Conceptual Design of a Low-Complexity Ultralight Aircraft

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Ultralight aircraft represent a unique segment of general aviation by allowing powered flight under reduced regulatory constraints when operated and manufactured in compliance with Title 14 CFR Part 103. These reduced requirements significantly lower both the cost and complexity associated with aircraft ownership and operation, improving accessibility to aviation. This paper presents a conceptual design of a low-complexity ultralight aircraft intended to further minimize manufacturing and operational costs while maintaining performance comparable to existing ultralight designs. This design is driven by regulatory, weight, and performance constraints inherent to Part 103 aircraft and emphasizes simplicity in configuration and systems. An "inside out" conceptual design methodology is applied to the primary aircraft components, including the fuselage, wing, and empennage. Preliminary analyses are conducted to estimate key design parameters such as wing loading, power-to-weight ratio, center-of-gravity limits, and tail sizing using classical aircraft design methods based on reference aircraft data and the design specifications. The resulting configuration demonstrates that a low-complexity design approach can satisfy ultralight performance requirements while remaining manufacturable using cost-effective construction techniques. This work provides a foundational design framework that may serve as a baseline for future refinement, prototyping, and student-led development of accessible ultralight aircraft.

I. Introduction

Ultralight aircraft represent a unique category of general aviation defined by regulatory constraints intended to promote simplicity, accessibility, and safety. In the United States, powered ultralight vehicles are governed by Title 14 CFR Part 103, which limits empty weight to 254 lb, fuel capacity to 5 gallons, maximum level-flight speed to 55 knots, and power-off stall speed to 24 knots [1]. These constraints eliminate certification and pilot licensing requirements, significantly reducing barriers to aircraft ownership and operation. The regulatory framework drives ultralight designs toward minimal structural complexity, lightweight construction, and modest propulsion systems. As a result, ultralights serve as a practical introductory platform for general aviation, and as a feasible project for student-lead development efforts. This paper presents the conceptual design of a low-complexity, single-seat ultralight aircraft developed to comply with Part 103 requirements while performing similar to existing market offerings. Emphasis is placed on structural simplicity, manufacturability, and reduced system complexity.

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II. Requirements and Fuselage Design

Ultralight aircraft in the United States are regulated under Title 14 CFR Part 103, which establishes strict performance and operational limits intended to ensure simplicity and minimize regulatory burden [1]. Aircraft meeting these criteria are not subject to airworthiness certification, registration, or pilot licensing requirements. Compliance with these constraints is therefore essential; exceeding any of these limits would fundamentally change the regulatory classification of the aircraft and increase both development and ownership costs. In addition to regulatory constraints, performance requirements are established through analysis of current production ultralight aircraft. A survey of several dozen fixed-wing ultralight models provided baseline values for useful load, range, and operational ceiling. The primary requirement is payload capacity. While the average useful load among reference aircraft is approximately 286 lb, this design targets a useful load of 300 lb to improve accessibility and broaden the potential pilot demographic. To eliminate ambiguity between useful load and payload, the final requirement specifies a payload of 270 lb and a fuel weight of 30 lb (corresponding to 5 gallons). A mission range requirement of 150 miles was selected to be comparable to other ultralights. A maximum operational altitude of 10,000 ft was established, which provides sufficient operational flexibility while maintaining realistic performance expectations. The summarized project requirements and constraints can be seen in Table 1.

Table 1: Ultralight Constraints and Requirements

Constraints		
Type	**Value**	**Unit**
Occupancy	1	persons
Dry Weight	254	lbs
Fuel Capacity	5	gal
Max Airspeed	55	knots
Max Stall Speed	24	knots
Requirements		
Type	**Value**	**Unit**
Payload (Useful Load)	300	lbs
Range	100	mi
Altitude	10,000	ft

A. Mission Design

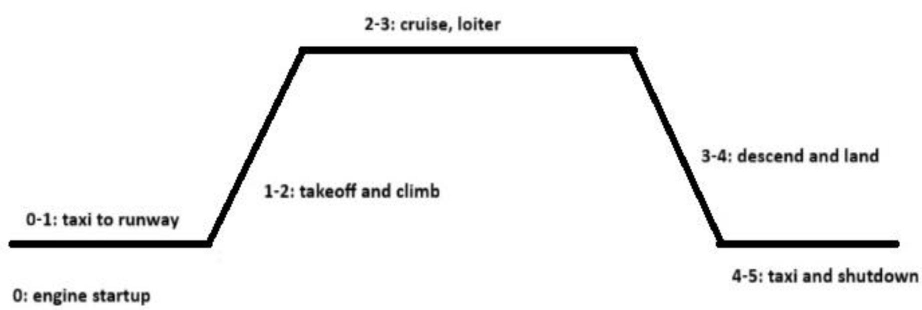

Figure 1: Ultralight Mission Profile

The design mission consists of a short-duration recreational flight departing from and returning to a grass airstrip. The total mission duration is approximately 60 minutes with a target range of 150 miles. The mission, Figure 1, is divided into the following phases:

0. Engine start and warm-up
1. Taxi to runway
2. Takeoff and climb
3. Cruise with optional loiter
4. Descent and landing
5. Taxi and shutdown

Although the ground operation phases contribute minimally to fuel consumption, they are included in the mission definition to ensure complete weight fraction accounting. The takeoff and climb segments drive power loading requirements, while the cruise segment dominates fuel fraction and aerodynamic efficiency considerations. The descent and landing phases primarily influence stall speed constraints and wing loading limits. The cruise segment is the primary driver of fuel sizing, as it accounts for most of the mission distance.

B. Concept Generation

The selected configuration adopts a conventional high-wing tractor layout with a single vertical tail and fixed landing gear, consistent with many existing ultralight aircraft. Because the fuselage lacks a fully enclosed cross-section, careful consideration must be given to bending and torsional stiffness to ensure structural adequacy. A defining feature of the configuration is a "two dimensional" fuselage constructed from a lightweight truss framework. This framework defines the nose, cockpit, and tail geometry while minimizing material usage. The layout is well represented among existing ultralights, providing validated stability characteristics and predictable flight behavior. Structural simplicity is treated as a primary design objective due to the 254 lb empty weight limitation imposed by Part 103.

C. Fuselage Design

An inside-out design methodology is employed, using a design pilot modeled as a 6 ft 3 occupant. Cockpit dimensions are established to provide adequate seating space, control reach, and forward visibility within the constraints of a lightweight structure.

The resulting fuselage maintains an approximate diameter of 4 ft in the cockpit region. Slenderness ratios are selected to define lengths of the fuselage, nose and tail: 0.25, 0.75, and 0.5 respectively. Using these ratios, the fuselage length is divided into a 3 ft nose, a 5 ft cockpit, and an 8 ft tail. This yields a total fuselage length of 16 ft. The tail geometry incorporates a 10° ground clearance angle and a 24° afterbody divergence angle to reduce aerodynamic separation while preserving propeller and ground clearance. The seat is sized to 20 in depth, 20 in height, and 22 in width, with an installation angle of 7°. From the estimated pilot eye position, the configuration provides an over-nose visibility angle of approximately 15°. While upward visibility is limited, the open-frame geometry minimizes the effect. A conventional taildragger landing gear configuration is selected to simplify controls and improve propeller ground clearance during takeoff and landing from grass surfaces. Landing gear geometry is defined relative to the estimated center of gravity location. In the pitch plane, the main gear is positioned 25° aft of the center of gravity and approximately 25 in below the fuselage reference line. In the roll plane, the main gear contact points are located 20° forward of the center of gravity, approximately 48 in aft of the nose. The geometric relationship between the main gear and tail wheel establishes a 10° ground attitude angle, ensuring adequate propeller clearance, stable ground handling characteristics, and approximately 6 in of fuselage ground clearance.

III. Preliminary and Propulsion Sizing

This section presents the preliminary sizing methods used to establish the aircraft's wing loading, power loading, fuel fraction, and propulsion requirements. The analysis employs a W/S–W/P design space to identify a feasible design point that satisfies regulatory and performance requirements. The first stage of the sizing process evaluates mission fuel requirements using weight fractions associated with each phase of flight. Normal flight segments, including startup, taxi, takeoff, climb, descent, and landing, are modeled using representative weight ratios for light homebuilt aircraft. The cruise and loiter segments are evaluated using propeller-driven range relations to determine the fuel fractions. These calculations yield an estimate of total fuel weight and establish the initial maximum takeoff weight. With an estimated takeoff weight defined, constraint analysis is conducted in the wing loading (W/S) versus power loading (W/P) domain. The design space is bound by the stall speed, take off distance, landing performance, climb rate and gradient, and cruise speed. The intersection of these constraints defines the design region where a design point can be selected. This process determines the required wing area and engine power for the ultralight. Following aerodynamic sizing, propulsion is sized, with engines commonly used in ultralight aircraft being evaluated. This sizing approach ensures that all requirements and constraints are addressed simultaneously.

D. Initial Weight Estimation

Fuel weight is estimated using a weight fraction approach consistent with classic aircraft sizing methods. The mission profile is divided into discrete segments including startup, taxi, takeoff, climb, cruise, loiter, descent, and landing. Non–fuel-intensive phases (startup, taxi, climb, descent, landing) are modeled using statistical weight fractions representative of light homebuilt and ultralight aircraft. Cruise and loiter fuel fractions are determined using the propeller-driven range relation. This approach captures both mission geometry and aerodynamic performance effects, rather than assuming a fixed fuel percentage.

