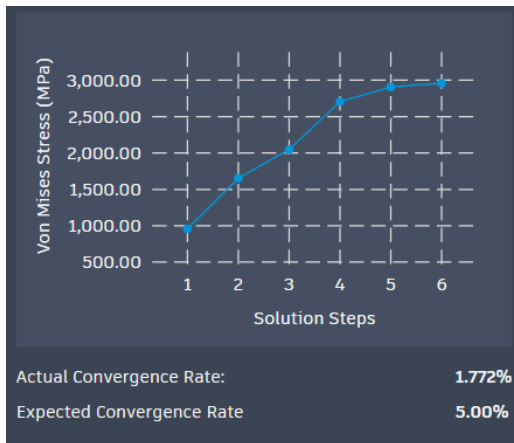
10.2.1. Meshing

In order to calculate the stress distribution in the hinge, it needs to be split in thousands of small elements, due to the complexity of the shape. For this study, 10-node parabolic tetrahedral elements are used. In comparison to the linear elements, that only have 4 nodes, connected to their vertices, the parabolic elements have 6 additional nodes placed in the middle of their sides. This allows the simulation to capture more complex deformations and provide higher fidelity of the results.

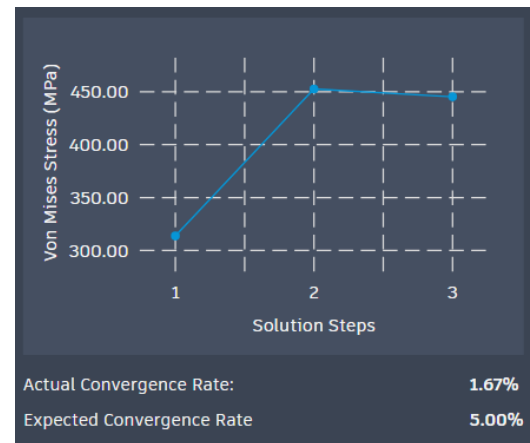
Next, the density distribution of the number of these elements needs to be determined. The more elements there are, the longer the computational time. However, if they are not enough the simulation might not represent the deformations in the part correctly. Also, some regions of the part have a shape that is much harder to mesh and require more elements.

To ensure the simulation is valid a mesh independence analysis is done through an iterative process. The simula-

tion software generates multiple mesh sizes until the Von Mises stress does not vary by more than 5%. The purpose of that is to show that reducing the mesh further will not increase the fidelity of the analysis meaning the result is independent of mesh size. In Figure 10.11 the Von Mises stress curves are shown for all the mesh iterations.



(a) Study 1



(b) Study 2

Figure 10.11: Mesh independence study using Von Mises stress convergence

The final mesh is shown in Figure 10.12.

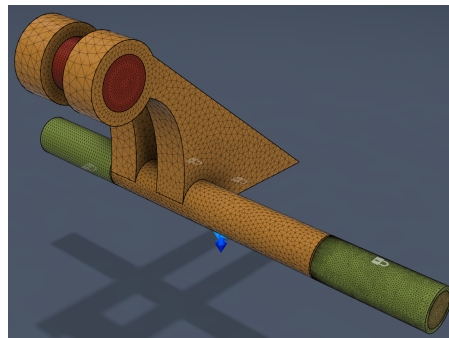
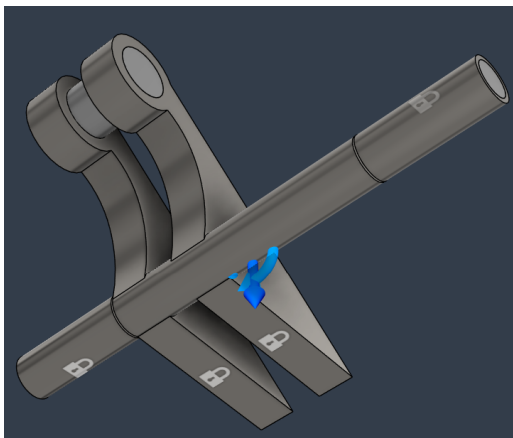


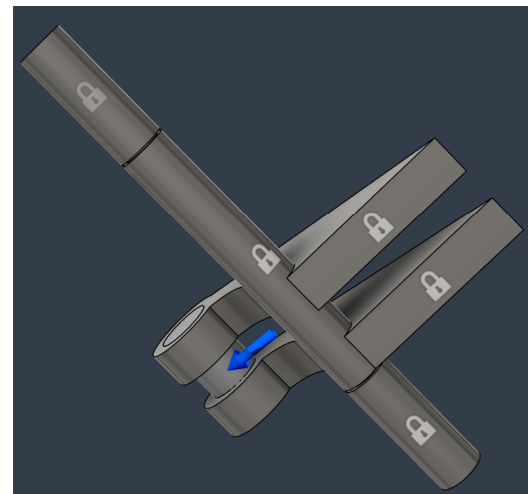
Figure 10.12: Meshed hinge

10.2.2. Forces and Constraints

In order to make the study easier to compute only the hinge was imported in the test environment. This means that before applying any forces on it, it needs to be constrained in a way that simulates its contacts with other parts realistically. In Figure 10.13a the faces with padlock icons are fully constrained. That is because these faces will be welded to the structure of the wing and are supposed to stay fixed.



(a) Study 1



(b) Study 2

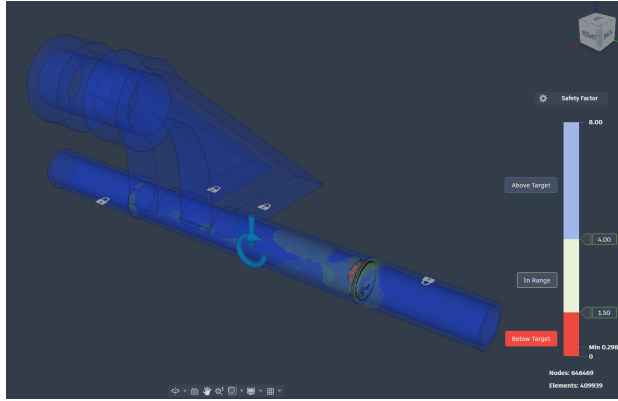
Figure 10.13: Constraints and forces for the two studies

The load applied on the component in the first study consists of a moment and a force applied perpendicularly to the hinge axis, as shown in Figure 10.13a. This simulates the resultant moment and shear force that the spar transfers to the hinge. The magnitude of the moment is 8500 [Nm] and of the force 3500 [N].

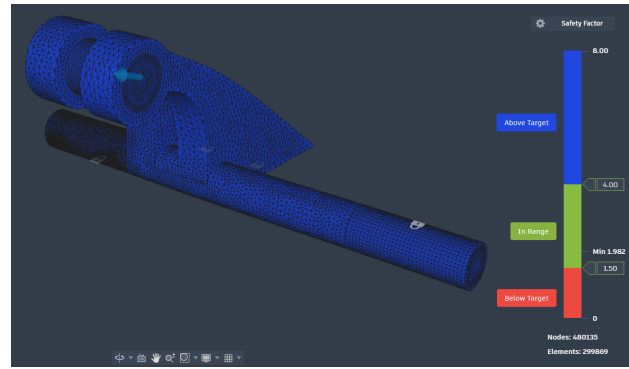
The load applied on the component in the second study is a point force placed in the middle of the pin connecting the hinge with the linear actuator. The direction and magnitude are for the case, when combination of resultant moment arm and required torque around the hinge axis of rotation results in the highest actuator force. The value for the force is 56 [kN].

10.2.3. Results

The results from the two studies are shown in Figure 10.14



(a) Results for Study 1.



(b) Results for Study 2.

Figure 10.14: Test results for the two studies.

For both configurations, the minimum required safety factor is 1.5. As can be seen in the first study there are regions on the connecting pin that are in the red region, which suggests that part of it might yield during the operation. To fix this issue two approaches are suggested.

The first one is to change the material of the pin to one with higher yield strength. The second one is to increase the diameter of the pin. A combination of the two approaches could also be used in order to create a reliable structure with minimal mass increase. In the second study it could be seen that the stress in the whole structure is above the safety factor, and in most regions it is even above 4. This means that some weight reductions might be done in the further development of the hinge design.

10.3. Discussion

The final design assessment supports the feasibility of Vold-e at the current design stage. Integrated-aircraft CFD predicts a cruise lift-to-drag ratio of $L/D = 13.17$, with the analysed cruise point producing lift within 1.9% of the lift-equals-weight requirement and showing no major surface flow separation. The mesh and turbulence-model studies indicate that these results are sufficiently robust for preliminary design, although propeller effects, transition aerodynamics and final-geometry updates remain to be assessed. The hinge FEA confirms that most of the structure has adequate margin, but identifies the connecting pin as a critical region requiring redesign before the folding mechanism can be considered fully feasible.

11 Verification & Validation

Verification and validation are required to ensure that the methods, assumptions and models used during the design process are sufficiently accurate and reliable. Since the current design is still at a preliminary stage, the focus of this chapter is mainly on defining a clear V&V procedure that can be applied consistently during the final design phase.

Section 11.1 outlines the procedure for model verification, including the reporting format and verification test catalogue. Section 11.2 describes the procedure for model validation necessary to assess whether the results represent the real physical behaviour of the aircraft with sufficient accuracy. Finally, Section 11.3 applies the defined approach to the current design models, including the validation of the battery sizing method using reference eVTOL data. Section 11.4 provides compliance matrix, in which fulfillment of the user requirements is assessed. Finally, Section 11.5 summarises the implemented verification and validation activities and identifies the further work required to demonstrate compliance with requirements that remain unfulfilled or cannot yet be verified.

11.1. Procedure of Model Verification

This section defines the procedure used to verify the models and calculations developed during the preliminary design phase. The purpose of model verification is to check whether the implemented methods correctly represent the intended equations, assumptions and input-output relations. This is especially important because several results used in the trade-off and performance estimation rely on simplified analytical models and code-based calculations.

11.1.1. Reporting Format

When performing verification, models are divided into a whole system, with subdivisions corresponding to modules and subsequently ending with units. As an example illustration the structures sub-system in the excel model is defined.



Figure 11.1: Depiction of sub-system and module differentiation

Within each module detailed in Figure 11.1, multiple units are available for testing, such as the values obtained for v_{cr} , which is what must be verified first.

To ensure repeatability and traceability, a 'test card' is used. This details the subject matter of what is being tested, along with the execution method and how a pass or fail decision is made. This method furthermore allows for the methodology to be audited, as clear explanation on tests allow for them to be reproduced by a third party.

Table 11.1: Information tracked in the 'Test Card' format

Test ID	Input	Expected Output	Actual Output	Pass Condition	Pass/Fail

Within Table 11.1, the ID corresponds to the name of the test taken. Firstly, the initial letters correspond to the (sub)system being tested (sub-systems are used for a larger overall model but for smaller systems a system identifier is deemed appropriate). ST refers to structures and AE to aerodynamics. Future abbreviations will be clarified upon their introduction. The letter after the first hyphen corresponds to the test type, with **U** for unit test, **M** for module test, **S** for (sub)system tests, and **V** for validation. The number afterwards confirms the specific test ID, which will be specified in future test catalogues. Lastly, the number after the second hyphen corresponds to the order that the test was taken and the letter to the iteration. For example, **ST-U1-1a** would refer to the first test and first attempt of unit test 1 on the structures sub-system. Further letters occur if a fail has been determined.

11.1.2. Test Type Catalogue for Verification

The procedure to complete the verification process follows a bottom up approach, in which individual units are first tested, and once passed, testing is conducted on higher levels such as modules. This culminates in the verification of the whole system, allowing validation to be performed.

Table 11.2: Unit test type catalogue used for verification of conceptual design models. Applicable to spreadsheet-based and Python-based implementations of any sizing, tradeoff or stability calculation produced during the project.

ID	Name	Failure mode that is detected	Execution method	Pass criterion
Formula Consistency Tests				

Continued on next page

ID	Name	Purpose / failure mode caught	Typical execution	Pass criterion
U1	Formula Inspection	Catches typos and copy-paste errors where the implementation differs from the equation specified in the literature study or referenced source.	Manually inspect each formula (spreadsheet cell or code line) against the source equation, including ranges of sums and units of constants.	Each formula matches the literature reference symbol-for-symbol.
U2	Unit Consistency Check	Catches errors where unit conversion factors are missing or wrong, producing numerically plausible but dimensionally inconsistent results.	Trace units through each calculation step, identifying conversion factors (e.g. $\text{kW} \times \text{s} \rightarrow \text{kJ}$, requiring divide-by-3600 to yield kWh).	Output units of every derived quantity match the units shown in the column header or variable definition.
Boundary and Null Input Tests				
U3	Null Input Test	Verifies the model returns a physically meaningful result when key inputs are set to zero, confirming the formulas do not introduce spurious offsets or divide-by-zero errors.	Set range, mass, or other dominant input variables to zero.	Outputs evaluate to zero where physically expected, or trigger a controlled error where division by zero would otherwise occur.
U4	Extreme Value Test	Ensures the model does not overflow, produce NaN, or silently truncate when extreme but physically valid inputs are entered.	Set inputs to physically large or small but valid values (e.g. MTOW $\rightarrow 10 \text{ t}$, $L/D \rightarrow 0.1$).	All outputs remain finite and ordered consistently with the input change.
Analytical Reference Tests				
U5	Hand-Computed Reference	Verifies that the model produces the correct output for a reference case that can be solved with pencil-and-paper.	Select two reference cases spanning typical regimes; compute the expected output by hand; compare with model output.	Agreement within 1% of the hand-computed reference value.
U6	Independent Implementation Cross-Check	Catches systematic errors in the spreadsheet by reproducing the same calculation in a different tool.	Implement the model in Python (or vice versa); run with identical inputs and compare outputs.	All outputs match within 0.1% across the full set of computed quantities.
Metamorphic Tests				
U7	Scaling Test	Verifies that output quantities scale correctly with input parameters	Vary the input by a multiplied factor and observe the change in output	Output should match the factor change expected within a specified tolerance
U8	Conservation Check	Verifies that summed quantities equal the total reported by the model, catching cases where a contribution is omitted from the aggregate.	Sum component contributions independently; compare with the total reported by the model (e.g. phase energies summing to mission energy, subsystem masses summing to MTOW).	Independent sum matches reported total within float tolerance.
Interface and Input/Output Tests				
U9	Input Range Check	Catches errors caused by inputs outside the model's valid range, particularly when models are used outside the regime they were calibrated on.	Verify the input value lies within the literature-validated range (e.g. disk loading $50\text{--}600 \text{ kg/m}^2$, L/D $2\text{--}12$ for eVTOLs).	Inputs are within the valid range, or the model documents the extrapolation explicitly.
U10	Output Format Check	Catches errors where the output type or units differ from what a downstream model expects.	Verify output column headers, units, and data types match the consumer's expected format.	All outputs satisfy the downstream consumer's interface contract.

Performing the relevant tests successfully is an indication that module tests can be performed. The naming convention for module tests is the same for the unit tests, but with an M in the test ID. These are higher level but ultimately follow the same tests as for individual units as depicted in Table 11.2.

Lastly, sub-system and system wide tests are performed in order to get a global sense that the model functions as intended. System tests are conducted to verify that the individual modules behave correctly when integrated into the full conceptual design pipeline. Unit tests treat each formula or block in isolation, system tests act on the model as a whole and address the failure modes that can only be observed once inputs flow from one module into another.

Table 11.3: *System test type catalogue used for verification of integrated conceptual design models.*

ID	Name	Failure mode that is detected	Execution method	Pass criterion
S1	Sensitivity Analysis	Identifies which input parameters dominate the model output, allowing the design team to focus uncertainty quantification on the parameters that matter, and catches non-linear interaction effects between inputs.	Perturb each input either via manual tweaking of values in the realm of around 10% or by running a more extensive sensitivity analysis model	Perturbations in parameters should not effect global conclusions made from the model
S2	Regression Test	Catches cases where a change to a formula, an assumption, or a literature source has inadvertently shifted a previously validated output.	Rerun model with new information, and check whether the output still aligns to the overall conclusions made	Outputs match the previous baseline within the regression tolerance, or the change is documented and the new baseline is committed.
S3	Validation Test	Verifies the model can reproduce the design of a known production eVTOL when fed its published input parameters, building confidence that the model is applicable to the project's novel design.	Run the model with the published parameters of a reference eVTOL (e.g. Joby S4, Vahana α); compare predicted MTOW, battery mass, L/D against published values.	Predicted outputs match published values within 20%, or any deviation is explained by documented modelling assumptions.
S4	Stress Test	Verifies the model returns defensible answers when applied to mission cases beyond the primary design point or to physically degraded inputs, ensuring the design is not fragile to off-nominal conditions.	Run the model with off-design mission inputs (e.g. ferry, hot-and-high) and with degraded inputs (e.g. battery at 80% of beginning-of-life energy density).	Outputs remain finite and physically reasonable and output degradation matches the expected physical relationship without sever model degradation.

11.2. Procedure of Model Validation

Validation establishes that the resulting outputs are a credible representation of physical reality and of the design's intended behaviour. Validation operates on the verified system as a whole and is therefore performed only after the unit, model and system tests of the verification procedure have been passed. Three complementary lines of evidence are used to validate the conceptual design models produced during the project: comparison against published reference eVTOL aircraft, cross comparison between independent modelling approaches, and physical reasonableness checks against first-principle expectations. Together these lines of evidence provide confidence that the models are applicable to the project's novel Prandtl-plane configuration.

Table 11.4: *Validation test type catalogue for conceptual-design models.*

ID	Name	Failure mode that is detected	Execution method	Pass criterion
V1	Reference-Aircraft Validation	Confirms the model produces realistic outputs when fed the published parameters of a known production eVTOL, building confidence that it can be applied to the project's novel design.	Use the published inputs of one or more reference eVTOLs (e.g. Joby S4, Vahana α , Cora) to drive the model; compare its predicted MTOW, battery mass, hover power, and L/D against the published values.	Predicted outputs match published values within 20%, or any larger deviation is explained by documented modelling assumptions.
V2	Cross-Model Comparison	Confirms that independent modelling approaches (e.g. semi-empirical sizing versus first-principles analytical sizing) produce mutually consistent results, ruling out systematic methodology bias.	Compare outputs of independent modelling approaches applied to the same design point (e.g. Ugwueze semi-empirical method versus Smart and Alonso semi-analytical method for the same MTOW).	Independent approaches agree within their stated method uncertainty; disagreement is documented and traced to known method limitations.

11.3. Verification & Validation Campaign for the Final Design Codebase

This subsection covers how the final design code is verified and validated.

The test catalogue from Section 11.1 is applied bottom up, using the block as the test unit. A group of functions behind one entry point is defined as a block. Each block is tested through its interface, which indirectly checks its helper functions. Here the module level matches the block level, so it is not tested separately. The system (S) and validation (V) levels are unchanged.

Block and system tests run as an automated pytest suite, so the regression test S2 runs continuously. The naming convention from Table 11.1 is kept (e.g. **EN-U8-1a**), where "unit" now denotes a block.

Validation uses a complete Fusion 360 model of the aircraft (V2). The reference production aircraft comparison (V1) is dropped, since the Prandtl plane configuration cannot be reliably compared to any existing eVTOL.

Table 11.5 lists the block units, each identified by its entry point, together with the internal functions verified

through it.

Table 11.5: Block level decomposition of the final design codebase. Each block unit is tested.

ID	Block unit	Functions covered
MT	converged_mass / compute_mtow	full MTOW iteration
ST-FUS	fuselage_mass	load conditions, shell beam gauge, buckling/yield
ST-WG	wing_geometry, wing_mass	spar/skin sizing, winglet sizing, geometry
ST-SEC	tail_mass, hinge_mass, misc_mass	tail/hinge/misc mass
PP	propulsion mass & performance	motor_mass, propeller_mass, hub_mass, PPE
LG	landing_gear_mass	size_all_architectures, scoring, concept models
EN	mission_energy + battery_mass	segment power functions, cell-count chain
MI	aircraft_inertia	shape closed forms, rotation, parallel axis
PR	pressurization_sizing	ISA, shell penalty, air supply
AX	energy_breakdown + ECS sizing	per load build up, bus split, compressor
AE	drag_polar + feasible_wing_loading_range	polar, matching constraints, calibration
SA	evaluate_stability	derivatives, x_{np} , SM, control margins
DS	dynamic stability analysis	state space build, eigenvalues, mode metrics
NO	aeroacoustic analysis	rpm, fundamental frequency, tip Mach
OP	optimiser evaluation	objective/constraint eval, parameter persistence
VD	vn_diagram, turn performance, climb, payload range	V-n / turn / climb / payload-range diagram construction

Table 11.6 gives the unit test types implemented per block unit. ✓ denotes a test already implemented. • denotes a test to be written. V2 marks the block units whose mass, geometry, c.g. or inertia outputs are cross validated against the Fusion 360 model.

Table 11.6: Verification and validation to be performed on the final design codebase. Block-unit rows carry the unit tests (U1–U10); the system tests (S1–S4) and the higher fidelity validation (V2) act on the complete integrated system. ✓ = implemented (run & log); • = to be implemented.

Test target	U1	U2	U3	U4	U5	U6	U7	U8	U9	U10	S1	S2	S3	S4	V2
MT – MTOW loop				✓			✓	✓		✓					
ST-FUS – fuselage	✓		✓	✓	✓		✓		✓	✓					
ST-WG – wing/winglet	✓			✓	✓		✓		✓	✓					
ST-SEC – secondary masses	✓		✓	✓	✓		✓			✓					
PP – propulsion	✓	✓	✓		✓		✓	✓	✓	✓					
LG – landing gear	✓			✓	✓		✓	✓	✓	✓					
EN – energy & battery	✓	✓	✓	✓	✓		✓	✓							
MI – mass & inertia				✓	✓		✓	✓	✓	✓					
PR – pressurization		✓	✓	✓	✓	✓	✓	✓		✓					
AX – aux loads & ECS	✓	✓	✓		✓		✓	✓	✓	✓					
AE – aerodynamics	✓	✓	✓		✓		✓	✓	✓	✓					
SA – stability & control	✓	✓	✓						✓	✓					
OEI – hover stability	✓		✓	✓			✓	✓		✓					
DS – dynamic stability	✓				✓	✓	✓	✓	✓	✓					
NO – noise / acoustics	✓	✓				✓	✓			✓					
OP – optimiser	<i>orchestration block, verified via its constituent blocks</i>														
VD – performance diagrams	✓	✓	✓	✓	✓		✓	✓	✓						
System (integrated pipeline)	<i>system level</i>										✓	✓	•	✓	✓

S1 = sensitivity, S2 = regression, S3 = system validation, S4 = stress. V2 = validation of the complete system against higher fidelity models: finite element (FEM) structural analysis, the Fusion 360 CAD model, and ANSYS CFD.

The optimiser block (OP) is an orchestration layer that evaluates designs by invoking the already-verified sizing and stability blocks; it therefore carries no unit tests of its own, and its behaviour is exercised at the system level through the sensitivity (S1) and regression (S2) tests on the converged design.

Because block level testing makes it harder to pinpoint the source of a fault, a few detailed tests are kept on the highest risk governing equations, so a failure can still be traced to a specific formula. These are formula inspection (U1) and hand reference (U5) tests on the fuselage shell buckling gauge (ST-FUS), the wing spar cap root moment (ST-WG), the mission energy segment build up (EN), and the neutral point and Helmbold lift slope expressions (SA).

Once the blocks are verified, the full pipeline is run through the implemented system tests (S1, S2 and S4). System validation against a reference aircraft (S3) is identified as future work for the detailed build phase.

S1 (sensitivity) runs a Sobol/tornado analysis over the design variables (both $\pm 10\%$ and global ranges) to confirm the overall conclusions are robust. **S2 (regression)** freezes the converged design, namely the MTOW and its full component mass breakdown, as a baseline, and checks that a fresh run reproduces it within 1%, so that an inadvertent change to any formula or constant is caught. **S4 (stress)** runs degraded and off-design inputs (battery at 80% of beginning-of-life energy density, and a heavier off-design payload), and requires the outputs to stay finite, physically reasonable, and to change in the expected direction.

Validation is performed by cross comparison against a set of independent, higher fidelity models, each applied to the part of the aircraft it is best suited to. A complete Fusion 360 CAD model provides the mass properties: every component mass, volume, c.g. position and inertia contribution is computed from solid geometry and material density, and the codebase's predictions for the fuselage, wings, winglets, tail, landing gear, propeller blades, hubs and battery box are compared against it. The aerodynamic block is validated against ANSYS CFD. The primary structure is validated by finite-element (FEM) analysis: the analytical Class-II spar, skin and shell sizing of the fuselage and wing/winglet is checked against FEM stresses and deflections under the design loads. In each case the pass criterion is agreement within the stated method tolerance, with any larger deviation documented and traced to a modelling assumption. This comparison is intended to surface discrepancies rather than merely confirm agreement: for example, the analytical propeller-blade regression predicts 3.46 kg per blade against the Fusion CAD value of 1.46 kg, a divergence flagged by V2 and resolved by adopting the geometry based CAD value, while the CFD and FEM checks similarly identify where the low-order aerodynamic and structural models require correction or recalibration.

The cross-model element of V2 is implemented in the automated suite: the class-II analytical models reproduce the optimiser's independently-converged design (battery mass to within 0.5%, wing area and aspect ratio to within 0.3%), validating the two pipelines against each other. The higher-fidelity CAD/CFD/FEM comparisons remain detailed-design-phase work.

Every executed test is recorded as a test card (Test ID, Input, Expected Output, Actual Output, Pass Condition, Pass/Fail) under the SUBSYS-U#-order convention, "U" denoting a block-unit test. Three limitations are noted: the campaign provides block level coverage, so a failure localises to a block rather than an individual function (mitigated by the targeted U1/U5 tests above); coverage is by test type rather than exhaustive line by line execution, so untested internal branches may remain; and the strength of the V2 validation is conditional on the Fusion 360 model being a faithful, fully defined representation of the design

Table 11.7: Summary of Verification Tests by Script

Script Name	Unit Tests (U)	Module Tests (M)	Pass Rate
ST-FUS – Fuselage mass (mass_components.fuselage_mass)	5	0	100%
ST-WG – Wing/winglet mass (wing_geometry, wing_mass)	5	0	100%
ST-SEC – Secondary masses (tail_mass, hinge_mass, misc_mass)	4	0	100%
MT – MTOW aggregation loop (mtow_sizing: compute_mtow, converged_mass)	5	0	100%
AE – Aerodynamics (dragpolar.py, matching_diagram.py, aero_model.py, airfoil_characteristics.py, h_b_lift_ratio.py)	14	0	100%
EN – Mission Energy & Battery (class_II_sizing/energy.py)	12	0	100%
AX – Auxiliary Loads & ECS (aux_loads.py, ecs_sizing.py)	18	0	100%
VD – Performance Diagrams (vn_diagram, turn, climb, payload-range)	14	0	100%
VD – Longitudinal Static Stability (stat_long_stab_anal_func.py)	8	0	100%
VD – VTOL OEI CG Envelope (VTOL_cg_envelope_det.py)	8	0	100%
PP – Propulsion Performance (ppe.py)	22	0	100%
NO – Noise functions in ppe.py (noise_spl, pwl_regression, pwl_to_spl_directional, spl_to_dba)	10	0	100%
DS – Dynamic Stability (test_dynamic_stab.py)	8	0	100%

As a representative example, the executed test cards for the structures and mass blocks are reproduced in Table 11.8; the remaining blocks follow the same test-card format, and their full test counts and pass rates are given in Table 11.7.

Table 11.8: Executed verification test cards for the structures blocks (MT, ST-FUS, ST-WG, ST-SEC). ★ marks the load-bearing tests to retain if the table must be shortened.

Test ID	Input	Expected Output	Actual Output	Pass Condition	P/F
ST-FUS – Fuselage mass (mass_components.fuselage_mass)					
★ ST-FUS-U1-1a	MTOW = 2000 kg	GD/Raymer regression formula	92.60 kg = formula	Exact match	Pass
ST-FUS-U3-2a	MTOW = 0	0 (mass $\propto$ MTOW ^{0.144})	0.0 kg	= 0	Pass
ST-FUS-U7-3a	MTOW 1700/2200, 10 t	Increasing, finite	90.5 < 92.6; 116.7 kg @ 10 t	Monotonic, no overflow	Pass
ST-FUS-U9-5a	MTOW = 2000 kg	Mass fraction in sane band	4.6% of MTOW	2–12%	Pass
ST-FUS-U10-4a	MTOW = 2000 kg	Finite positive scalar	92.60 kg (float)	Interface contract	Pass
ST-WG – Wing/winglet mass (wing_geometry, wing_mass)					
★ ST-WG-U5-1a	MTOW = 2000 kg	$S = W/(W/S)$	25.82 m ²	Exact	Pass
★ ST-WG-U1-2a	MTOW = 2000 kg	tip = λ root, S_{half} = split-S, $A = b^2/S$	tip 0.62 = 0.45(1.37); $A = 6.55$	Exact	Pass
ST-WG-U7-3a	MTOW 1700/2200, 10 t	Increasing, finite	107.9 < 123.7; finite @ 10 t	Monotonic	Pass
ST-WG-U9-5a	MTOW = 2000 kg	Mass fraction in sane band	6.2% of MTOW	3–15%	Pass
ST-WG-U10-4a	MTOW = 2000 kg	Finite positive scalar	123.7 kg (float)	Interface contract	Pass
ST-SEC – Secondary masses (tail_mass, hinge_mass, misc_mass)					
★ ST-SEC-U1-1a	MTOW = 2000 kg	hinge = 0.05 MTOW	100.0 kg	Exact	Pass
ST-SEC-U1-2a	MTOW = 2000 kg	misc = 0.10 MTOW + 32	232.0 kg	Exact	Pass
ST-SEC-U3-3a	MTOW 0 / 1700 / 2200	Aero floor > 0, monotonic	tail(0) = 19.96 kg; rising	Floor + monotonic	Pass
ST-SEC-U10-4a	MTOW = 2000 kg	tail/hinge/misc finite > 0	23.9 / 100 / 232 kg	Interface contract	Pass
MT – MTOW aggregation loop (mtow_sizing.compute_mtow)					
★ MT-U8-1a	compute_mtow(2200)	MTOW = Σ components	1965.28 kg = Σ	Exact (float tol)	Pass
MT-U7-3a	Input 1700 vs 2200	Estimate increases	1698.73 < 1965.28 kg	Monotonic	Pass
MT-U4-4a	Input 3500 kg	Finite, ordered	2592.84 kg	No NaN/overflow	Pass
MT-U10-2a	compute_mtow(2200)	All component lines finite > 0	11 masses, all > 0	Interface contract	Pass

11.4. Requirements Verification

In this section, the requirements fulfilment will be assessed by means of a compliance matrix. This matrix will check whether all user requirements as specified in [56] are met. It will state the requirement ID, the requirement itself, the verification method, whether the requirement has been met and the rationale behind it. The verification method has been based on [57] and includes: inspection, analysis, demonstration and testing. Since the design is only in the conceptual stage, only inspection and analysis are feasible verification methods. If the requirement is not met, however, demonstration and testing can also be mentioned as future verification methods. The compliance matrix can be found in Table 11.9. In addition, Table 11.10 will show the future steps that need to be taken to ensure compliance with all requirements that have not been met.

Table 11.9: Engineering compliance matrix for the top-level project requirements.

ID	Requirement	Verification method	Met?	Rationale
Performance				
REQ-MS-01	The vehicle shall be able to transport a minimum payload of 400 kg.	Analysis	✓	The whole design is based around a payload of 400 kg.
REQ-SYS-02	The vehicle shall fit within a maximum on-ground envelope of $10 \times 6 \times 3 \text{ m}^3$.	Inspection	X	The total height of the vehicle will be 4 m, the width 6.2 m.

Continued on next page

ID	Requirement	Verification method	Met?	Rationale
REQ-MS-08	The vehicle shall achieve a cruise speed of at least 200 km/h.	Analysis	✓	The propulsion system has been sized to achieve this, as seen in Section 5.1
REQ-MS-10	The vehicle shall comply with applicable European noise-emission regulations.	Testing	X	The maximum noise generated is 94 dBA, as seen in Table 5.9
REQ-MS-03	The vehicle shall achieve a range of at least 200 km plus reserve.	Analysis	✓	The vehicle has been designed with a mission 200 [km] in mind and a reserve of 40 [km].
REQ-STK-14	The vehicle lifetime shall exceed 10 years.	Demonstration	X	This could not be determined yet, as the effect of age has not been assessed.
Safety				
REQ-SUB-55	The vehicle's flight management system shall include a certified auto pilot.	Inspection	✓	The Collins Aerospace FMC-4212 flight management computer [58] will be used.
REQ-MS-26	The vehicle shall be able to operate at a distance of at least 5 m from people and 3 m from objects, including in densely populated areas.	Analysis	X	This requires detailed simulation of the vehicles flight, which is outside the scope of the DSE.
REQ-SYS-20	The vehicle shall provide a maximum cabin noise level of 60 dBA.	Testing	X	For this, cabin noise analysis is needed. This is deemed outside the scope of the DSE.
REQ-MS-05	The vehicle shall be allowed and equipped to fly overnight.	Inspection	✓	The vehicle includes lights to be visible at night
REQ-MS-14	The vehicle shall be able to safely abort operations in wind conditions up to Beaufort 8.	Demonstration	X	For this, a prototype would need to be build to fully demonstrate compliance. However, in cruise configuration, the aircraft can handle gusts of 8 Beaufort as shown in Subsection 6.4.2.
REQ-MS-15	The vehicle shall be operable in heavy rain.	Demonstration	X	For this, detailed analysis on the effects of rain on the vehicle has to be performed. This is deemed outside the scope of the DSE.
REQ-SYS-21	The propulsive system shall allow continued flight with at least 75% of the propulsion system actively working.	Analysis	✓	The aircraft has been designed so that any of the engines can fail and continued flight in hover and forward flight is still possible.
Costs and Sustainability				
REQ-MS-19	The final product cost shall not exceed 4 MEUR.	Analysis	✓	As shown in Section 14.4, the initial costs of Volde will be 3 MEUR estimated in financial year 2026.
REQ-MS-17	The vehicle main structure shall be fit for reuse in other vehicle components for at least 10 years at the vehicle end-of-life.	Demonstration	X	This could not be determined yet, as the effect of age has not been assessed.
REQ-SYS-29	The vehicle shall use modular structural components designed for disassembly and reuse.	Analysis	X	For this, a detailed assembly plan for Volde needs to be designed. This is deemed outside the scope of the DSE
REQ-MS-18	After reuse, the main structure shall be recyclable.	Inspection	X	Composite materials are used in the structure, making it harder to recycle.

Table 11.10: *Future steps to ensure compliance with the requirements that are not met.*

ID	Future steps to ensure compliance
Performance	
REQ-SYS-02	This requirement was not met, as the aircraft is too high and too wide. It is too high due to ground clearance issues with the current wing position in folded configuration. In order to decrease this, the front wing would need to move up. The effect of this, can be quite substantial, as explained in Chapter 4 and will influence other design aspects as well. In terms of the width, the aircraft turned out too wide because of the current propeller location. This location was chosen to ensure proper folding of the wings and less obstruction in the flow of the propeller. In order to decrease both the height and the width, more design iterations would be necessary to meet this requirement.
REQ-MS-10	In order for this requirement to be met, a more detailed external design for the reduction of noise needs to be made. For example, ducted propellers can be a solution. However, extensive testing would need to be performed, in order to be sure that this requirement is met.
REQ-STK-14	It cannot yet be determined whether this requirement is met, as the effect of age has not been assessed. In order for this to be analysed a detailed fatigue analysis on the aircraft will need to be performed. However, fatigue testing of the prototype would be necessary to ensure compliance.
Safety	
REQ-MS-26	For this requirement to be met, a detailed simulation of the vehicles flight within urban space would need to be made.
REQ-SYS-20	In order for this requirement to be met, a more detailed internal design for the the reduction of noise needs to be made. For example, the fuselage lining needs to be considered. However, extensive tesing would need to be performed, in order to be sure that this requirement is met.
REQ-MS-14	A full wind analysis was not part of this DSE. However, in order to truly comply with this requirement, a prototype would need to be build that demonstrates Vold-e's capability of flying in windy conditions in all flight phaess.
REQ-MS-15	For this, an analysis of the effects of rain will need to be performed after the DSE. Subsequent testing would then show whether the aircraft is truly waterproof and whether it can still fly safely in heavy rain.
Costs and Sustainability	
REQ-MS-17	Similar to REQ-STK-14, it cannot yet be determined whether this requirement is met. More detailed fatigue analysis and testing would be necessary.
REQ-SYS-29	For this requirement, it also cannot yet be determined whether is it is met. For this, a detailed assembly plan would need to be made, including the types of bonds and fasteners that will be used in assembly.
REQ-MS-18	As composite components are used, the aircraft is harder to recycle. In order to meet this requirement, more research would need to be performed on other feasible materials that are easier to recycle.

11.5. Discussion

The verification and validation campaign provides structured evidence that the conceptual design models used for Vold-e are implemented consistently and produce robust outputs within their intended application range. The bottom-up testing approach, from individual design blocks to integrated sensitivity, regression and stress tests, ensures traceability throughout the final design process. All implemented automated verification tests achieved a (100%) pass rate, while the regression tests provide a mechanism for detecting unintended changes to the converged design baseline during further development.

Validation was additionally supported through comparison with independent, higher-fidelity design tools. The Fusion 360 model was used to verify the geometric feasibility of the folding mechanism and to cross-check mass properties. Furthermore, ANSYS Fluent CFD was used to assess the aerodynamic performance of the integrated aircraft (Section 10.1) and finite-element analysis was applied to the hinge design to identify critical stress regions and evaluate its structural feasibility (Section 10.2). Together, these analyses provide confidence that the main aerodynamic, geometric and structural outputs of the preliminary design are physically plausible and not based solely on low-order analytical models.