The maximum takeoff weight is determined by combining empty weight, payload weight, fuel weight, and trapped fuel fraction effects. The Part 103 empty weight limit of 254 lb represents a maximum allowable structural and installed engine weight. This value is therefore adopted as the initial empty weight target. The resulting take off weight is 577 lb. This takeoff weight is consistent with observed values among reference ultralight aircraft.

Figure 2 compares empty weight versus takeoff weight for surveyed ultralight aircraft, along with a linear regression trendline. The high coefficient of determination indicates a strong linear relationship between empty weight and takeoff weight within the reference dataset. The resulting point lies just below the trendline, indicating that the proposed configuration demonstrates feasibility.

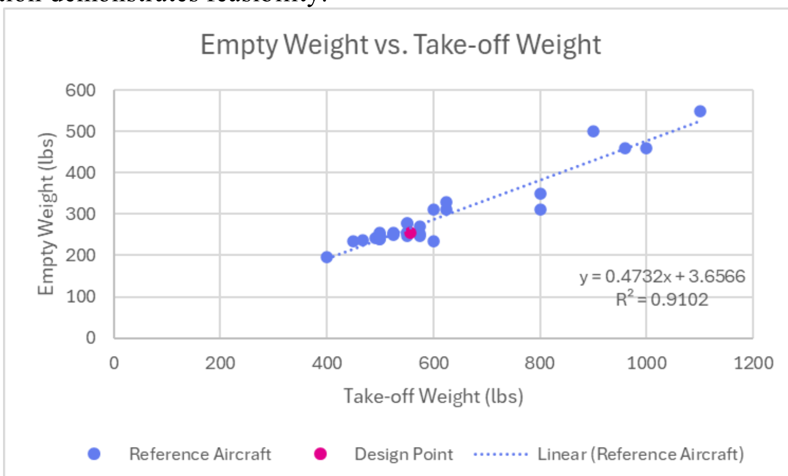

Figure 2: Reference Empty Weight vs Take Off Weight

E. Weight to Power and Wing Loading

Constraint analysis is performed in the wing loading (W/S) versus power loading (W/P) design space to determine a feasible configuration that satisfies regulatory and mission performance requirements. This method allows aerodynamic, takeoff, landing, climb, and cruise constraints to be evaluated simultaneously, with the intersection of constraints defining an allowable design region. The W/P – W/S diagram is shown in Figure 3.

The stall speed limitation imposed by FAR Part 103 establishes an upper bound on allowable wing loading. For a range of assumed lift coefficient values (1.2–2.0), vertical lines are generated in the W/S domain. Higher maximum lift coefficients allow greater wing loading, reducing required wing area. The stall constraint forms the left boundary of the feasible design region. Takeoff performance is modeled using the Takeoff Performance Parameter (TOP) formulation for propeller-driven aircraft. This TOP is defined empirically by reference aircraft. A takeoff distance of 150 ft from a grass runway is imposed, with a grass surface correction factor incorporated into the analysis. This constraint produces downward-sloping curves in the W/P–W/S space, limiting power loading at higher wing loadings. Landing distance requirements impose an additional upper bound on wing loading. Using statistical data, this constraint appears as additional vertical bounds in the design space and primarily influences maximum allowable wing loading. Climb performance is evaluated using excess power relations, where maximizing climb rate requires maximizing $C_L^{3/2}/C_D$. Rewriting in terms of W/P and W/S produces upward-sloping constraint curves in the design space. Climb gradient constraints are similarly formulated using the ratio c/V. For the selected mission and regulatory envelope, climb constraints are not the primary limiting factors in defining the feasible region. FAR Part 103 limits maximum level flight speed to 55 knots [1]. The cruise constraint is derived from the steady level flight power requirement. This constraint defines an upper bound on W/P to ensure sufficient power for level flight at the regulatory maximum cruise speed. Although exceeding 55 knots is prohibited, the aircraft must be capable of reaching this speed with power margin. Engine output may be mechanically limited to comply with regulations.