Nevertheless, the present V&V campaign remains limited by the conceptual maturity of the aircraft. The CFD, CAD and finite-element models still rely on modelling assumptions and have not yet been validated against experimental measurements or a physical prototype. In addition, several top-level requirements, including the ground-envelope, noise, lifetime, all-weather operability and end-of-life reuse requirements, cannot yet be fully demonstrated. Further work should therefore focus on detailed component modelling, experimental validation, fatigue testing, full-system integration tests and prototype-based verification before Vold-e can progress towards certification.

12 Technical Risk Assessment and Safety Concerns

This chapter assesses the principal technical risks and safety considerations associated with the Vold-e design. Section 12.1 identifies the main technical risks, defines mitigation and contingency measures, and evaluates the resulting residual risks. Section 12.2 assesses the reliability, availability, maintainability, and safety characteristics of the aircraft, including redundancy measures, maintenance planning, and safety-critical functions. Finally, Section 12.3 summarises the main findings and identifies the further analyses required to support a more detailed safety and certification assessment.

12.1. Technical Risk Assessment

The technical risk assessment identifies and manages uncertainties that could negatively affect the schedule, technical performance, or cost of the project. This final assessment builds on the list of risks established in the Baseline Report [28] and on the risk updates made in the Midterm Report [30]. It will be presented as an update log on these risks rather than as a complete risk register. The risks are grouped into six categories: system integration (SI), aerodynamics and aeroacoustics (AA), power and propulsion (PP), structures (ST), vehicle dynamics and control (VDC), and business and operations (BO). For each risk, a likelihood (L) and consequence (C) are assigned both before and after mitigation. These values are qualitative estimates on a continuous scale from 0.0 to 1.0. They are not based on statistical data, as this is not available at this design stage, but rather on engineering judgement.

The likelihood represents the estimated probability that the identified risk will occur at some point during vehicle development, while the consequence represents the severity of the impact on the design, schedule, or requirements. The interpretation of these values is summarised in Table 12.1.

Table 12.1: Interpretation of likelihood and consequence values used in the technical risk assessment.

Value	Likelihood interpretation	Consequence interpretation
0.1	Very unlikely	Negligible impact
0.3	Unlikely but plausible	Minor performance loss, addressable with local rework
0.5	Roughly even chance of occurring	Requirement partially missed, subsystem-level redesign required
0.7	More likely than not	Major requirement missed, multi-subsystem redesign required
0.9	Expected to occur unless actively mitigated	Driving requirement violated, project objective at risk

12.1.1. Identification of Technical Risks, Risk Consequences and Likelihood

Table 12.2 presents the final technical risk update log. The original risk IDs from the Baseline Report [28] and the updated ID's from the Midterm Report [30] are retained to keep the risks traceable

Table 12.2: Final technical risk update log: identification and initial assessment

ID	Change type	Description	Reference requirement	L	C	Responsible member
System Integration						
TR-SI-UP-03	Updated	The folding mechanism cannot operate because it conflicts with structural components, propellers, landing gear, or passenger access.	REQ-SYS-02, REQ-SYS-12, REQ-SYS-44	0.6	0.9	Ivan
TR-SI-UP-07	Updated	The propulsion layout cannot be integrated with the folding wing, including clearance, and folding motion.	REQ-SYS-12, REQ-SYS-21, REQ-SYS-36	0.6	0.8	Luca
TR-SI-UP-08	Updated	Access for inspection, replacement, or maintenance of critical components is blocked or inconvenient due to the integrated layout.	REQ-SYS-32	0.5	0.5	Luca
Aerodynamics and Aeroacoustics						
TR-AA-UP-01	Updated	Propeller slipstream and wing-wake interaction cause excessive drag, flow separation, or thrust losses during transition.	REQ-SYS-13, REQ-SYS-14	0.6	0.6	Wiktoria
TR-AA-UP-03	Updated	The aerodynamic forces and moments during the folding transition are inaccurately modelled.	REQ-SYS-12	0.7	0.7	Luca
TR-AA-UP-04	Updated	The hover-cruise transition causes unstable or uncontrollable aircraft behaviour.	REQ-SYS-11, REQ-SYS-12	0.5	0.8	Wiktoria

ID	Change type	Description	Reference requirement	L	C	Responsible member
Power and Propulsion						
TR-PP-UP-07	Updated	The propulsion layout exceeds the acceptable ground footprint or interferes with the folding mechanism.	REQ-SYS-02, REQ-SYS-12	0.5	0.7	Björn
TR-PP-UP-12	Added	Electrical, signal, or cooling routing through moving wing sections is difficult to integrate safely.	REQ-SYS-21, REQ-SYS-36, REQ-SYS-40	0.5	0.7	Eren
Structures						
TR-ST-UP-02	Updated	The transforming wing hinges cannot withstand deployment, hover, transition, cruise, landing, and gust conditions.	REQ-SYS-22, REQ-SYS-23, REQ-SYS-25	0.7	0.9	Antonio
TR-ST-UP-03	Updated	The folding mechanism must provide sufficient stiffness without causing an excessive mass penalty.	REQ-SYS-01, REQ-SYS-22, REQ-SYS-23, REQ-SYS-25	0.6	0.7	Utku
Business and Operations						
TR-BO-UP-02	Updated	Take-off, landing, folding, unfolding, boarding, charging, or ground handling does not fit within the required ground operation footprint.	REQ-SYS-02, REQ-SYS-38	0.5	0.8	Björn
TR-BO-UP-05	Updated	Maintenance requirements and costs are higher than expected due to complex hinges, locks, actuators, or moving wing sections.	REQ-SYS-32, REQ-SYS-33	0.7	0.6	Carlijn

12.1.2. Suitable Mitigation Measures, Contingency Plan and Post-Mitigation Risk

Table 12.3 presents the proposed mitigation action and contingency plan for each updated/added risk, in addition to post-mitigation values for the likelihood (L-Post) and the consequence (C-Post).

Table 12.3: Final technical risk update log: mitigation, contingency and residual risk

ID	Mitigation action	Contingency plan	L-Post	C-Post
System Integration				
TR-SI-UP-03	Develop and maintain a shared CAD model; reserve folding and unfolding motion envelopes for wings, propellers, landing gear, and access zones after each major design change.	Modify the folding path, relocate obstructing components or locally redesign the surrounding structure.	0.2	0.6
TR-SI-UP-07	Integrate propulsion placement and folding geometry together; define propeller clearance, wiring routes, support loads, and folding requirements..	Change propeller position, adjust the number or size of propeller, reroute wiring, or simplify the folding-wing architecture.	0.3	0.5
TR-SI-UP-08	Include maintainability checks for hinges, locks, actuators, and batteries; reserve access panels for critical components.	Add removable panels, redesign component mounts to improve access, or accept longer maintenance time in the operational assumptions.	0.3	0.3
Aerodynamics and Aeroacoustics				
TR-AA-UP-01	Analyse propeller-wing interaction during transition using simplified CFD; maintain sufficient slipstream clearance.	Reposition propellers, adjust transition schedule, modify wing geometry, or add aerodynamic fairings to reduce interference losses.	0.4	0.3
TR-AA-UP-03	Validate transition aerodynamic forces and moments with CFD before finalising the folding schedule.	Redesign the transition schedule, reduce the transition speed range, or apply a more conservative stability assessment.	0.3	0.5

ID	Mitigation action	Contingency plan	L-Post	C-Post
TR-AA-UP-04	Define a safe transition corridor and evaluate stability and control margins throughout the complete folding manoeuvre.	Limit transition conditions, perform transition more gradually, or revise the control and propulsion strategy to maintain stability and controllability.	0.3	0.5
Power and Propulsion				
TR-PP-UP-07	Check propeller diameter, clearance, and folding envelopes against the ground-footprint constraint after every major propulsion or layout change.	Reduce propeller diameter, relocate propellers, or modify the folding sequence.	0.2	0.4
TR-PP-UP-12	Define routing paths for wiring, signal cables, and cooling lines through moving sections.	Reroute cables through fixed structure, add flexible protected routing, or relocate electrical components.	0.2	0.4
Structures				
TR-ST-UP-02	Define hinge load cases for deployment, hover, transition, cruise, landing, and gust conditions; size them using conservative safety factors.	Reinforce or redesign the hinge load path, reduce transformation loads, or switch to a simpler transformation mechanism.	0.3	0.6
TR-ST-UP-03	Evaluate the stiffness-to-mass ratio of the folding mechanism; include lock stiffness and hinge flexibility in the structural model.	Redesign the folding mechanism, or accept a mass increase and redistribute the structural mass budget.	0.3	0.4
Business and Operations				
TR-BO-UP-02	Check the complete take-off, landing, folding, unfolding, boarding, charging, and ground-handling envelope against the required ground footprint.	Modify the operating procedure, restrict folding or unfolding to designated areas, or revise the layout.	0.3	0.5
TR-BO-UP-05	Include hinges, locks, actuators, and moving wing sections in the maintenance schedule and operational cost estimate.	Increase maintenance allowance or replace complex mechanisms with simpler alternatives.	0.3	0.4

12.1.3. Risk Maps

The technical risks are visualised using pre- and post-mitigation risk maps, shown in Figure 12.1a and Figure 12.1b, respectively. In both figures, each identified risk is represented by its likelihood and consequence. Furthermore, a colour scale is introduced, such that risks with a higher probability of occurrence and a more severe impact (red colour) can be distinguished more easily from lower-priority risks (green colour). The likelihood and consequence values are mapped to the five qualitative map bins using equal width intervals: *Very Low*: [0.0, 0.2), *Low*: [0.2, 0.4), *Medium*: [0.4, 0.6), *High*: [0.6, 0.8), *Very High*: [0.8, 1.0].

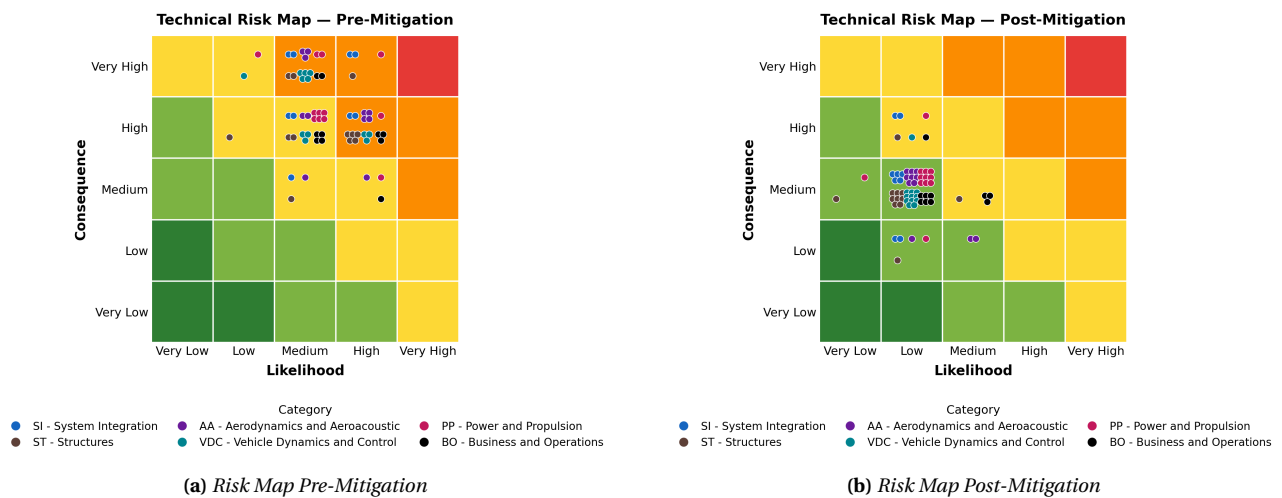


Figure 12.1: Risk maps before and after mitigation

Overall, the mitigation strategy reduces the most critical risks to an acceptable preliminary design level.

12.2. RAMS Characteristics

In this section, the RAMS characteristics of the aircraft will be explained. Firstly, the reliability requirements on the eVTOL will be discussed and a way to determine the reliability of the eVTOL will be established. Subsequently, the maintenance of the eVTOL will be discussed and the resulting availability estimated. Lastly, the safety of the eVTOL will be addressed and a list of safety critical functions will be established.

12.2.1. Reliability

Reliability is mainly determined by *REQ-STK-07*: The vehicle shall be safe to be used for transporting passengers. This requirement makes that the vehicle should comply with the EASA VCA [59] regulations, more specifically, the VTOL.2510 regulations on equipment, systems and installations. For this analysis, the focus will be on catastrophic failures. Given that the envisioned aircraft falls under the enhanced category ¹, the regulations require a probability of each catastrophic failure of less than 10^{-9} per flight hour. In addition, the regulations also mention that: "In any system or subsystem, the failure of any single element, component, or connection during any one flight should be assumed, regardless of its probability. Such single failures should not be catastrophic." [59], meaning that single point failures thus may not lead to a catastrophic failure. Easa defines a catastrophic failure in MOC VTOL.2510 as: "[...] failure conditions, which are expected to result in one or more fatalities, or incapacitation of a flight crew member, usually with the loss of the aircraft. Failure conditions that would prevent continued safe flight and landing of the aircraft are also considered catastrophic" [59]. In total, there are thus two important regulations to comply with: the probability of each catastrophic failure shall be less than 10^{-9} per flight hour and no single point failure shall lead to a catastrophic failure.

Firstly, the single point failures that can lead to a catastrophic failure are identified, and a redundancy measure is applied. In order to do this, the vehicle can be broken down into subsystems. The subsystems used in this section are largely similar to the ones used in the technical risk assessment, presented in Section 12.1. In contrast to the technical risk assessment, however, the communication system is also added as a subsystem. This is due to the fact that this system is deemed critical to the envisioned eVTOL. The subsystems and potential single point failures that can occur are listed in Table 12.4. Some of the single point failures are inspired by the risks identified in the technical risk assessment. When this is the case, the corresponding technical risk is referred to between brackets.

Table 12.4: Possible single point failures and the redundancy measures applied

Subsystem	Possible single point failures	Redundancy measures
Aerodynamics	A critical control surface fails or is unable to generate the needed force	Integrate a redundant set of control surfaces
Power & propulsion	Failure of the battery will leave all systems without power	Add auxiliary battery
	Battery management system fails	Add a backup battery management system
Structures	Failure of the hinge actuators will make the wings unable to (un)fold	Each wing has its own actuator, making for a total of four actuators. Given that they are connected with the winglets, one actuator will be able to (un)fold both wings.
Control system	The flight controller / computer fails (TR-VDC-05)	Add three redundant flight controllers
	One of the flight critical sensor fails (refer to Figure 3.8)	Have a redundant set of critical sensors
Communication system	Communication to the ground station fails when aircraft cannot operate autonomously (TR-VDC-10, TR-VDC-12)	Use a redundant communication system (cellular and satellite (refer to Section 3.2)) & implement an emergency landing procedure

Now that the single point failures are discussed, the total reliability of the vehicle can be determined based on a forecast of catastrophic failures. This forecast will be based on the single point failures as described above, with the redundancy measure applied. The relation between the reliability of subsystems and the reliability of the entire eVTOL can be visualised in a block diagram, as explained in [60]. The diagram for the envisioned eVTOL is shown in Figure 12.2. In this diagram, the blocks in yellow show subsystems, the blocks in red show failure modes and the blocks in orange show the related reliability. Some subsystems need extra explanation, these will be explained below:

1. **Propulsion and power** : two engines failures are included as according to the requirement REQ-SYS-21, the aircraft shall be able to continue flight with 75% of the propulsion system working. Given that there will be six engines, two engines need to fail in order for a catastrophic failure to happen.
2. **Parachute system** : It is envisioned that in case of any failure that would be catastrophic, the parachute can deploy. This would not help for certification, as it does not ensure continued flight, however, it would save the

¹ Electric VTOL News, <https://evtol.news/news/easa-publishes-sc-vtol>, Accessed 19-05-2026

passengers.

3. **Communication subsystem:** This subsystem was dropped as a loss of communication would not cause a catastrophic failure once all redundancy measures, specified in Table 12.4, are taken.

In terms of reliabilities, parallel blocks mean that they are redundant and thus increase the reliability of the aircraft. A risk in series mean that a failure in the chain leads to an entire system failure, and thus makes the eVTOL less reliable. As such, all subsystems are linked in series, whereas redundant components within the subsystems are linked in parallel. The reliabilities in series and parallel can be calculated as shown in Equation 12.1 [60].

$$R_{series} = R_1 \cdot R_2 \cdot \dots \cdot R_n, \quad R_{parallel} = 1 - ((1 - R_1) \cdot (1 - R_2) \cdot \dots \cdot (1 - R_n)) \quad (12.1)$$

Looking at the entire diagram, this gives the formula as shown in Equation 12.2.

$$R_{system} = (1 - (1 - R_{cs1})(1 - R_{cs2})) \cdot (1 - (1 - R_{e1})(1 - R_{e2})) \cdot (1 - (1 - R_{b1})(1 - R_{b2})) \cdot (1 - (1 - R_{bs1})(1 - R_{bs2})) \cdot (1 - (1 - R_{h1})(1 - R_{h2})) \cdot (1 - (1 - R_{fc1})(1 - R_{fc2})(1 - R_{fc3})) \cdot (1 - (1 - R_{sf1})(1 - R_{sf2})) \quad (12.2)$$

$$R_{Total} = 1 - (1 - R_{System})(1 - R_{par})$$

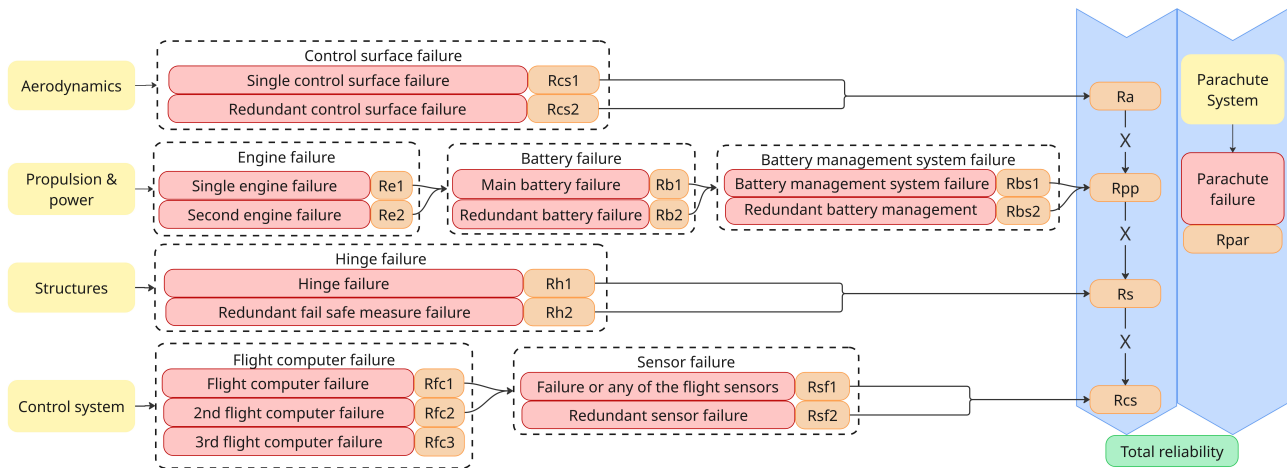


Figure 12.2: Reliability block diagram

In order to get a numerical value out of this formula, the reliability of all entries must be determined. In Table 12.5 the failure rate of all components that are currently known are given.

Table 12.5: Known failure rate values for subsystem components

Reliability component	Failure rate value	Note
$R_{cs1} = R_{cs2}$	$10^{-6} - 10^{-7}$	EASA gives this order of magnitude as an indication in the CS-25 regulations, under AMC 25.671 [53]. It is established from experience with in-service airplanes. However, it must be noted that EASA also mentions not to use this number for a probabilistic assessment.
$R_{e1} = R_{e2}$	9.26×10^{-6} per hour	Value extracted from [61].
R_{b1}	9.31×10^{-5} per hour	Taken from [62].
R_{b2}	$1 * 10^{-6}$	This reliability is based on the mean time between failure of the Socomec MODULYS XM UPS (Uninterruptible Power Supply) ¹ .
$R_{bs1} = R_{bs2}$	$1 * 10^{-7}$ per hour	This value was extracted from research [63] and based on ISO26262 ASIL-C automotive safety goals. The value reported was 100 failures in 10^9 hours. This was converted to a failure rate per hour.
$R_{h1} = R_{h2}$	$0.5 * 10^{-5}$	As no mean time between failure for the chosen actuator could be found, the Linak LA76 has been chosen as reference ² . The website states that it can operate on max load for a minimum of 200 000 cycles. Assuming 1 cycle per flight, the mean time between failure would be 200 000 hours. This leads to the stated failure rate.

Continued on next page

Table 12.5: *Known failure rate values for subsystem components – continued*

Reliability component	Failure rate value	Note
$R_{fc1} = R_{fc2} = R_{fc3}$	2.6×10^{-4} per hour	Based on the mean time between failures of >3800 hours of the Collins Aerospace FMC-4212, found in the product brochure [58]. In addition it is assume that all three redundancy measures are the same.
$R_{sf1} = R_{sf2}$	1×10^{-5} per hour	Based on the mean time between failures of the Thales High Performance Inertial Reference System ³ .
R_{par}	Very low	In the guide to the Cirrus Airframe Parachute System (CAPS) [64], it was stated that in the 85 times that CAPS was activated as of 2014, no fatalities occurred when it was activated within the conditions as specified in the guide. This makes for a very low failure rate estimate, but assuming a failure rate of 0 is deemed too optimistic. Detailed testing of the system that will be implemented should demonstrate the exact failure rate.

Using these values and excluding the parachute, a preliminary reliability value can be calculated using Equation 12.2. In order to do so, all failure rate values have been converted to reliabilities using Equation 12.3 [65], where λ is the failure rate per unit of time t . In order to use this formula, a constant failure rate is assumed [65]. In addition, when a failure rate bound is given Table 12.5, the highest value is used. This gives a preliminary reliability value of the eVTOL, that can be converted using Equation 12.3 again to a failure rate value. This gives a catastrophic failure rate of 3.2×10^{-10} per hour. This value would be low enough for certification, as in order for this value to be lower than 1×10^{-9} , the failure rate of each subsystem must also be lower than 1×10^{-9} , as all subsystems are in series. However, this value is also a bit misleading. In this analysis, failure of the battery sliding mechanism is not taken into account, as well as other failure modes depending on the integration of the different subsystems. Additionally, in terms of failing sensors, only the IMU is taken into account. Lastly, this block diagram is far from exhaustive: a study from Boeing [61], contracted by NASA, has shown that the failure rate per flight hour of the propulsion system of a full electric quad rotor eVTOL concept would be in the order of $10^{-3} - 10^{-4}$. In spite of the fact that the envisioned eVTOL would have six rotors, this value still demonstrates a large discrepancy with the estimated value. It can thus be said, that the real failure rate of Vold-e would be quite a bit higher. As such, more failure modes must be assessed to assess whether the aircraft can indeed be certified.

$$R = e^{-\lambda t} \quad (12.3)$$

12.2.2. Availability and Maintainability

In terms of availability, REQ-STK-15 is driving, which states that the vehicle shall be able to perform at least six stretches of the proposed mission per day. Given that the route from Courchevel to nice is expected to take approximately one hour and add to this a maximum turnaround time of one hour as specified by REQ-MS-28, this makes for 12 hours of availability per day. This leaves 12 hours per day for possible maintenance. In a study performed by Naru and German [66], maintenance technicians explained their view on maintainability of electric aircraft. In this study, it was mentioned that in general, electric motors require less maintenance and also less unscheduled maintenance, due to less vibrational loads, compared to combustion engines. In addition, it was mentioned that, when a distributed electric propulsion systems is used, the engines can quickly be replaced when an issue arises. It was, however, also noted, that these motors likely are not in stock, and need a timely delivery. The study, however, also mentioned some challenges, mainly related to the battery. According to the participating technicians, a modular battery concept that can be removed by hand would be better for maintenance. In addition, it was also mentioned that removing the wing in order to be able to access a battery stored inside of the wing, introduces additional wear and is therefore not recommended. Furthermore, propeller strikes were mentioned as a particular threat that would require a lot of maintenance time. It was therefore suggested by the respondents, to keep clearance in mind. It was also emphasised by the respondents that a proper design for maintainability can drastically reduce the time needed for maintenance. Therefore, accessibility and modularity were described as being key aspects in order to make the aircraft easily maintained. This extends to components that are not expected to need regular maintenance. From this study, a couple of takeaways are envisioned to be incorporated in the aircraft, these are listed below.

1. Design for proper accessibility. This is also covered in requirement REQ-SYS-32.
2. Make sure that spare parts are either in stock, or can quickly be supplied to Nice airport.
3. Design the battery modularly, and easily removable.

¹Socomec, <https://emea.socomec.com/en/minimise-ups-failures>, Accessed 17-06-2026

²Linak, <https://www.linak.nl/producten/lineaire-actuatoren/la76/>, Accessed 17-06-2026

³Thales, <https://www.thalesgroup.com/en/solutions-catalogue/civil-aviation/high-performance-inertial-reference-system-civil-and-military>, Accessed 21-05-2026

In order to make the maintenance of the aircraft more concrete and to provide a preliminary maintenance schedule, the Pipistrel Velis electro and Alpha electro⁴ are taken as reference aircraft, as these are both small electric general aviation aircraft. From the Pilot's Operating Handbook of the Pipistrel Velis Electro [67] it becomes clear that the aircraft must have a complete inspection every year and every 100 flight hours. In addition, it lists that "the spinner and backing plate must be cleand and inspected for cracks frequently" [67] and that before each flight, the propeller has to be inspected. Extending this to the operations of Vold-e, this would already introduce three maintenance / servicing periods: before every flight, every day and every 100 hours. In addition to these maintenance periods, it becomes clear from Chapter 5, that the battery needs to be replaced every year, making for the fourth maintenance period. The last maintenance period comes from the mean time between failure values as found in Table 12.5. From this table, it becomes clear that the smallest mean time between failure is that of the flight management computer, being 3800 hours. This means that after 3800 flight hours, critical components of the aircraft are statistically likely to fail, and must therefore be replaced. This makes for a total of five scheduled maintenance intervals.

In order to estimate how long each maintenance check will take, the maintenance checks that needs to be performed are listed in Table 12.6. It must be noted that this list is far from exhaustive and mainly serves as an example for the maintenance checks that are needed. These will again be largely based on the Pipistrel Velis Electro Pilot's Operating Handbook [67] and the Pipistrel Alpha Electro Aircraft Maintenance Manual [68]. The table also includes the unscheduled maintenance that might be necessary. Based on [68], unscheduled maintenance might be necessary after a wing strike, hard/overweight landings, exceeded speed limits, severe turbulence, lightning strike, high drag/side loads, ground gusts and operation in harsh environments. In addition to this, bird strikes was also added to this list. Looking at the service checks needed before every flight, this is assumed to take a couple minutes. The daily maintenance check can take a maximum 12 hours, as dictated by REQ-STK-15. The 100 hour maintenance is expected to take around three days⁵. Since the annual maintenance includes many of the tasks that are also in the 100 hour maintenance and since the battery will be designed modularly, this is also expected to take around 3 days. The 3 year maintenance however, is expected to take a similar amount of time to a C check⁶, being between one to two weeks. The time needed for unscheduled maintenance is highly dependent on the type of maintenance to be carried out. However, it will take at least as long as a daily maintenance. The full maintenance schedule is summarised in Figure 12.3 and includes examples for Table 12.6.

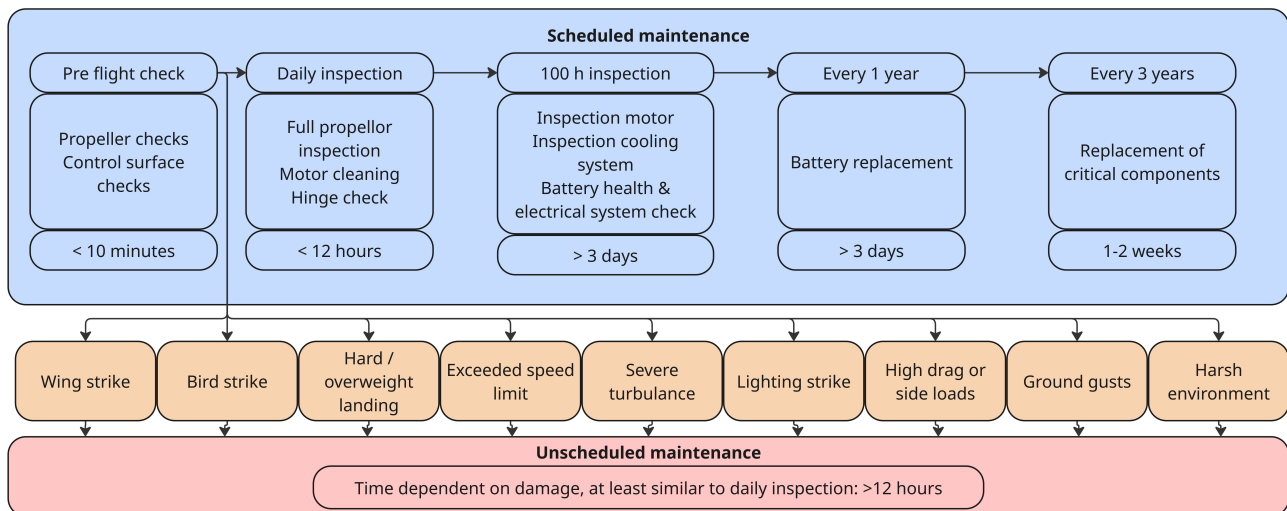


Figure 12.3: Summary maintenance schedule

Looking at these intervals, it becomes clear that some conflicts will arise in relation to the envisioned operations and availability. Firstly, as the aircraft is autonomous and thus has no crew on board, pre flight checks are hard. A possible solution to this could be to ask passengers to take pictures of the aircraft and upload this to an app for analysis. Another solution could be to place cameras at strategic places on the exterior of the aircraft to check the propeller and control surfaces. Secondly, the 100 hour maintenance will take longer than the available 12 hours per day. This means that the aircraft won't be able to operate 200 days non stop and will be out of service for at least three days every 16 days, given 6 flight hours per day. For the route from Nice to Courchevel, this is especially detrimental during the winter season. To solve this, the only solution is to have a second aircraft ready. However, having an aircraft only operate when the other aircraft is in maintenance would not be optimal. Therefore, this would only work for a fleet of five aircraft with one spare. Thirdly, the amount of unscheduled maintenance is a big unknown. This is dependent on the severity of the damage and of the supply change, the latter is interpreted from [66]. In order to

⁴Pipistrel, <https://www.pipistrel-aircraft.com/products/velis-electro/>, Accessed 16-06-2026

⁵AJ Aviation Services, <https://ajaviationservices.com/annual-100-hour-inspection/>, Accessed 16-06-2026

⁶National Aviation Academy, <https://www.naa.edu/types-of-aviation-maintenance-checks/>, Accessed 16-06-2026

cover for this and 100 h maintenance, it might be best to have one spare aircraft per four operative aircraft.

Table 12.6: *Categorised (un)scheduled maintenance tasks*

Category	Maintenance task	Rationale
Pre flight maintenance	Propeller check for dents, scratches and corrosion	Taken from [67]
	Check control surfaces and skin for damages	This is deemed necessary to check if any damage has gone unnoticed
Daily maintenance	Spinner and backing plate inspection and cleaning	Adapted from [67]
	Motor cleaning	Taken from the EMRAX Motor Installation and Maintenance Manual [69]
	Check hinges for damages	As explained in Subsection 12.2.1, the hinges can make for a single point failure and must therefore thoroughly be checked.
100 Hour maintenance	Visual inspection of electric motor	Taken from [68]
	Visual inspection of cooling system	Adapted from [68]
	Inspections of control surfaces and control systems	Adapted from [68]
	Visual inspection pitot tube	Taken from [68]
	Check battery health	Interpreted from [66]
	Check communication system from proper functioning	This is deemed necessary, since the communication system is a critical safety feature as explained in Subsection 12.2.1.
	Check electric system	This is deemed necessary to ensure proper functioning of the electric system and to discover any potential short circuits
Yearly maintenance	Everything included from the 100 hour maintenance	For efficiency reasons to not have two maintenance checks close to each other
	Replacement of the battery	As explained in Chapter 5, the battery needs to be replaced every year
	Check critical components as listed in Table 12.5	It is deemed important to monitor the health of components that can cause a catastrophic failure
3 year maintenance	Replacement of the flight management computer	This is the point where the flight management computer reaches its statistical mean time between failures.
	Replacement of other critical components	The other critical components as listed in Table 12.5 have not yet reached their mean time between failure. Depending on their condition, they can stay operational or be replaced.
Unscheduled	Propeller strike: check blades for signs of damage	From [68]
	Hard landing: check skids	Adapted from [68]
	Exceed speed limit: Inspect wings for damage	Adapted from [68]
	Damage after a bird strike	Bird strikes could happen during flight and might cause damage that needs to be repaired

Continued on next page

Table 12.6 – continued from previous page

Category	Maintenance task	Rationale
	Implement airworthiness directives and safety	EASA might enforce changes to be made to the vehicle by means of airworthiness directives and safety directives ⁷ .

12.2.3. Safety

Now that failure points of the aircraft are clear, a list of safety critical functions can be established as well. As mandated by EASA's VCA regulations VTOL.2005 and MOC VTOL.2000 [59], an enhanced category aircraft must be able to continue flight and perform a safe landing after a single point failure or a combination of non catastrophic failures. Based on this regulation, a list of critical functions is established, as well as the measures taken to perform these functions in case of malfunctioning, as shown in Table 12.7. The critical functions are based on the functional flow diagram to be found in the baseline report [28].

Table 12.7: List of critical functions and measures taken to make sure these functions can be performed in case of failure

Critical function	Explanation	Measures taken
F1.0.9 Conduct safety checks	This step is critical to check for any critical failures before the flight commences.	Detailed system tests need to be designed to reduce the risk of an unnoticed failure.
F1.4 Cruise: continued flight must be possible	When parts of the propulsion system fails, the aircraft shall still be able to fly and maintain altitude.	As specified by REQ-SYS-21, the aircraft shall be able to continue flight with at least 75% of the propulsion system working.
	In case the autonomous system fails, the aircraft must still continue flight.	The vehicle is designed such that a remote pilot can take over control in case of emergency.
F1.2 / F1.6 Transition: At any point during transition, the aircraft should be able to fly.	In case of hinge failure during the transition, the aircraft shall be able to land safely.	As noted in Table 12.4, one actuator failing still allows for the wings to be unfolded.
F1.7 Landing: The aircraft must be able to land.	In case of a jammed hinge, the aircraft must also be able to land.	The aircraft will be designed such that the passengers are not harmed when a conventional landing is necessary.
	In case the autonomous system fails and the communication with the remote pilot as well, the vehicle still needs to perform a landing.	Three autopilots are implemented for triple redundancy.

12.3. Discussion

In this section, the reliability, availability, maintainability and safety of the aircraft have been discussed. In terms of reliability, a list of single point failures and a reliability block diagram was made to assess the reliability. This resulted in a reliability value of $3.2 \cdot 10^{-10}$. As already mentioned, this value is very low, and mainly indicates that a lot of failure modes have not been found or assessed. More detailed failure analysis will need to be performed to ensure compliance with the certification requirements. In terms of maintainability, a schedule for scheduled and (un)scheduled maintenance has been set up. It must be noted, however, that this schedule has been partly based on the Pipistrel Velis electro and Alpha electro ⁸. It is not unlikely that a commercially used eVTOL would require more stringent maintenance. However, already with this maintenance schedule, a bigger fleet is necessary to ensure continuous operations. In future steps, it is therefore recommended to do a more in-depth failure analysis and to establish the minimum fleet size necessary to ensure continuous operation.

⁷ EASA, <https://www.easa.europa.eu/en/domains/aircraft-products/airworthiness-directives-ad>, Accessed 20-05-2026

⁸ Pipistrel, <https://www.pipistrel-aircraft.com/products/velis-electro/>, Accessed 16-06-2026

13 Sustainability and Life-Cycle Impact

Sustainability is a key aspect of the Prandtl-plane eVTOL design, as the aircraft is intended to provide a lower-emission alternative for inter-city transport while still satisfying demanding performance, operational and footprint requirements. Although the vehicle is designed to have no direct in-flight emissions, its overall environmental impact also depends on factors such as electricity production, battery manufacturing, material use, component lifetime, maintenance, recyclability and infrastructure requirements. For this reason, sustainability is considered from both a vehicle and design-process perspective.

Section 13.1 explains the life cycle of the eVTOL and links each stage to the associated environmental impact category. Section 13.2 then presents the design choices incorporated to improve the sustainability of the vehicle across the three main impact areas: operational emissions, recurring and non-recurring life-cycle impact, and infrastructure-related land-use impacts. Then, Section 13.3 describes the sustainable design process used to keep the technical development feasible, efficient, and aligned with the environmental objectives of the project. Finally, Section 13.4 concludes the principal sustainability benefits and discusses the need for a quantitative life-cycle assessment in future design phases.

13.1. Lifecycle of the Vehicle

This section considers the sustainability of the eVTOL from an environmental perspective. The assessment does not only consider the emissions during flight, but also the wider life-cycle impact of the aircraft. This needs to be done because an electric aircraft removes in-flight emissions, but it is not necessarily emissions-free over its full life-cycle. The environmental impact therefore also depends on how efficiently the aircraft uses energy, how long the components remain in service, and how recyclable the structure is. The circular life cycle assessment of the vehicle is presented in the Figure 13.1.