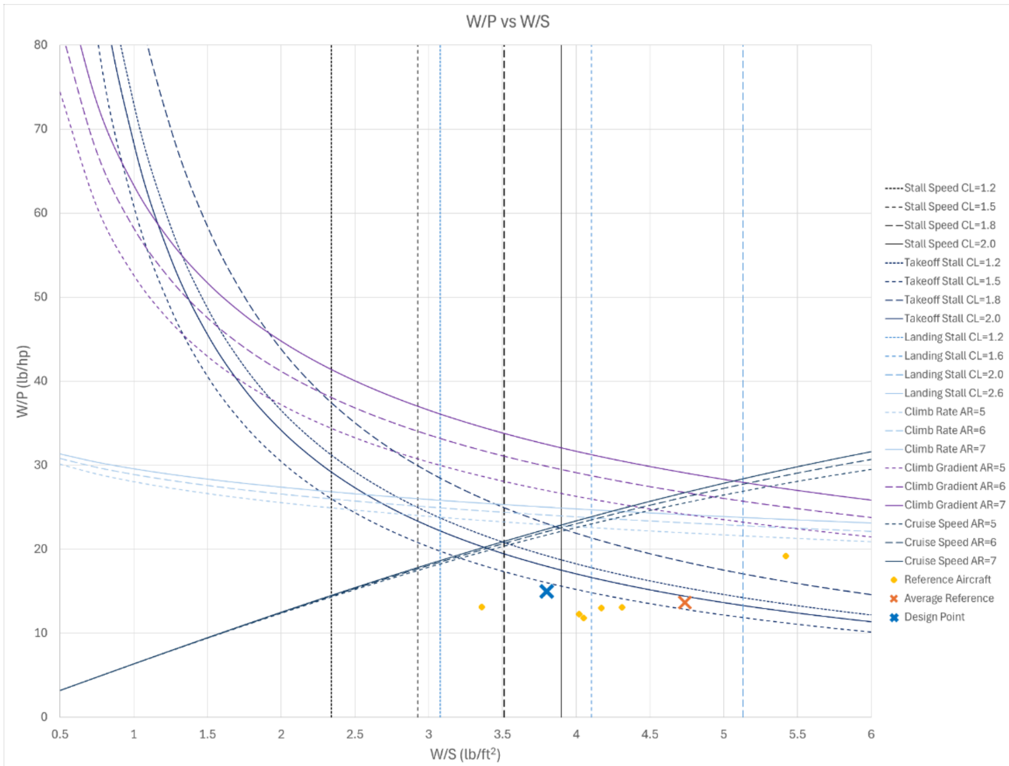

Figure 3: W/P - W/S Diagram

F. Design Region and Design Point

A design point is selected at W/S = 3 lb/ft^3 and W/P = 15 lb/hp. For a takeoff weight of approximately 557 lb, this corresponds to a required engine power of 37 hp and a wing area of 146 ft^2. Comparison with reference aircraft (shown on the diagram) indicates similar power loading but slightly lower wing loading. This reduction is attributed to conservative assumptions for maximum lift coefficient and regulatory stall speed compliance. The selected design point lies comfortably within the feasible region while maintaining adequate performance margin across all constraints.

G. Propulsion Sizing

Propulsion sizing is performed following selection of the design point in the W/S–W/P constraint analysis. At the selected wing loading of 3 lb/ft^2 and power loading of 15 lb/hp, the required installed power for a takeoff weight of approximately 557 lb is 37 hp. The maximum power requirement occurs during takeoff, calculated as approximately 37.1 hp. Cruise power is estimated assuming 90% of maximum power, yielding a cruise power of 33.8 hp. These values establish a minimum engine power of 37 hp with a modest margin.

IV. Wing Design

Wing sizing is performed to satisfy the selected wing loading from the constraint analysis while ensuring compliance with stall, takeoff, landing, and cruise performance requirements. Initial airfoil selection is guided by infinite wing analysis to determine the required section lift coefficient at cruise. To account for maneuver and gust margins during cruise, the lift requirement is conservatively increased by 10%, yielding a required lift force of 609 lb and a design lift coefficient of 0.396. This value represents the required section lift coefficient at cruise for the selected wing loading and design speed. Due to the low speed of the aircraft and the emphasis on manufacturability, no sweep or taper angle is incorporated into the wing.

H. Airfoil Selection

The Clark Y airfoil is selected based on the following considerations:

- Proven use in light aircraft and ultralight applications
- Moderate camber providing favorable low-speed lift characteristics
- Gentle stall behavior
- Relatively small pitching moment coefficient
- Compatibility with simple, flat-bottom construction techniques

The required design section lift coefficient of 0.396 falls within the efficient linear lift region of the Clark Y airfoil, ensuring cruise operation occurs well below stall while maintaining favorable drag characteristics. Airfoil performance is evaluated at representative cruise and stall Reynolds numbers. These Reynolds numbers are then used in JAVAFOIL to generate aerodynamic polars: Drag coefficient $C_d$versus Lift coefficient C_l (Figure 4) Lift coefficient $C_l$versus angle of attack (Figure 5) and Moment coefficient $C_m$characteristics (Figure 5) [2].