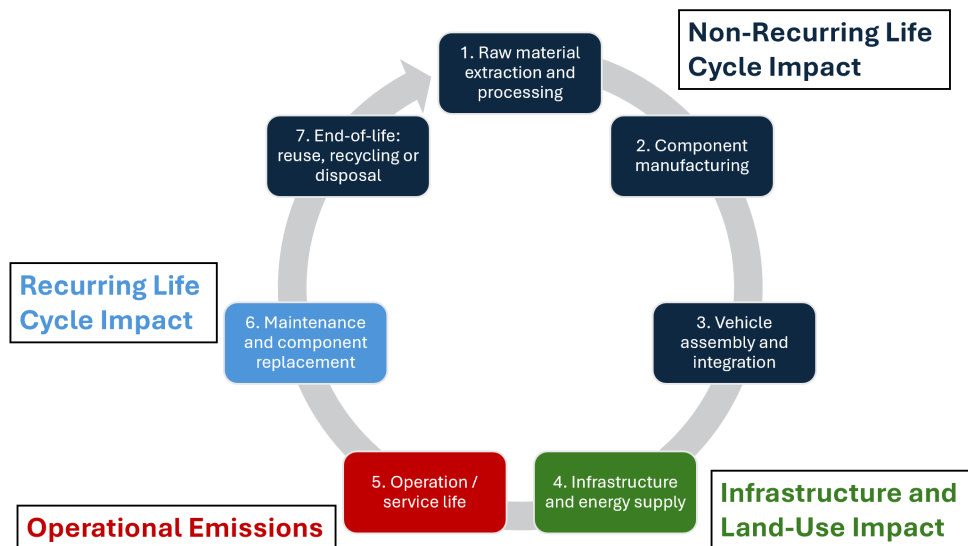


Figure 13.1: Vehicle's life cycle assessment.

The sustainability analysis is divided into four main environmental impact categories, with each life-cycle stage assigned to one of them. First, operational eVTOL emissions associated with in-flight emissions, corresponding to segment 5 in Figure 13.1. Second, life-cycle pollution divided further into recurring impact related to maintenance and component replacement, corresponding to segments 6 in Figure 13.1 and non-recurring impact that arise from raw-material extraction, production, vehicle assembly and end-of-life treatment, corresponding to segments 1, 2, 3 and 7. Finally, infrastructure-related impacts are associated with the facilities required for eVTOL operation, such as charging stations, vertiports, and maintenance facilities, corresponding to segment 4 in Figure 13.1.

13.2. Sustainable Design Choices

Since the environmental performance of an electric eVTOL depends on more than the absence of in-flight emissions, environmentally focused design choices were implemented to improve the sustainability of the vehicle across all three environmental impact areas: operational emissions, recurring and non-recurring life-cycle impact, and infrastructure-related land-use effects. Table 13.1 summarises the sustainable design choices incorporated in the final aircraft concept and links them to the three main sustainability categories. More in detail description of each area of sustainability is provided in the following subsections. Subsection 13.2.1 discuss recurring impact including both recurring life cycle pollution and operational Vold-e pollutions. Then, Subsection 13.2.2 discussed non-recurring

impact related to beginning- and end-of life of the vehicle. Finally, the additional environmental impact caused by the infrastructure required for eVTOL operation is assessed in Subsection 13.2.3.

Table 13.1: Sustainable design choices and remaining environmental impacts

Design choice	Rationale	Remaining pollution
Operational emissions		
Battery-electric propulsion	Removes in-flight emissions; supports zero in-flight CO ₂ .	Emissions from electricity generation.
Prandtl-plane config, max gap $h = 2.1$ m	Increases aerodynamic efficiency, lowering energy and battery capacity per km.	Manufacturing impact of wing/structural materials.
Low-noise electric propulsion	Removes combustion noise; improves urban acceptance.	Propeller noise during hover, take-off, landing.
Recurring life-cycle emissions		
Battery state-of-health monitoring	Avoids premature replacement; enables condition-based maintenance.	Degradation and eventual replacement unavoidable.
Composites in airframe	Reduce weight and energy per flight; break-even at 300,000 km [70].	Higher manufacturing emissions than aluminium until break-even reached.
Remaining-useful-life prediction	Redirects retired batteries to second-life use.	Production still needs energy and critical raw materials.
Modular battery replacement	Safer, less destructive maintenance.	Recurring material/manufacturing impact of packs.
Modular load-bearing structure	Wing/tail panels and gear attachments inspectable and replaceable without destroying surroundings.	Material production and recycling impacts remain.
Accessible vulnerable components	Batteries, motors, inverters, hinges, locks, avionics easier to service.	Spare parts and discarded components create waste.
Avoid unnecessary permanent joints	Improves separability, repairability, recyclability.	Bonded/mixed-material interfaces still hard to recycle.
Replaceable hinge/lock modules (folding wing)	Fatigue-sensitive parts inspected/replaced separately from main structure.	Wear and fatigue need scheduled inspection over folding cycles.
Material/design with more fatigue cycles (trading weight)	Reduces replacement frequency of critical, hard-to-access transformation parts.	Wear, fatigue, inspection remain; added weight raises flight energy.
Skid gear with reusable elastomeric absorbers	Absorbs landing energy without full gear replacement after normal landings.	Absorbers, rails, attachments need inspection; replacement after hard landings.
Non-recurring life-cycle emissions		
High-discharge-rate cells	Easier to repurpose in other EVs such as cars.	Higher manufacturing cost and emissions [71].
Battery pack sized to mission/reserve	Avoids excess capacity, reducing cell material demand.	Required capacity still has production and end-of-life impact.
Physics-based structural sizing	Sizes wing, tail, gear to load cases rather than conservative mass assumptions.	Scrap, tooling, processing still contribute.
Material selection for manufacturability/recyclability	Choices balance mass with production feasibility and recovery; see Table 13.2.	Reprocessing needs energy and may reduce quality.
Design for material separability	Eases separation of batteries, composites, metals, electronics.	Mixed/bonded interfaces still hard to separate.
Infrastructure and land-use impact		
Low-carbon charging electricity	Prevents shifting emissions to a fossil-intensive grid.	Grid emissions remain without full renewables.
One-hour charging battery and infrastructure	Supports turnaround time; enables multiple daily mission cycles.	High rates raise thermal, safety, degradation demands.
	Station sized for daily operation limits number of stations.	High utilisation raises local grid peak loads.
	Avoiding oversizing limits material, space, peak-load impact.	Dedicated grid connection and equipment still required.
	Standardised connection simplifies operation and compatibility.	Connector hardware and safety systems add material.
Charging system sized for one-hour turnaround	Avoids delays and oversizing; limits material, space, peak load.	Power electronics, safety systems, grid capacity needed.
Folding mechanism; footprint $10 \times 6 \times 3$ m ³	Reduces required vertiport area.	Site adaptation may still be required.
	Supports helipads, airports, parking, resort sites.	Charging and maintenance facilities still affect sites.
	Limits parked size vs larger fixed-wing concepts.	Clearance for folding, boarding, access still needed.

Design choice	Rationale	Remaining pollution
Integrated maintenance/charging facilities	Concentrates operations, avoids duplicated infrastructure.	Hangars, tools, spares still have material/energy impact.

13.2.1. Recurring eVTOL Life Cycle Impact

Recurring life-cycle impacts are associated with the service life of the Prandtl-plane eVTOL and include maintenance activities, component inspection, part replacement, and the durability of components exposed to repeated load cycles. For this reason, the recurring impact is strongly affected by how easily the aircraft can be maintained and how often major components must be replaced. An overview of the inspection impact and replacement for the main components can be found in Section 12.2.

The battery system is one of the most important recurring life-cycle contributors. Although battery-electric propulsion removes local in-flight emissions, battery degradation during repeated operation can lead to regular battery-pack replacement. Therefore, the battery system shall include state-of-health monitoring and remaining-useful-life prediction (REQ-SYS-NEW-05), to avoid premature battery replacement, enable condition-based maintenance, and allow battery packs that are no longer suitable for flight-critical operation to be redirected to less demanding second-life applications.

The structure also has to be designed with maintainability in mind. The main load-bearing structure should be modular where possible, so that damaged or worn sections can be inspected, removed, repaired or replaced without destroying the surrounding structure (REQ-SYS-29, REQ-SYS-30). Furthermore, vulnerable components such as batteries, motors, inverters, hinges, locks and avionics should remain accessible for inspection and replacement (REQ-SYS-32). Therefore the interfaces should remain accessible, unnecessary permanent joints should be avoided, and maintenance actions should be possible without removing large unrelated structural sections.

Durability directly reduces the recurring life-cycle impact by lowering material use, maintenance demand and operational waste from component replacement. Therefore, aircraft lifetime and durability requirements, such as REQ-MS-27 and REQ-SYS-41, support not only safety, but also sustainability and cost reduction. This is particularly important for the folding mechanism, where fatigue-sensitive components such as hinges and locks must be selected and sized for repeated load cycles.

13.2.2. Non-Recurring eVTOL Life Cycle Impact

Non-recurring life-cycle impacts are associated with the beginning and end of the aircraft life cycle. These include raw-material extraction and processing, component manufacturing, vehicle assembly and integration, and end-of-life treatment. Although these impacts do not occur during every flight, they can form a significant part of the total environmental burden of an electric aircraft, especially because of the battery system and the use of high-performance structural materials.

At the beginning of life, the battery system is expected to be one of the largest contributors. The production of high-performance lithium-ion batteries requires energy-intensive manufacturing and critical raw materials. Therefore, the battery pack should be sized to meet the mission, reserve and power requirements without unnecessary excess capacity to limit the upstream production impact. Structural material production is another important non-recurring impact. Reducing unnecessary structural mass lowers the amount of raw material required and reduces the impact of component manufacturing.

At end of life, the aircraft should be designed to allow major components and materials to remain useful after the first operational life of the vehicle (REQ-STK-10). Components should be reusable, recyclable or separable where possible, rather than permanently integrated in a way that prevents recovery. Table 13.2 summarises the main materials used in critical components and identifies their main beginning-of-life and end-of-life impacts.

Table 13.2: *Beginning-of-life and end-of-life impacts of selected eVTOL materials*

Main material or technology	Components	Beginning-of-life impact	End-of-life consideration
Lithium-ion battery cells	Battery pack	Cell manufacturing is energy- and material-intensive, particularly due to the extraction and processing of lithium, nickel, cobalt and graphite.	Batteries can be repurposed for lower-demand second-life applications before recycling. Material recovery is possible, but requires specialised recycling processes.

Main material or technology	Components	Beginning-of-life impact	End-of-life consideration
Carbon-fibre-reinforced polymer (CFRP)	Wing skin; wing spar caps and spar web; tail structure; propeller blades	Carbon-fibre and resin production are energy-intensive. However, the low mass of CFRP can reduce energy consumption during aircraft operation.	Composite recycling is more difficult than metal recycling and often results in recovered fibres with reduced material quality.
Aluminium	Wing stringers; landing gear	Primary aluminium production is energy-intensive. Its relatively high density compared with composites can also increase aircraft mass and operational energy use.	Aluminium can be recycled efficiently with substantially lower energy use than primary production and can be readily reused in other applications.
Electromechanical and electronic materials	Electric motors and inverters	Motors and inverters require copper, electrical steel, permanent magnets and electronic components. Their extraction and manufacturing can be material- and energy-intensive, particularly for rare-earth permanent magnets and semiconductor devices.	Metals such as copper and steel can be recovered, while magnets and electronic components require specialised disassembly and recycling processes.
Titanium alloy with stainless-steel pins	Hinges and locking mechanisms	Titanium production is energy-intensive, but its high strength, corrosion resistance and fatigue performance can reduce maintenance and replacement requirements. Its low mass can also reduce operational energy use.	Both titanium and stainless steel can be recycled and reused, although separation of the different metals is required.

13.2.3. Infrastructure and Land-Use Impact

Since the vehicle is designed for inter-city operations aspects such as the charging system, vertiport layout and maintenance facilities will significantly contribute to the total environmental impact. The design should therefore minimise the amount of new dedicated infrastructure required, while ensuring that the infrastructure which is needed supports low-emission operation.

A first important aspect is the electricity source used for charging. If the aircraft is charged using electricity from fossil-intensive sources, part of the emissions are shifted from the aircraft to the power grid. Therefore, the operational concept should prioritise charging from low-carbon or renewable electricity sources where possible. The charging infrastructure also affects the sustainability of the service. According to REQ-SYS-42, the aircraft refuelling or recharging process shall take a maximum of one hour. Thus, the charging station must be sized not only for overnight charging, but also for repeated daily operation. The infrastructure must therefore include a suitable grid connection, power electronics, safety systems, thermal-management support and enough charging availability to avoid excessive turnaround delays. However, larger electrical installations require more materials, occupy more space and may increase the peak load on the local grid. Therefore, from a sustainability perspective, the system should avoid oversizing and aim for integration with the local electricity grid with minimal additional land use. A balanced charging strategy should therefore account for the one-hour turnaround requirement, realistic battery thermal limits, and appropriate charging-station sizing.

The second main aspect is to reduce the need for large dedicated vertiports and major land-use by reducing ground footprint of the vehicle. A central objective of the Prandtl-plane eVTOL is to reduce the land area needed for take-off, landing and ground handling compared to conventional fixed-wing or large-span eVTOL concepts. This is directly connected to REQ-MS-02 and REQ-SYS-02, which require the vehicle to remain within the defined ground footprint and the maximum on-ground envelope of $10 \times 6 \times 3 \text{ m}^3$. The land-use impact is not limited to the parked aircraft size, but also includes take-off and landing clearance, passenger boarding, charging, inspection, maintenance access, emergency access and possible folding or unfolding motion of the wing system. The aim of the small ground footprint is to make use of compact helipads, existing airport, parking spaces or resort infrastructure.

13.3. Sustainable Design and Development Process

A sustainable design process is defined to ensure that the technical design process remains feasible, efficient and aligned with the long-term environmental impact of the vehicle. In this context, sustainability is interpreted as the ability of the design to satisfy the mission requirements while limiting unnecessary mass, energy use, cost growth, maintenance demand and environmental impact over the vehicle life cycle. The strategy is therefore directly linked to the technical risks identified in Section 12.1.

The overall sustainability strategy was established in the Baseline Report [28]. This section therefore focuses only on the sustainability measures that were updated during the subsequent design process.

Configuration-Specific Sustainable Design Strategies:

- Treat the folding and unfolding sequence as a system-level design driver. The full motion envelope of the wings, propellers, landing gear, cabin access zones, wiring routes and maintenance access shall be checked in the shared CAD model after each major design change. (TR-SI-UP-03, TR-SI-UP-07, TR-PP-UP-07, TR-BO-UP-02)
- Define routing paths for high-voltage wiring, signal cables and cooling lines through the moving wing sections early in the layout process. Bend radii, strain relief, insulation, fatigue protection and inspection access shall be considered to avoid late integration issues. (TR-PP-12, TR-SI-UP-07, TR-SI-UP-08)
- Define a transition corridor for the Folding Prandtl concept, including limits on speed, altitude, attitude, wind condition, centre-of-gravity position, folding rate and allowable altitude loss. Aerodynamic-centre movement, trim capability and control margins shall be evaluated throughout the manoeuvre. (TR-AA-UP-03, TR-AA-UP-04, TR-VDC-UP-08, TR-VDC-UP-09)
- Treat hinges and locking mechanisms as critical lifecycle components. Their sizing shall include deployment loads, flight loads, stiffness requirements, repeated transition fatigue, wear, looseness and vibration sensitivity. The design should favour inspectable, replaceable and fail-safe hinge and lock modules. (TR-ST-UP-02, TR-ST-UP-03, TR-ST-UP-10, TR-ST-UP-11, TR-BO-UP-05)
- Include a maintainability gate for the folding mechanism. Hinges, locks, actuators, routing interfaces and batteries shall have defined inspection paths, removal clearances and replacement procedures before the layout is frozen. (TR-SI-UP-08, TR-BO-UP-05, TR-ST-UP-11, TR-PP-12)
- Control the ground footprint, including the dynamic folding and unfolding sequence. After each design change, verify that the parked configuration, transformation motion, propeller clearance, boarding, charging, inspection, maintenance and emergency access remain compatible with the footprint requirement. (TR-BO-UP-02, TR-PP-UP-07, TR-SI-UP-03)

Overall, these strategies ensure that the folding Prandtl configuration is not only optimised for aerodynamic and structural performance, but also for integration feasibility, maintainability and long-term operational sustainability. By linking each action to the updated technical risks, the design process remains focused on preventing late-stage mass, cost and complexity growth.

13.4. Discussion

The sustainability assessment shows that Vold-e can reduce the environmental impact of inter-city transport by eliminating direct in-flight emissions and by improving aerodynamic efficiency through the Prandtl-plane configuration. However, the aircraft cannot be considered emissions-free over its complete life cycle. Its overall impact remains strongly dependent on the carbon intensity of charging electricity, the production and replacement of lithium-ion battery packs, and the manufacture and end-of-life treatment of composite and electronic components.

The design therefore incorporates sustainability measures beyond operational efficiency, including battery health monitoring, modular and accessible components, replaceable folding-mechanism modules, structural sizing to avoid unnecessary material use, and a compact folded ground footprint. These measures reduce maintenance-related waste, support repair and component reuse, and limit the need for dedicated vertiport infrastructure. Nevertheless, the folding mechanism introduces additional components that are subject to fatigue, inspection and replacement, while the use of composites and high-performance batteries creates significant upstream manufacturing and recycling challenges.

The present assessment is qualitative and identifies the main environmental drivers and mitigation measures. A future design phase should therefore perform a quantitative life-cycle assessment, including battery production and replacement rates, electricity-mix scenarios, composite recycling pathways, manufacturing scrap, maintenance demand and infrastructure construction. This would allow the environmental benefit of Vold-e to be compared directly with conventional rotorcraft, road transport and alternative electric aircraft concepts.

14 Financial Analysis and Post DSE Activities

This chapter focuses on calculating the total production and operational costs of Vold-e as well as outlining the post DSE activities. Initially in Section 14.1 a short market analysis is completed considering the expected costs and prices within the eVTOL market. In Section 14.2 a budget breakdown is completed to show the expected costs of producing Vold-e. Afterwards the operational costs are determined and in the current and future market Section 14.3. Furthermore, to assess the profit possibilities with Vold-e, two cash flow statements were made for both the production and operational side. From these two statements a return on investment can be calculated as seen in Section 14.4. To analyse the post DSE activities, first, a process to outline how Vold-e is produced is described in Section 14.5. Next two separate organisational charts are created to outline all the necessary steps to fully finalize the design. In addition to this, all these tasks are outlined in a Gantt chart to provide a clear timeline overview all depicted in Section 14.6. Finally, Section 14.7 summarises the financial feasibility of Vold-e, compares the estimated costs with a representative competitor, and identifies the main uncertainties and future work required to further develop the business case.

14.1. Literature Review on Financial Landscape

In this section, the values for the possible investment size, market size and annual orders will be discussed. In addition, the estimated value from the baseline report [28] for the acquisition cost will be stated and compared to the calculated value in subsequent sections. In 2021, Porsche Consulting [1] noted that starting in the vertical mobility sector is challenging. It predicts that no positive return on investment can be expected before 2030 in the most probable case and conservatively this moves to 2035. A study from Roland Berger in collaboration with the DLR [6] has shown that UAM is still a viable market, in spite of investments dropping: investments in advanced air mobility saw their peak in 2021 with total investments worth 7.5 billion, but dropped to 1.3 billion in 2023. The study also mentions recent investments in eVTOL companies, which are between 119 and 450 USD million. This gives a range of investment sizes that can be expected for Volde.