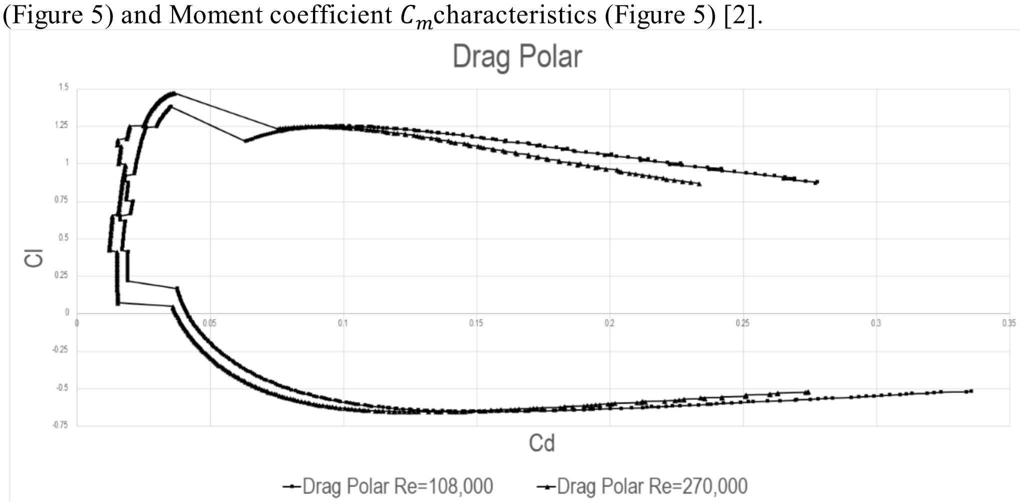

Figure 4: Clark Y Drag Polar

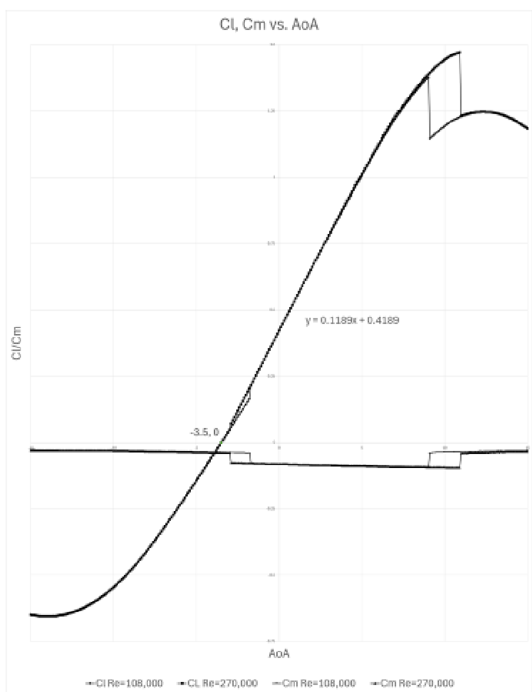

Figure 5: Clark Y Lift and Moment Curve

I. Finite Wing Corrections

An aspect ratio of 7 is selected to produce a highly efficient wing, resulting in a wingspan of 32 ft based on the planform area derived in the W/P – W/S diagram. Since the Sweep angle and taper ratio are set to 0 and 1 respectively, the chord is constant across the span, and the planform shape is rectangular. This results in a mean aerodynamic chord (MAC) equaling the constant chord of 4.56 ft. This also sets the spanwise location of the MAC equal to a quarter of the span, or 8ft. the aerodynamic center can be approximated as a quarter of the chord or 1.14 ft.

Because the aircraft operates at very low Mach numbers (< 0.1), the Prandtl–Glauert compressibility correction factor can be neglected. With these parameters established, the DATCOM method is employed to account for the 3D effects of a finite wing [3].

$$C_{L\alpha} = \frac{2\pi A}{2 + \sqrt{4 + \left(\frac{A\beta}{\eta}\right)^2 \left(\frac{tan^2 \Lambda_{c/2}}{\beta^2}\right)}}$$

The DATCOM method accounts for the effects of Mach, aspect ratio, and the sweep angle and can accurately predict the lift effectiveness of the wing, especially at low Mach numbers [3]. Substituting the parameters into the equation results in a lift effectiveness of 0.0799/deg at cruise, and 0.0797/deg at the landing velocity. The clean wing parameters can be seen in Table 2.

Table 2: Clean Wing Parameters

Parameter	Value	Unit
$C_{L\alpha,cruise}$	.0799	1 / degrees
$C_{L\alpha,landing}$	.0797	1 / degrees
α_{stall}	12.01	degrees
C_{Lmax}	1.23	

Using these values, a lift curve is created and is seen in Figure 6. The cruise and landing configuration are nearly identical and therefore there is no distinction between the two on the plot.

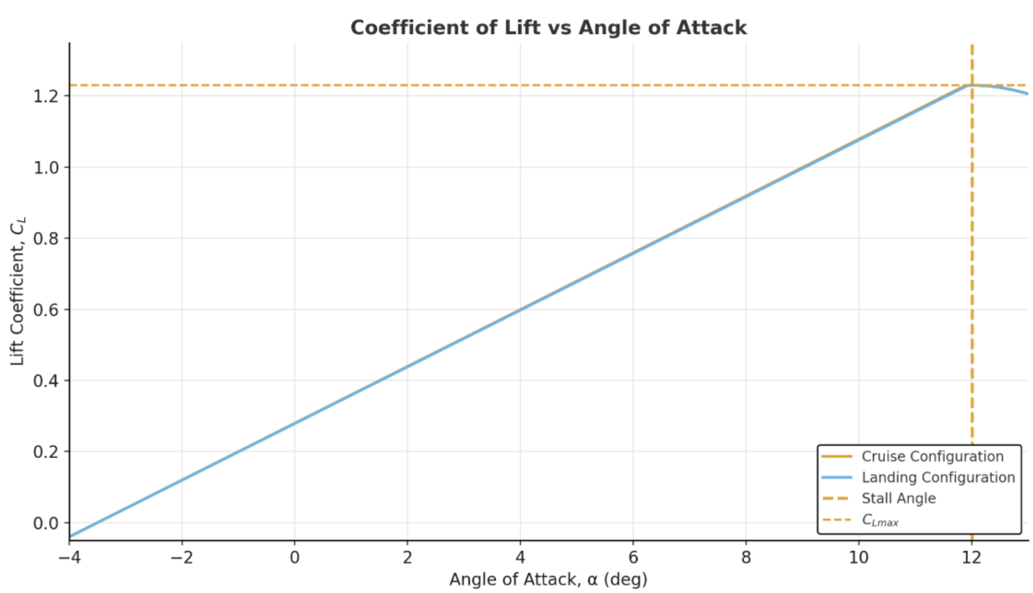

Figure 6: Ultralight Clean Wing Lift Curve

J. High Lift Devices

To satisfy the lift requirements for takeoff and landing, high lift devices are needed. Only trailing edge devices that do not alter the section chord length are examined to maintain the simplicity and manufacturability of the design. With a clean wing max lift coefficient of 1.23 and a required lift coefficient of 1.98, the high lift device must be able to increase the lift coefficient by 0.75. To achieve this increase, slotted flaps are incorporated into the wing. The resulting shift in the lift curve is seen in Figure 7.

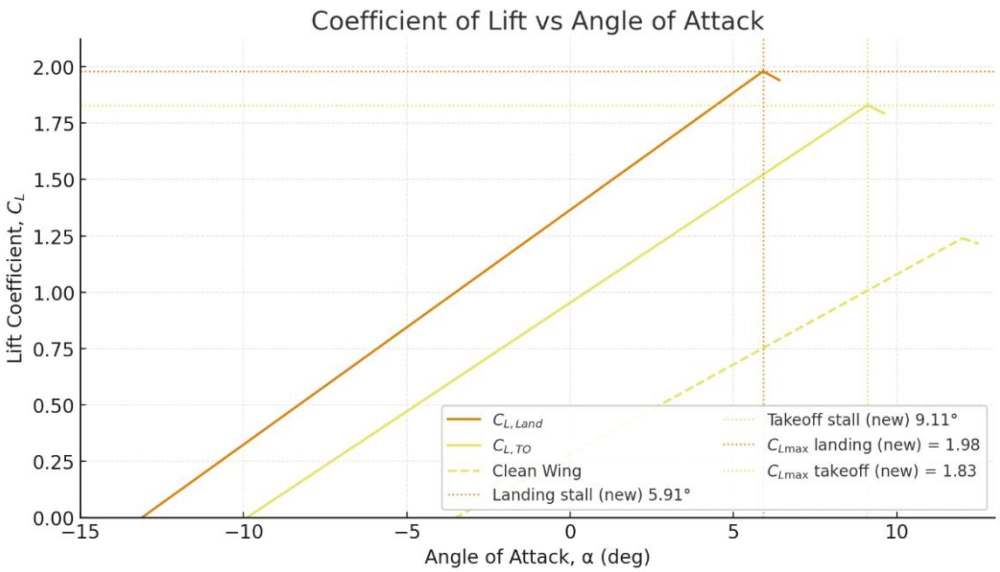

Figure 7: Ultralight Lift Curve with High Lift Devices

V. Center of Gravity Analysis and Tail Design

Following preliminary wing sizing, a Class II weight estimation is performed to refine component weight distributions and establish an accurate center-of-gravity (CG) location. These results are used to define the CG range under multiple loading conditions and to size the empennage using classical tail volume coefficient methods.