To estimate the costs of purchasing the envisioned eVTOL, prices from competitors are used, given in Table 1.2. Given that the Joby and E20 aircraft are the most similar to the envisioned eVTOL, 1 million USD is deemed as a feasible lower limit for the acquisition price. In order to ensure market competitiveness, an upper limit of 3 million USD is taken, based on the price of the airbus ACH125 helicopter.

In terms of market size, the study from Roland Berger and DLR study noted that a lot of companies might disappear from the market. It is therefore assumed, that a market share of 5% is achievable for the envisioned eVTOL in the European market. Using the market size of €4.2 billion in 2030 as forecasted by EASA [2], this would make for an annual revenue of €210 million. Assuming a similar profit margin to airbus of 7.1%¹, an annual profit of €15 million is expected and deemed a conservative estimate once the eVTOL would be in stable production. Given the fact that EASA [2] estimates the European market to be roughly 31% of the whole market, in combination with the 10 000 [6] orders already placed and a European market share of 5%, 155 orders can safely be assumed as a minimum for the envisioned aircraft. Looking again at the Porsche Consulting study [1], it was noted that once eVTOLs are used a lot, each manufacturer will sell on average 1000 eVTOLs a year. This will therefore be taken as the upper bound for the envisioned aircraft.

14.2. Production Costs

Analytical methods were combined from [72], [73] and [74] to calculate the costs of production. In order to correctly assess the production costs, the overall cost of an eVTOL was split into two distinct sections, development costs and manufacturing costs. In addition to this, the production was analysed over a timeline of 15 years. This was done in order to assess the estimated costs of producing in the future market in comparison to the current market.

Initially, it was estimated that in the first 5 years (up to 2031) Vold-e would need additional years of development and testing as shown in Section 14.6. However, currently within the methods used, there is no direct calculation to estimate the development costs. These development costs are included into the total price of producing an eVTOL. To still account for this development stage properly, it was assumed that these costs include the engineering costs, tooling costs, development costs and flight test operation costs. The rest of the costs are assumed to be for the actual manufacturing of the eVTOL.

Overall, the development costs were calculated for the total estimated number of eVTOLs produced and then divided over the ten years. During phase one development costs are expected to continue as development will be focused on preparing the factories for mass production whilst initially it is focused on the actual function of the eVTOL. It was also assumed that initially the eVTOL is sold with a set of batteries, but these have to be changed yearly which are then added into the operational costs. In Table 14.1 the overall cost per eVTOL for phase 1 and 2 are predicted and a cost breakdown is shown.

From this preliminary cost breakdown, a detailed cash flow statement can be made for the upcoming few years highlighting the difference between the costs and income during phase 1 and phase 2

¹ Yahoo! Finance: <https://finance.yahoo.com/quote/EADSF/key-statistics/>. Accessed on 29th of April 2026

14.3. Operational Costs

Similar to the manufacturing costs, the operational costs have been based on research by Kirste and Stumpf [75]. To determine the operational costs for Vold-e, only the air traffic control, vertiports operational costs, maintenance, energy and flight crew costs have been considered. This was done as it is assumed that the operator will not build the vertiports and the charging infrastructure fee is incorporated in the price per [kWh]. In addition, the acquisition cost and the battery costs will be accounted for in the cash flow and as such, will not be put in the operational costs. The resulting costs break down can be found in Table 14.1. From this breakdown, it becomes clear that an operating costs of around 242 [€] per flight can be expected for operations in the Courchevel area. For transparency, all values used in the equations, including the rationale, are shown in Table 14.2.

Table 14.1: Production and operational cost breakdown of Vold-e

Production costs [Million € FY2026, per eVTOL]

	Phase 1	Phase 2
Assumed no. of eVTOLs made in 5 years	420	2300
Development	460	820
Development cost per eVTOL	1.1	0.36
Manufacturing	0.48	0.30
Propulsion	0.19	0.16
Avionic	0.10	0.08
Batteries	0.17	0.17
Total production cost	0.94	0.71
Total cost per eVTOL	2.0	1.1

Operational costs [€ FY2026, per flight]

Cost category	Phase 1 [€]	Phase 2 [€]
ATC costs	42	17
Operational vertiport costs	63	25
Maintenance costs	54	42
Energy costs	60	33
Flight crew costs	23	9
Total	242	122

Table 14.2: Values used in the calculation of the operating costs

Symbol	Definition (taken or adapted [75])	Value Phase 1	Value Phase 2	Rationale
n_{vp}	Number of vertiports	1 [-]	1 [-]	Since no real vertiports will be used in phase 1, this is assumed a conservative estimate to include landing fees
n_{shifts}	Number of labor shifts for ATC	1 [-]	1 [-]	Default taken from [75]
c_{salary}	Salary of ATC personnel	50k [€]	25k [€]	Default taken from [75] and expected to decrease because of U-space in phase 2
Σ	Number of annual operating days	1200 [flights]	3000 [flights]	Sigma is adjusted to the total flights per year to calculate the operational costs per flight.
c_{netop}	Annual operating cost per vehicle	75k [€]	75k [€]	Default taken from [75]
$n_{fleet} (ATC)$	Size of the fleet	1 [-]	1 [-]	Based on the one route from Nice to Courchevel
c_{maint}	Maintenance cost factor	50 [€/FH]	50 [€/FH]	Default taken from [75]
c_e	Energy price	0.40 [€/kWh]	0.25 [€/kWh]	Taken from IONITY ¹ and rounded up
E_{total}	Total energy consumption	150 [kWh]	150 [kWh]	Based on mission analysis
f_{pilot}	Binary variable to include pilot	1 [-]	1 [-]	A remote pilot is included
c_{pilot}	Annual salary of the pilot	140k [€]	140k [€]	Taken from Nationale beroepen gids ²
n_{crew}	Crew size	1 [-]	1 [-]	Only the remote pilot is needed
$n_{fleet} (Pilot)$	Fleet size	0.2 [-]	2 [-]	We assume one remote pilot per five flights

¹IONITY, <https://www.ionity.eu/subscriptions>, Accessed 16-06-2026

²Nationale beroepen gids, <https://www.nationaleberoepengids.nl/salaris/piloot>, Accessed 16-06-2026

14.4. Return on Investment

Using the estimated cost values from Section 14.2, the return on investment can be calculated from two different perspectives. Once for the manufacturers side, and once for the operational side. This aims to show whether producing the eVTOL is profitable as well as estimating the ticket price from the operational site. Again, similar to how costs were calculated previously, there is a clear distinction between the phase 1 market and the phase 2 market. This is largely seen due to the number of eVTOLs produced.

Firstly, the return on investment from the manufacturing side will be calculated. In order to do this, a detailed cash flow will be made. This cash flow is split in three phases: the development phase, the first phase of operations and the second phase of operations. In the development phase, the aircraft will be designed and now revenue will be made. This phase is expected to last for six years. This is based on the fact that the Boeing 787 Dreamliner had an expected development time of five years, but in reality took nine years². Given that the 787 is a lot larger than Vold-e, a one year delay is accounted for, making for a six years development phase. In the first phase of operations, the eVTOL will be used on the route from Nice to Courchevel or similar routes in the premium market. In the second phase, the eVTOL will be used at large as urban and inter-city air taxis. The latter two phases are largely based on the phases determined by [1]. However, in order to prepare the production for mass production, additional development costs are made in phase 1.

As explained in Section 14.2, the amount of eVTOLs in phase two is expected to rise. As such, the production cost per eVTOL will also drop. Putting these costs in a cash flow and accounting for the three phases, the needed investment can be extracted from the negative cash. This makes for a total investment need of € 1 000 000 000. As explained earlier, the expected investment size is between 119 and 450 USD. Based on this, it is assumed that three investment rounds will be needed, each lasting two years, until the first revenue is made in the first phase of operations. This cash flow can be found in Table 14.3. Overall this means that Vold-e can be sold in phase 1 for a total of €3 million whilst in phase 2 this price decreases to €1.4 million. It must be noted that in Table 14.3 and Table 14.4 all costs are depicted in the respective financial year value, as each value is corrected for an estimated inflation of 3%.

Table 14.3: Vold-e Cash Flow Statement

	Development Phase						Phase 1					Phase 2				
Year	2026	2027	2028	2029	2030	2031	2032	2033	2034	2035	2036	2037	2038	2039	2040	2041
Year count	0	1	2	3	4	5	6	7	8	9	10	11	12	13	14	15
eVTOLs sold	0	0	0	0	0	0	20	50	80	120	150	300	500	500	500	500
Inflation factor from 2026	1	1.0	1.1	1.1	1.1	1.2	1.2	1.2	1.3	1.3	1.3	1.4	1.4	1.5	1.5	1.6
Inflation factor from 2032							1	1.03	1.06	1.09	1.1	1.2	1.2	1.2	1.3	1.3
Development Costs [M€]	-60	-130	-130	-130	-140	-140	-150	-150	-160	-150	-160					
Production Costs [M€]							-22	-60	-100	-150	-190	-290	-500	-520	-540	-550
Total costs [M€]	-60	-130	-130	-130	-140	-140	-170	-210	-250	-300	-360	-290	-500	-520	-540	-550
Price per eVTOL [M€]							3.0	3.1	3.2	3.3	3.4	1.4	1.4	1.5	1.5	1.6
Investment [M€]	300		250		450											
Income [M€]	0	0	0	0	0	0	60	150	250	390	510	420	720	750	760	790
Profit [M€]	-60	-130	-130	-130	-140	-140	-110	-50	3.7	86	150	130	220	230	230	240
Net Cash [M€]	240	110	230	99	410	270	160	110	110	190	350	470	690	910	1100	1400
ROI [%]	-0.20	-0.62	-0.58	-0.82	-0.59	-0.73	-0.84	-0.89	-0.89	-0.81	-0.65	-0.53	-0.31	-0.09	0.14	0.38

From this table, it is clear that the return of investment will only be positive in 2040, and although this is quite a long time period, it is expected that after this stage the return of investment will increase drastically. This is mostly due the fact that mass production will start picking up and will exponentially increase the profit.

Looking at the operational side of Vold-e during phase 1, Table 14.4 shows the expected cash flow over Vold-e's lifetime (ten years). At the beginning the eVTOL must be purchased for a price of € 3 million and hence the total cash in the beginning is negative indicating that an initial investment is required. From this cash flow statement it was possible to calculate the return on investment over the ten years, as well as estimate a ticket price per person per flight to account for a profit. From this cash statement it was estimated that for phase 1 the ticket price for a flight between Nice and Courchevel is € 330 and that the total return on investment calculated through:

$$ROI = \frac{Netprofit - Totalinvestment}{Totalinvestment} \quad (14.1)$$

²Simple Flying, <https://simpleflying.com/modern-airliners-development-cycle-cost-guide/>, Accessed 16-06-2026

Using this equation it was calculated that the return on investment is 9%. A similar process is carried out for the second phase of operation and the cash statement is shown in Table 14.5. Here it is clear that ticket price for a positive return on investment is equivalent to 140 [€] and a return of investment equivalent to 15% after 10 years.

Table 14.4: Operational Costs Cash Flow Statement Phase 1

Year count	0	1	2	3	4	5	6	7	8	9
Passenger fraction from max [%]	0.8	1	1	1	1	1	1	1	1	1
Number of flights [-]	960	1200	1200	1200	1200	1200	1200	1200	1200	1200
Number of passengers	3500	4300	4300	4300	4300	4300	4300	4300	4300	4300
Ticket price [€]	330	340	350	360	370	380	390	400	420	430
Costs per flight [€]	240	250	260	260	270	280	290	300	310	320
Capital expenditure (CAPEX) [Million €]	-3.0	-0.17	-0.18	-0.18	-0.19	-0.19	-0.20	-0.21	-0.21	-0.22
Operating expenditure (OPEX) [Million €]	-0.23	-0.30	-0.31	-0.32	-0.33	-0.34	-0.35	-0.36	-0.37	-0.38
Total expenditure [Million €]	-3.2	-0.47	-0.49	-0.50	-0.52	-0.53	-0.55	-0.56	-0.58	-0.60
Income [Million €]	1.1	1.5	1.5	1.6	1.6	1.7	1.7	1.8	1.8	1.9
Net Cash flow [Million €]	-2.1	1.0	1.0	1.1	1.1	1.1	1.2	1.2	1.2	1.3
Total cash [Million €]	-2.1	-1.1	-0.07	0.99	2.1	3.2	4.4	5.5	6.8	8.0

Table 14.5: Operational Costs Cash Flow Statement Phase 2

Year count	0	1	2	3	4	5	6	7	8	9
Passenger fraction from max [%]	0.8	1	1	1	1	1	1	1	1	1
Number of flights in thousands [-]	2.4	3.0	3.0	3.0	3.0	3.0	3.0	3.0	3.0	3.0
Number of passengers in thousands [-]	8.6	10.8	10.8	10.8	10.8	10.8	10.8	10.8	10.8	10.8
Ticket price [€]	140	144	149	153	158	162	167	172	177	183
Costs per flight [€]	122	125	129	133	137	141	145	150	154	159
Capital expenditure (CAPEX) [Million €]	-1.4	-0.17	-0.18	-0.18	-0.19	-0.19	-0.20	-0.21	-0.21	-0.22
Operating expenditure (OPEX) [Million €]	-0.29	-0.38	-0.39	-0.40	-0.41	-0.42	-0.44	-0.45	-0.46	-0.48
Total expenditure [Million €]	-1.7	-0.55	-0.57	-0.58	-0.60	-0.62	-0.64	-0.66	-0.68	-0.70
Income [Million €]	1.2	1.6	1.6	1.7	1.7	1.8	1.8	1.9	1.9	2.0
Net Cash flow [Million €]	-0.48	1.0	1.0	1.1	1.1	1.1	1.2	1.2	1.2	1.3
Total cash [Million €]	-0.48	0.53	1.6	2.6	3.7	4.9	6.0	7.2	8.5	9.8

14.5. Production Plan

In order to better visualize the production process of the aircraft, a production plan was made. Due to the aircraft design still being in a preliminary stage, many details about production processes and integration steps remain ambiguous. As such, the plan does not go into detail with individual processes, and instead provides a general overview of the steps that will be taken. It includes whether or not parts will be produced in house or outsourced/bought, which parts will form into an assembly, and critical integration steps that will be taken.

14.6. Future Work & Project Gantt Chart

Throughout the project, substantial work went into finalizing the technical aspects of Vold-e. Some areas, however, remain incomplete. The transition phase in particular was not analysed, and it must be studied in detail before Vold-e can be produced. Several other technical aspects were likewise not fully developed and require further attention. To outline this remaining work, a project design and development diagram was created. It outlines all outstanding tasks for the technical design, extends to Vold-e's operational structure within the market, and continues through to the activities required in the end-of-life phase. Finally, to translate these tasks into a clear plan and timeline, a Gantt chart was produced assigning a time slot to each necessary task.

14.7. Discussion

The financial analysis indicates that Vold-e can become profitable. From the manufacturing side, a total investment of roughly €1 billion is required across three rounds, and the return on investment does not turn positive until 2040, around fourteen years after development begins; this is consistent with the long payback periods predicted by Porsche Consulting [1], and profit is expected to grow rapidly once mass production scales up in phase 2. From the operational side, both phases are profitable over the ten-year vehicle lifetime, yielding a return on investment of 9% in phase 1 (at a 330 [€] ticket price on the Nice–Courchevel route) and 15% in phase 2 (at a 140 [€] ticket price). A limitation of these estimations, is that the analysis depends heavily on external cost models [72, 75] not developed specifically for Vold-e and not considering the actual price of certain components.

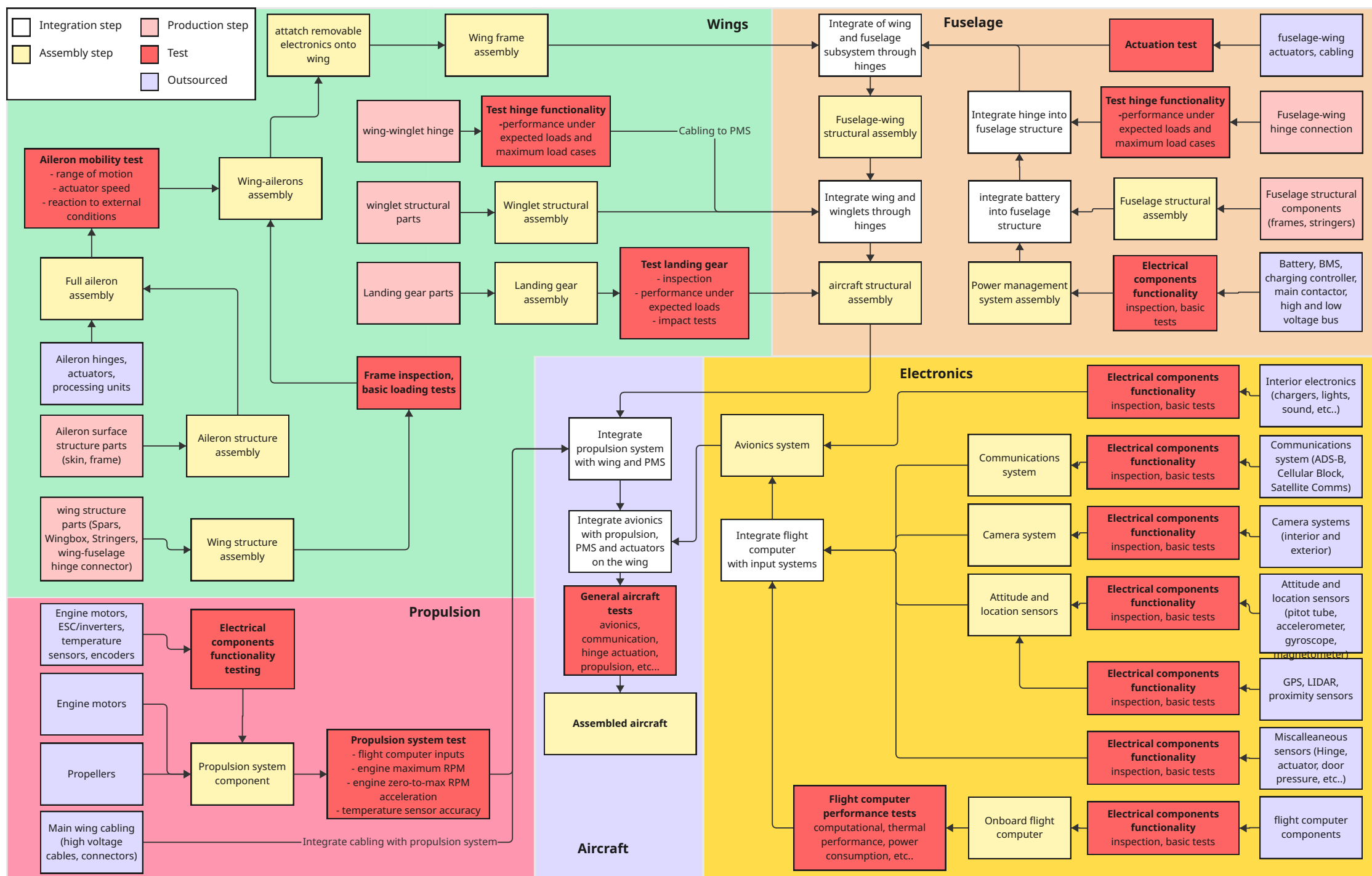
It is therefore recommended that in the future further analysis must be completed on the actual costs of certain components considering both the manufacturing side and operational side.