K. Tail Volume Estimation

For a preliminary sizing of the tail to be completed, the V bar method is used to approximate the overall size and shape of the horizontal and vertical stabilizers. Data from reference general aviation aircraft is used to compute V bar. These V bar ratios are then applied to the ultralight design. This preliminary sizing results in a horizontal stabilizer area of 19 ft^2 and a vertical stabilizer area of 16 ft^2. While the horizontal stabilizer is revisited for further refinement, coupling effects between roll and yaw require complex methods to accurately model; therefore, the vertical stabilizer geometry is estimated based on this preliminary sizing. Like other available designs, the vertical tail has an aspect ratio of 1.6 and a taper ratio of 0.7.

L. Load Factor

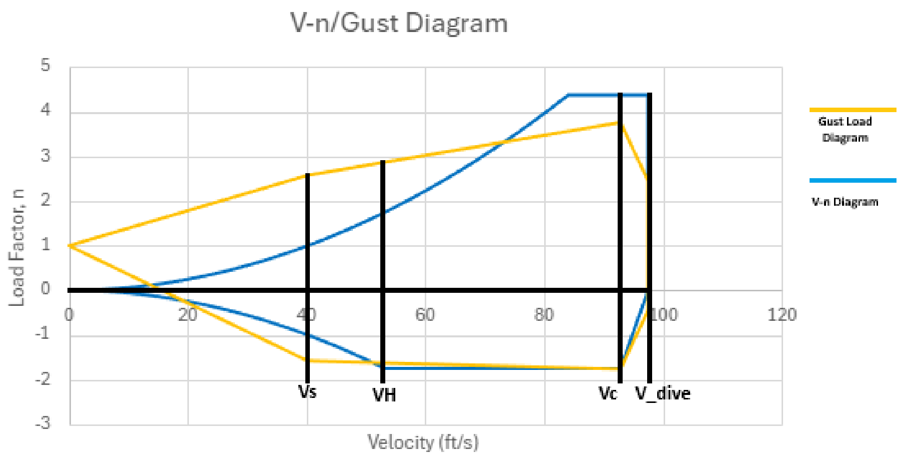

Figure 8: V - n and Gust Loading Diagram

Figure 8 presents the composite V – n and Gust loading diagram and represents the flight envelope the ultralight must be able to operate within. The maximum load factor ranges from –1.76 to 4.37at cruise conditions. A factor of safety of 1.5 is applied to the maximum load factor, for an ultimate load factor of –2.64 to 6.56.

M. Class II Weight Estimation

Now that the geometric properties of the major aircraft components are established, and the structural requirements of the airframe are considered via the load factors, empirical methods are used to generate a more accurate weight estimation of the aircraft. For this paper, the Corke method is employed [4]. The Corke method consists of a series of relationships for each component, that are functions of vehicle parameters and coefficients that are derived from empirical data of a large sample of a category of aircraft [4]. Coefficients related to general aviation aircraft are used in this estimation.

Table 3: Corke Class II Weight Estimation

Component	Wing	H Stabilizer	V Stabilizer	Fuselage	Main Gear	Tail Gear	Engine	Misc	Total
Weight (lb)	131.3	7.8	4.7	60.2	17.9	1.5	96.6	77.8	398

Table 4: Adjusted Corke Weight Estimation

Component	Wing	H Stabilizer	V Stabilizer	Fuselage	Main Gear	Tail Gear	Engine	Misc	Total
Weight (lb)	83.8	5	3	38.4	11.4	1	61.7	49.7	254

The accuracy of these weight estimations is heavily dependent on how well the empirical coefficients capture the class of aircraft. Despite being the closest related category general, general aviation aircraft vary greatly from ultralight aircraft in terms of weight. This is the likely cause of the large discrepancy between the Corke estimation, Table 3, and the design empty, with that discrepancy being a factor of 64%. For this design, the weight fractions of general aviation aircraft components are assumed to be equal ultralight aircraft, Table 4.

N. Center of Gravity Range

Next, the center of gravity (CG) is estimated for the wing group (wing + main gear), the fuselage group (fuselage + tail + engine + tail gear) using the method outlined by Torenbeek [5]. The location of the wing is varied as a function of the MAC and "loaded" with the payload and fuel to examine the range of CG at different wing positions. These ranges for the hourglass curve seen in Figure 9.

O. Stability and Control Analysis

Like the wing location, the stability and controllability of the aircraft can be represented as a function of the CG. This linearized relationship defines the ratio between the horizontal stabilizer and wing planform areas. Notice that stability and controllability are inversely related, with the stability curve increasing as the CG shifts farther down the MAC. The area above both curves represents the region where the aircraft is both stable and controllable. To minimize the tail size, both the CG range and scissor plot are projected on each other and aligned at $X_{CG} = 0$. The area between stability and control curves must intersect both extremes of the CG range. For the ultralight design, this occurs when the leading edge of the wing is located at 31% of the fuselage length, and the horizontal tail area is 23% of the wing area. The resulting wing location and horizontal tail area is 5 ft from the nose of the aircraft and 33.5 ft^2, respectively. The horizontal tail is assigned an aspect ratio of 1.6 and not taper, resulting in a rectangular tail with a span of 14 ft and a chord of 2.4 ft.