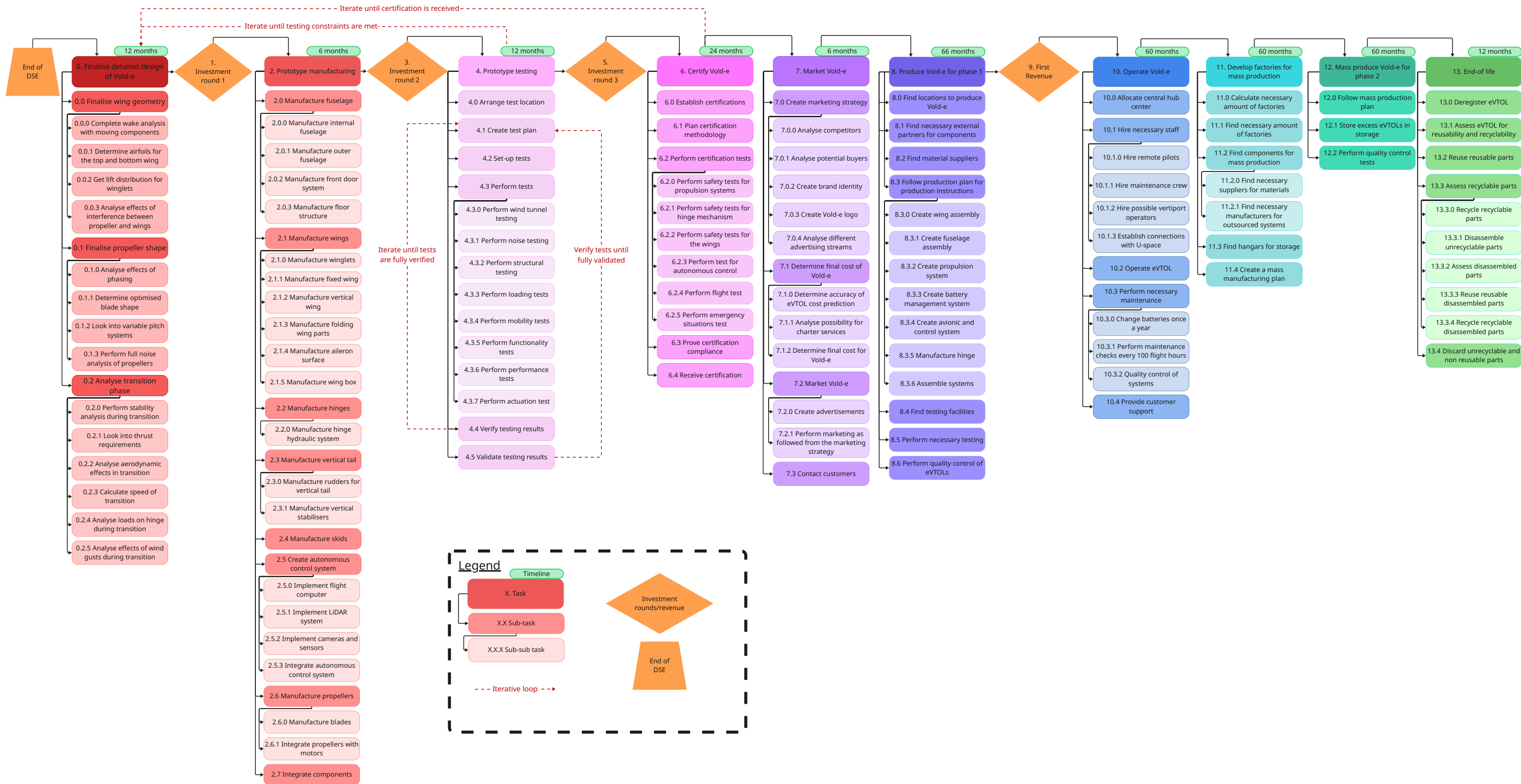
To validate the financial results, Vold-e's pricing figures are compared against Joby Table 14.6. The comparison shows that Vold-e's production cost, selling price, and operational cost per flight are all of the same order of magnitude as Joby's, which provides confidence that the estimates produced here are realistic. Vold-e's figures are somewhat higher across than Joby, which is consistent as Joby is already in a further stage of development than Vold-e. Notably, Vold-e's ticket price per kilometre of 1.7 [€] is well below Joby's 3.0 [€], suggesting that Vold-e could remain price-competitive even against an established operator. In addition to the financial costs, all tasks that still needed to be completed were shown in the post DSE plans.

Table 14.6: *Comparison of Vold-e against the Joby eVTOL*

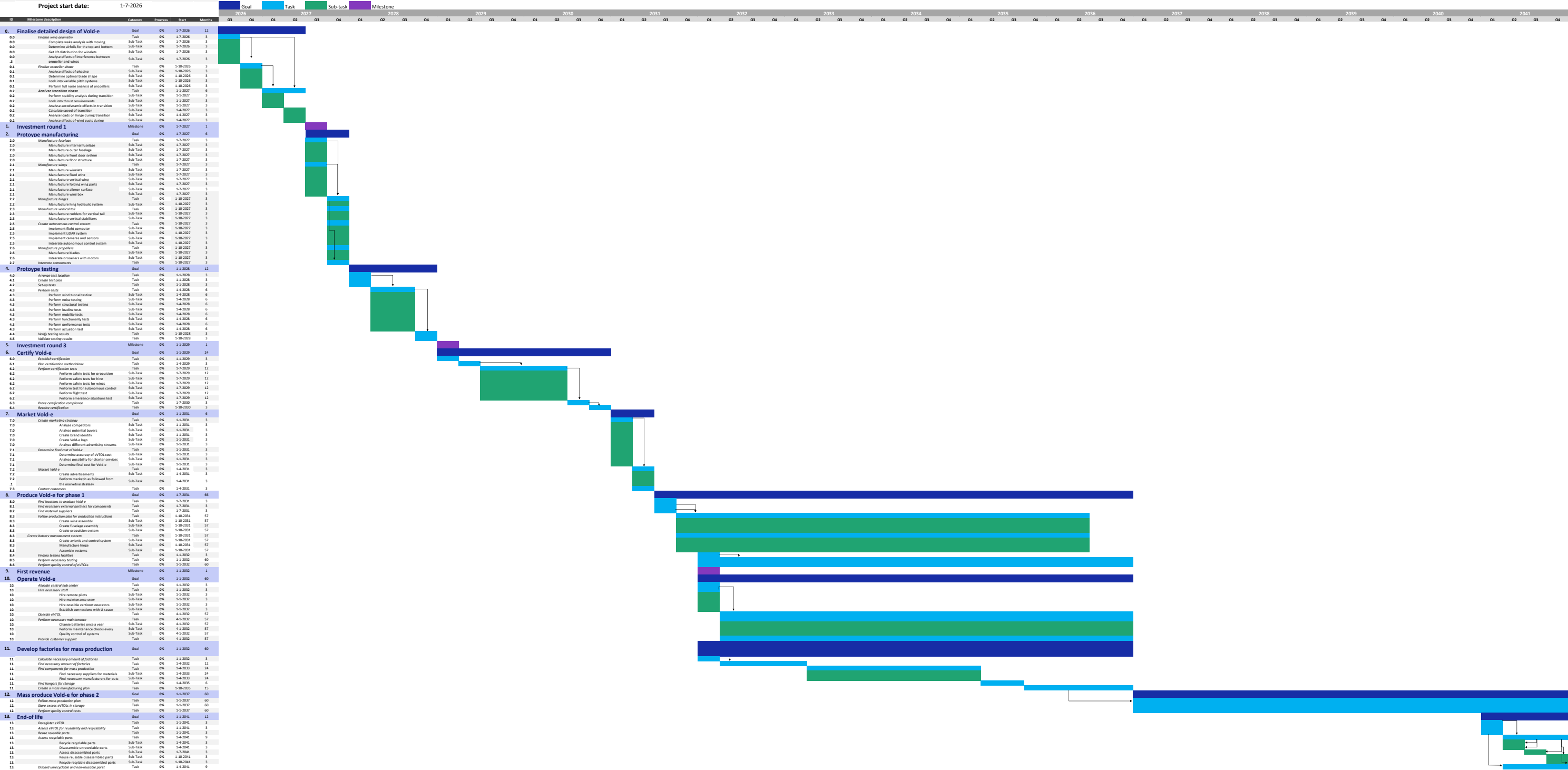
Parameter	Vold-e	Joby ¹
Cost of production [Million €]	2.0	1.3
Selling price [Million €]	3.4	1.8
Operational cost per flight [€]	242	190
Ticket price per km [€]	1.7	3.0 (JFK to Manhattan)

¹Risk Premium, <https://riskpremiumresearch.substack.com/p/joby-unit-economics>, Accessed on 23rd of June 2026





Post DSE Design and Development phases



15 Conclusion

In conclusion, this report presents the final design of Vold-e, a fully electric autonomous Prandtl-plane eVTOL intended for low-noise, zero in-flight emission transport across cities and geographical barriers. The design was developed around the reference mission from Nice to Courchevel, which was selected due to its 200 km range, mountainous terrain, and high-altitude destination. The final aircraft is designed to carry four passengers at a cruise speed of 200 [km/h], while remaining within the imposed ground footprint constraint. The selected configuration can be seen in Figure 15.1. This concept was chosen after a trade-off between several candidate configurations, where aerodynamic efficiency, structural performance, stability, controllability, sustainability and noise were considered. The Folding Prandtl configuration was found to provide the best overall compromise between cruise efficiency and compact ground operation, as it allows the aircraft to benefit from the aerodynamic characteristics of a box-wing layout, while the folding mechanism allows the span to be reduced during take-off, landing and ground handling. This directly addresses one of the main challenges of inter-city eVTOL design: combining a high aspect-ratio wing with a compact footprint.

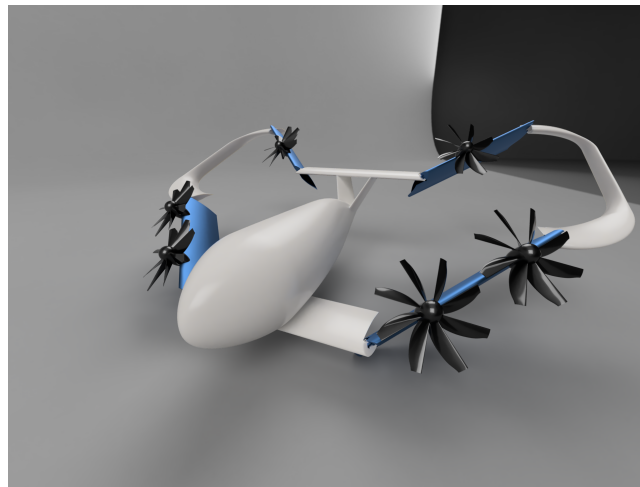


Figure 15.1: Final Vold-e Design during Transition

The most important design parameters can be seen in Table 15.1

Table 15.1: Summary of Vold-e's final key design parameters.

Category	Parameter	Value
Mission	Passengers	4
	Payload	400 [kg]
	Max range	200 [km]
	Cruise speed	200 [km/h]
Geometry	Wingspan	13 [m]
	Fuselage length	7 [m]
Category	Parameter	Value
Aerodynamics	Airfoil	NASA LS(1)-0417
	Cruise α	5.49°
Power & propulsion	Battery mass	540 [kg]
	Total energy	142.4 [kWh]
Mass	MTOW	1980 [kg]
Cost	Acquisition (phase 1)	2.0 [M€]
	Operational (phase 1)	242 [€/flight]

Throughout the detailed design phase, the selected configuration was sized and analysed across the main technical disciplines. The aerodynamics department defined the wing planform, airfoil selection, box-wing spacing, vertical tail and fuselage geometry. Power and propulsion sized the battery to complete the full mission including reserve energy. Vehicle dynamics assessed stability and controllability. In addition, the autonomous control architecture, and sensor layout were defined to support the intended autonomous operation. Structural sizing was performed for the main lifting surfaces, fuselage, landing gear, tail, hinges and folding mechanism, after which the results were integrated into an optimisation loop.

The optimisation process resulted in a feasible preliminary design with a MTOW of 1980 [kg]. The propulsion system was sized to have six variable-pitch rotors, allowing the aircraft to operate in both vertical and wing-borne flight. The final performance assessment showed that the aircraft can meet several key mission requirements, although some areas such as noise and transition performance require further investigation.

Although the final design satisfies the main objective of producing an Prandtl eVTOL concept, the result should be interpreted as a preliminary design rather than a certification-ready aircraft. Several methods used in the report are

based on first-order analytical models, empirical relations and simplified numerical tools. In particular, the transition phase, folding mechanism dynamics and propeller-wing interaction require more detailed analysis before the design can be considered mature.

Future work should therefore focus on validating the most critical assumptions and increasing the fidelity of the design. A starting point would be wind-tunnel testing or scaled flight testing to validate the predicted lift distribution, drag polar, stability derivatives and transition behaviour. A second priority is a more in depth structural development and testing of the folding mechanism. Since the folding mechanism is central to the concept, its reliability and structural integrity must be proven before the configuration can be further developed. Further development is also required for propulsion, battery, operational, and certification aspects. More detailed aeroacoustic modelling and rotor optimisation should be investigated.

Overall, Vold-e demonstrates that a Folding Prandtl eVTOL is a feasible and promising concept for urban air mobility where range, compact ground operation and low emissions are required. The design meets the main goals of the DSE by integrating aerodynamic, structural, propulsion, control, operational, financial and sustainability considerations into one coherent aircraft concept. At the same time, the design remains semi-preliminary and requires further validation before practical implementation. The most important outcome of this report is therefore not only the final e-VTOL configuration, but also the identification of the key technical challenges that must be solved to bring a compact, efficient, and sustainable Prandtl eVTOL closer to reality.

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A Ansys Fluent Simulations Setup

This appendix provides additional information on the setup of the simulations in ANSYS Fluent. Section A.1 describes the geometry preparation used to convert the detailed CAD model into a geometry suitable for CFD simulation, Section A.2 presents the meshing strategy, and Section A.3 outlines the solver setup.

A.1. Geometry

To prepare each CAD model for the CFD simulation, a fluid enclosure has to be created around the body to represent the external flow domain. Therefore, each model was enclosed in a rectangular computational box, with the domain dimensions defined relative to the reference length, L_{ref} , as shown in Table A.1. This ensured that the far-field boundaries remained sufficiently far from the vehicle while keeping the computational domain size manageable.

Table A.1: Computational domain dimensions used for the fuselage CFD simulations

Boundary direction	Distance from body	Purpose
Upstream	$3L_{\text{ref}}$	Allows the inlet flow to develop before reaching the body
Downstream	$7L_{\text{ref}}$	Allows the wake to develop before reaching the outlet
Side boundaries	$3L_{\text{ref}}$	Reduces lateral blockage effects
Top boundary	$3L_{\text{ref}}$	Reduces far-field boundary influence
Bottom boundary	$3L_{\text{ref}}$	Reduces far-field boundary influence

The fluid domain was generated by subtracting the vehicle body from the enclosing box, leaving only the surrounding air volume for meshing and simulation. The resulting internal model surface was assigned as the wall boundary, while the outer box faces were assigned as the inlet, outlet, side, top and bottom boundaries.

Fuselage Geometry Preparation

Since the fuselage geometries were relatively simple and did not include small external details, openings, protrusions or complex intersections, no additional geometry simplification or de-featuring was required before meshing.

Each fuselage was enclosed in a rectangular computational box described in Table A.1, with the fuselage length used as the reference length, $L_{\text{ref}} = L_{\text{fus}}$. As a result, the physical size of the test volume varied between the different designs, while the boundary distances remained consistent relative to the corresponding fuselage length. This reduced the influence of far-field boundary placement while avoiding an unnecessarily large domain for the shorter fuselage configurations. The same geometry was used for the selected fuselage after the trade-off process, when its final aerodynamic characteristics were assessed, the increased fidelity of the final analysis was introduced only from the meshing and solver setup onwards.

Final Design Geometry Preparation

The final aircraft geometry required more extensive preparation than the isolated fuselage geometries before it could be used for CFD analysis. The detailed CAD model included several small features, such as hinge details, openings, small gaps and local geometric discontinuities. These features are relevant for the mechanical design, but they can create problems during mesh generation by producing very small cells or poor-quality elements. Therefore, the final design geometry was simplified before meshing. Measures implemented to obtain simplified but representative external-flow model suitable for the final CFD analysis are concluded in Table A.2.

Table A.2: Main geometry simplifications applied for the CFD model

Detailed CAD element	CFD representation	Expected influence
Propeller assemblies	All propellers, hubs and spinners were removed.	Rotor slipstream, thrust and rotor-airframe interaction were not captured.
Vertical tail	The detailed vertical-tail geometry was excluded; only the simplified rear-wing support was retained.	Directional effects and local vertical-tail interference were not captured.
Transformation mechanisms	Hinges, actuator arms, pins, brackets and associated exposed components were removed.	Local drag and separation caused by these components were neglected.

Detailed CAD element	CFD representation	Expected influence
Gaps and discontinuities	Gaps, clearance regions, panel lines and small steps between moving components were closed or merged.	Leakage flow and highly localised separation effects were neglected.
Minor details	Small fasteners, attachment features, sharp sliver faces and minor protrusions were removed or simplified.	Small-scale parasite and junction-drag contributions may be underestimated.
Internal cavities	Hollow regions and small openings were sealed to create watertight bodies.	Possible internal flow paths were not modelled.

After simplification, a fluid enclosure was created around the complete aircraft to represent the external-flow domain. The test volume was created according to the dimensioning described in Table A.1 with the fuselage length used as the reference length, $L_{\text{ref}} = L_{\text{fus}} = 7$.

Furthermore, to reduce the computational cost of the final design simulations, only half of the aircraft geometry was analysed. The created test volume was cut along aircraft longitudinal symmetry plane, and a symmetry boundary condition was applied at the cut surface. This approach is valid for the analysed straight-flight cases, where the geometry and freestream conditions are symmetric. Using a half-domain allowed a finer mesh to be generated around the aircraft and in the wake region without increasing the total cell count excessively.

A.2. Meshing

This section describes the meshing procedures used for the CFD simulations. The mesh strategies were adapted to the purpose and required fidelity of each analysis, ranging from the computationally efficient fuselage trade-off meshes to the near-wall-resolved meshes used for the final characteristics assessments. The boundary-layer sizing method, individual meshing strategies and mesh independence study are described in the following subsections.

Boundary Layers Sizing

To generate an appropriate boundary-layer mesh, the first layer height, y_1 , was estimated from the target non-dimensional wall distance, y^+ . The required first layer height depends on the freestream conditions, fluid properties, reference length and selected target (y^+).

First, the Reynolds number was computed using the fuselage reference length, (L_f), the freestream velocity, (U_∞), the air density, (ρ), and the dynamic viscosity, (μ). Then, the turbulent skin-friction coefficient was then estimated using an empirical flat-plate relation [76].

$$Re = \frac{\rho U_\infty L_f}{\mu} \quad C_f = 0.058 Re^{-0.2} \quad (\text{A.1})$$

Using this skin-friction coefficient, the wall shear stress was approximated and then from the wall shear stress the friction velocity, (U_τ), was obtained [76]. Finally, the required first layer height was calculated from the definition of (y^+) [76] using Equation A.2. The target (y^+) values used for the different turbulence-modelling approaches are summarised in Table A.3.

$$\tau_w = \frac{1}{2} C_f \rho U_\infty^2 \quad U_\tau = \sqrt{\frac{\tau_w}{\rho}} \quad y_1 = \frac{\mu y^+}{\rho U_\tau} \quad (\text{A.2})$$

Table A.3: Target (y^+) values for the turbulence models used in the fuselage CFD simulations

Turbulence model / wall treatment	Target (y^+)
$k - \omega$ SST / near-wall resolved approach	(≈ 1)
Realizable $k - \varepsilon$ / scalable wall functions	(30-100)

Fuselage Trade-Off Meshing Strategy

The fuselage trade-off required a large number of CFD simulations, since 13 fuselage configurations were analysed at three angles of attack: $\alpha = 0^\circ$, $\alpha_{\text{cruise}} = 2.6^\circ$, and $\alpha = 5^\circ$. This resulted in a total of 39 simulations. Therefore, a computationally efficient meshing and solver approach was selected for the trade-off stage. A realizable $k - \varepsilon$ turbulence model with wall functions was used, as this allowed the different fuselage geometries to be compared at a reasonable computational cost while still capturing the main pressure and viscous drag trends.

The near-wall mesh was sized using the first-layer-height calculation procedure described in the previous section, Section A.2. For the trade-off simulations, a target value of $y^+ = 50$ was selected, which is suitable for a wall-function-based $k - \varepsilon$ setup. Since the Reynolds number depends on the reference length, and the reference length was taken as

the fuselage length, the required first layer thickness varied between the different fuselage designs. Therefore, the first layer height was calculated separately for each configuration. The resulting y^+ distribution for the fuselage selected after the trade-off is shown in Figure A.1 for reference. This figure shows that wall y^+ values of approximately 50 were obtained over most of the fuselage surface. The values remain largely within the intended wall-function range, with the highest values occurring at the front of the fuselage and the lowest values appearing near regions of stronger curvature at the end of the fuselage. Similar y^+ distributions were obtained for the remaining fuselage geometries considered during the trade-off process.



Figure A.1: Wall y^+ distribution of the fuselage selected after the trade-off prepared for the trade-off stimulation using $k - \epsilon$ model.

The only additional local mesh refinement applied during the trade-off was the inflation layer on the fuselage wall. The inflation setup used the calculated first layer thickness, 20 inflation layers and a growth rate of 1.2. This provided sufficient near-wall resolution for the selected wall-function approach while keeping the total cell count manageable. The remaining global meshing settings that were changed from the default setup are summarised in Table A.4.

Table A.4: Mesh settings used for the fuselage trade-off CFD simulations

Setting	Value	Purpose
Global element size	2 [m]	Sets the baseline mesh size for the computational domain.
Growth rate	1.2	Limits the rate at which the cell size increases away from refined regions.
Maximum size	4 [m]	Limits the largest allowed mesh elements in the domain.
Mesh defeaturing	No	Preserves the original fuselage geometry, since no simplification was required.
Capture curvature	Yes	Refines the mesh in regions with curved surfaces.
Curvature minimum size	2.0×10^{-3} [m]	Sets the smallest element size allowed by the curvature-based refinement.
Curvature normal angle	16 [°]	Controls how strongly curved regions are resolved.
Capture proximity	No	Avoids additional proximity refinement, as the fuselage geometries did not contain narrow gaps.
Inflation layers	20	Resolves the boundary-layer region near the fuselage wall.
Inflation growth rate	1.2	Controls the thickness increase between consecutive inflation layers.
First layer thickness / Target y^+	varying / 50	Provides a suitable near-wall mesh for the wall-function-based $k - \epsilon$ model.

Final Fuselage Meshing Strategy

After the fuselage trade-off, the selected geometry was analysed again using a higher-fidelity CFD setup. The same CAD geometry and external-flow domain used for the winning trade-off configuration were retained, so that the final results could be compared directly with the trade-off outcome. The changes were introduced only from the meshing stage onwards, mainly through a refined boundary-layer mesh suitable for the selected turbulence model.

For the final fuselage assessment, the $k - \omega$ turbulence model was used. This model requires a near-wall-resolved mesh, meaning that lower y^+ value was required. Therefore, the boundary-layer mesh was updated to target $y^+ \approx 1$. Following the first-layer-height estimation procedure described in Section A.2, a smaller first layer thickness was calculated and applied for the selected fuselage.

The global mesh settings were kept mostly consistent with the trade-off mesh in order to maintain a similar overall spatial resolution in the external-flow domain. The main refinement was applied to the inflation layer on the fuselage wall. The number of inflation layers was increased from 20 to 30, while the first layer thickness was reduced to satisfy the lower y^+ target. This allowed the boundary layer to be resolved more accurately, which is especially important for predicting viscous drag and skin-friction behaviour in the final fuselage analysis.

Final Design Meshing Strategy

Due to the geometric complexity of the complete aircraft model and the aim of obtaining realistic CFD results, a more structured refinement strategy was used to correctly capture the near-body flow field. This required a significantly

finer mesh than for the isolated fuselage simulations. The final mesh of the integrated design contained approximately 15 million cells.

The mesh refinement was built up in several levels, as detailed in Table A.5. Several local sizing methods were applied, including local refinement boxes and curvature-based refinement. Local refinement regions were placed around the front and rear wings using a small target mesh size, so that the lifting surfaces and their near-wake regions were resolved with sufficient detail. In addition, four larger refinement boxes were placed around the complete aircraft. These boxes were gradually coarsened away from the vehicle, providing a smooth transition from the fine near-body mesh to the coarser far-field mesh. The refinement layering is presented in the Figure A.2.

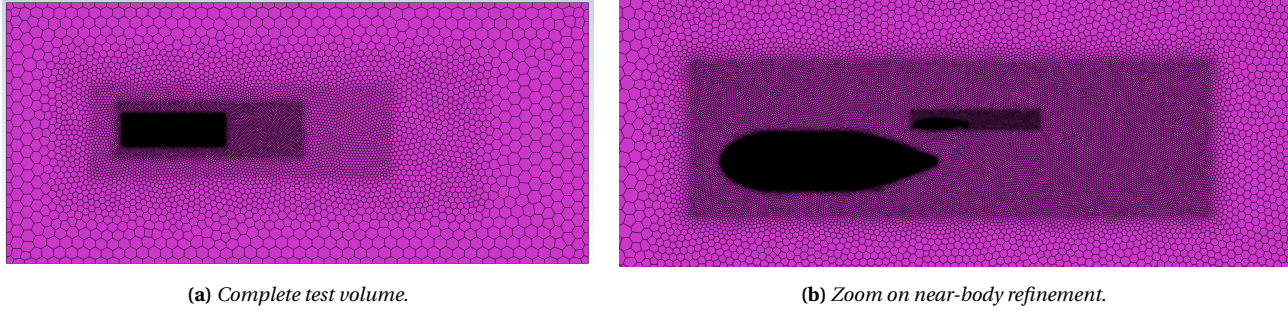


Figure A.2: Poly-hexcore mesh generated for the cruise configuration of the integrated final design.

A curvature-based local sizing control was also applied to the selected aircraft surface labels. This was used to preserve the surface curvature of the tail, winglets and other aerodynamic surfaces. The surface mesh was then generated using curvature and proximity size functions, while the volume mesh was generated using a poly-hexcore approach.

Boundary-layer inflation was generated with a first-layer height of 9.5235×10^{-6} . This value was determined using the method described in Section A.2, with a target of $y^+ = 1$ to obtain a near-wall-resolved mesh suitable for the SST $k-\omega$ turbulence model used in the final CFD assessment. The resulting wall y^+ distribution for the cruise configuration is shown in Figure A.3. The intended near-wall range was achieved across the aircraft surface, with most regions maintaining $y^+ < 1$ and only a minor exceed of the target value, reaching a maximum of $y^+_{\max} \approx 1.08$.

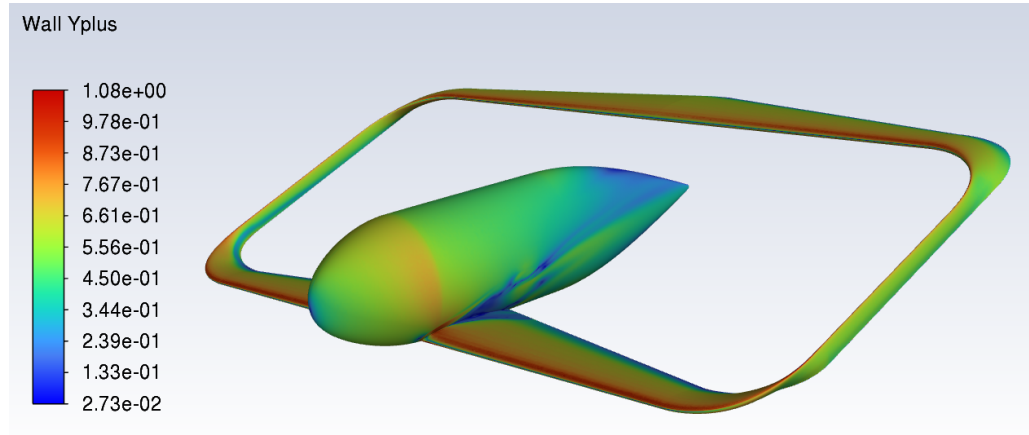


Figure A.3: Wall y^+ distribution of the final cruise-configuration mesh, showing near-wall-resolved values close to the target $y^+ = 1$.

Table A.5: Mesh settings used for the final design CFD simulations

Category	Function name	Parameters	Affected geometry
Local refinement	local-refinement-rear-wing	Type: Box Coordinate specification method: Ratio relative to geometry size Mesh size: 0.032 Select by: label Selected label: rear-wing $X_{\min} = 0.1$, $X_{\max} = 1.5$ $Y_{\min} = 0.1$, $Y_{\max} = 1.0$ $Z_{\min} = 0.5$, $Z_{\max} = 0.0$	Rear wing and its immediate near-field region

Category	Function name	Parameters	Relevant geometry
	local-refinement-front-wing	Type: Box Coordinate specification method: Ratio relative to geometry size Mesh size: 0.032 Select by: label Selected label: front-wing $X_{\min} = 0.1$, $X_{\max} = 1.5$ $Y_{\min} = 0.1$, $Y_{\max} = 1.0$ $Z_{\min} = 0.5$, $Z_{\max} = 0.0$	Front wing and its immediate near-field region
	local-refinement-global-1	Type: Box Coordinate specification method: Directly specify coordinates Mesh size: 0.064 $X_{\min} = -5$, $X_{\max} = 12$ $Y_{\min} = -2$, $Y_{\max} = 3$ $Z_{\min} = -7.5$, $Z_{\max} = 0$	Fine aircraft near-field region
	local-refinement-global-2	Type: Box Coordinate specification method: Directly specify coordinates Mesh size: 0.256 $X_{\min} = -6$, $X_{\max} = 25$ $Y_{\min} = -4$, $Y_{\max} = 5$ $Z_{\min} = -10$, $Z_{\max} = 0$	Intermediate near-wake and surrounding aircraft region
	local-refinement-global-3	Type: Box Coordinate specification method: Directly specify coordinates Mesh size: 0.512 $X_{\min} = -10$, $X_{\max} = 40$ $Y_{\min} = -7$, $Y_{\max} = 8$ $Z_{\min} = -15$, $Z_{\max} = 0$	Main wake refinement region
	local-refinement-global-4	Type: Box Coordinate specification method: Directly specify coordinates Mesh size: 1.024 $X_{\min} = -15$, $X_{\max} = 55$ $Y_{\min} = -12$, $Y_{\max} = 12$ $Z_{\min} = -12$, $Z_{\max} = 0$	Outer wake and transition region towards the far field
Local sizing	curvature_1	Size control type: Curvature Local minimum size: 5.0×10^{-4} Maximum size: 0.016 Growth rate: 1.2 Curvature normal angle: 9° Scope to: faces and edges Select by: label Selected labels: aircraft surface labels	Curved aircraft surfaces
Surface mesh	GeneratetheSurfaceMesh	Minimum size: 0.002 Maximum size: 2.048 Set volume mesh max size: No Growth rate: 1.2 Size functions: Curvature and Proximity Curvature normal angle: 18° Cells per gap: 1 Scope proximity to: faces and edges Separate boundary zones by angle: No	All aircraft and domain surfaces
Inflation	last-ratio_1	Add boundary layers: Yes Offset method type: Last ratio Number of layers: 20 Transition ratio: 0.272 First height: 9.5235×10^{-6} Add in: fluid regions Grow on: all aircraft surfaces	Aircraft boundary layer
Volume mesh	GeneratetheVolumeMesh	Solver: Fluent Fill with: Poly-hexcore Peel layers: 1 Minimum cell length: 5.0×10^{-4} Maximum cell length: 2.048 Parallel meshing: Enabled	Fluid domain surrounding the aircraft

Mesh Independence Study

The mesh independence study was performed for the final design mesh. The mesh produced following the steps described in previous section was retained as the medium mesh for the mesh independence study. Coarse and fine variants were generated by uniformly scaling the principal local-refinement, surface and volume mesh sizes by a refinement factor of 1.4, resulting in mesh levels of $1.4h$, h and $h/1.4$, respectively.

The inflation-layer settings, including the first-layer height, number of layers and growth behaviour, were kept unchanged for all mesh levels. Consequently, the near-wall resolution and resulting y^+ range were not affected by the mesh refinement, ensuring that all meshes remained suitable for the near-wall-resolved SST $k-\omega$ turbulence model. The detailed mesh settings applied to each level are summarised in Table A.6.

A.3. Solver Setup

The CFD simulations were performed using the Ansys Fluent solver. For the fuselage trade-off, each candidate geometry was simulated at three angles of attack, namely $\alpha = 0^\circ$, $\alpha = 2.6^\circ$ and $\alpha = 5^\circ$, all at the cruise speed of 55.6 m/s. The final aerodynamic assessment for the selected fuselage was then performed only at the preliminary design cruise condition.

The final aerodynamic assessment for the complete aircraft, was first performed over a wider angle-of-attack range (angle of attack varied from $\alpha = -4^\circ$ to $\alpha = 22^\circ$) to obtain the aerodynamic polars and associated characteristic parameters. This allowed to determined the approximate cruise angle of attack: $\alpha_{\text{cruise}} = 5.49^\circ$. Then additional ANSYS Fluent simulations were performed for the cruise angle of attack using the same remaining numerical setup.

In all simulations, the freestream velocity was set equal to the cruise velocity, $V_{\text{cruise}} = 55.6$ m/s and the atmospheric properties were taken at the cruise altitude of 12,500 ft. The detailed solver settings used for the all CFD simulations are summarised in Table A.7.

Table A.7: CFD solver setup for the fuselage trade-off, final fuselage assessment and final design cruise assessment

Setup entry	Fuselage trade-off	Final fuselage characteristics assessment	Final design characteristics assessment: cruise
Solver	Realizable $k-\epsilon$ with scalable wall functions	SST $k-\omega$	SST $k-\omega$
Pressure-velocity coupling	SIMPLE		
Spatial discretisation	Gradient: least squares cell based Pressure: second order Momentum: second-order upwind Turbulent kinetic energy: second-order upwind (Specific) dissipation rate: second-order upwind Pseudo time method: off		
Boundary conditions, $\alpha = 0^\circ$	Inlet: velocity inlet, $V = 55.6$ m/s Outlet: pressure outlet Top, bottom, right and left: symmetry Fuselage surfaces: wall, no slip	N/A	Inlet: velocity inlet, $V = 55.6$ m/s Outlet: pressure outlet Top, bottom, right and left: symmetry Vehicle surfaces: wall, no slip
Boundary conditions, angled case	Inlet: velocity inlet, $V = 55.6$ m/s, with flow angle prescribed Outlet: pressure outlet Top: pressure outlet Bottom: velocity inlet, $V = 55.6$ m/s, with same flow angle as inlet Right and left: symmetry Fuselage surfaces: wall, no slip	Inlet: velocity inlet, $V = 55.6$ m/s, with flow angle prescribed Outlet: pressure outlet Top: pressure outlet Bottom: velocity inlet, $V = 55.6$ m/s, with same flow angle as inlet Right and left: symmetry Fuselage surfaces: wall, no slip	Inlet: velocity inlet, $V = 55.6$ m/s, with flow angle prescribed Outlet: outlet and top boundary set as pressure outlets Top: positive α: pressure outlet; negative α: velocity inlet, $V = 55.6$ m/s, same flow angle Bottom: positive α: velocity inlet, $V = 55.6$ m/s, same flow angle; negative α: pressure outlet Right and left: symmetry Vehicle surfaces: wall, no slip
Working fluid / material	Density $\rho = 0.836$ kg/m ³ ; dynamic viscosity $\mu = 1.67 \times 10^{-5}$ kg/(m s); ratio of specific heats $\gamma = 1.4$.		
Operating conditions	Operating pressure $P = 63200$ Pa; temperature $T = 263.4$ K; freestream velocity $V = 55.6$ m/s.		
Energy equation	Energy equation: OFF. $M \approx 0.17$ and the flow is assumed incompressible.		
Gravity	Gravity: OFF. Gravitational body forces are not relevant for the external aerodynamic force prediction in this setup.		
Convergence criteria	Residual convergence criterion = 10^{-3} for continuity, momentum and turbulence quantities.		
Reference values	Reference area: $S_{\text{ref}} = S_{\text{fus,frontal}} = \pi \cdot 0.9^2 = 2.54$ m ² Reference length: fuselage length, varying per design	Reference area: $S_{\text{ref}} = S_{\text{fus,frontal}} = \pi \cdot 0.9^2 = 2.54$ m ² Reference length: fuselage length, $l = 7$ m	Reference area: $S_{\text{ref}} = S_{\text{wing,total}} = 31.33$ m ² Reference length: mean aerodynamic chord, $l = \text{MAC} = 1.262$ m

Solver Independence Study

A turbulence-model independence study was performed using the standard medium mesh selected from the mesh independence study. The SST $k-\omega$ model used for the final simulations was compared with the Realizable $k-\epsilon$ model with enhanced wall treatment and the Spalart-Allmaras model. All other simulation settings, including the geometry, mesh, boundary conditions, numerical schemes and convergence criteria, were kept identical so that differences in the predicted aerodynamic coefficients could be attributed only to the turbulence model.