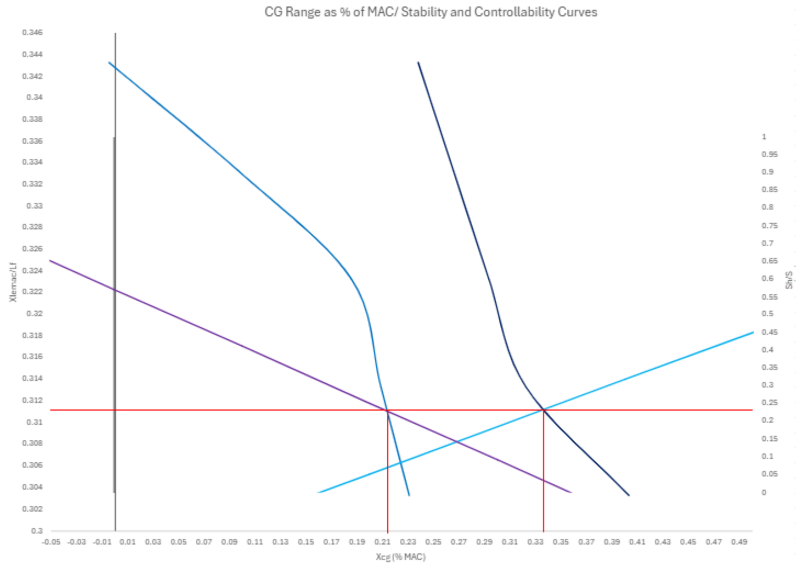

Figure 9: CG Range and Stability - Control Scissor Plot

VI. Improved Drag Estimation

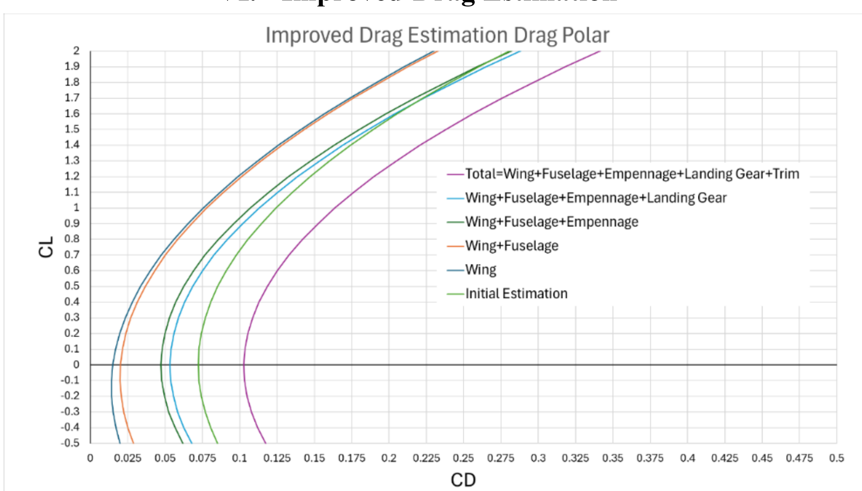

Figure 10: Improved Drag Estimation

Now that the major parameters of the aircraft are established, a final estimation of the drag is made. Figure 10 is the summation of each component and its drag seen on the ultralight design. The initial drag estimation can be seen as the green line with the improved drag estimation on the magenta line. Evaluating intersections of the initial and improved estimation and the landing lift coefficient (1.8) shows a difference in the drag coefficient of 0.055562, a percentage error of 23.07%, from the initial estimation. The individual component drag contributions can be seen in Table 5.

Table 5: Component Drag Contributions at C_L = 1.8

Component	Drag Contribution
Wing	0.19153
Fuselage	0.00239
Empennage	0.04377
Landing Gear	0.00600
Trim	0.05273

VII. Conclusions and Future Work

A conceptual design for a Part 103 compliant ultralight aircraft is developed using integrated sizing methods combining mission analysis, constraint evaluation in the W/S–W/P design space, aerodynamic refinement, and stability verification. The final configuration satisfies all regulatory constraints, including the 254 lbs empty weight limit, 24-knot stall speed requirement, 55-knot maximum cruise speed, and 5-gallon fuel capacity. The selected design point provides adequate takeoff performance from grass runways, positive climb capability, and sufficient static stability and controllability across the defined center-of-gravity envelope. Airfoil selection and high-lift device integration ensure stall compliance while maintaining acceptable cruise performance. Although the design achieves compliance and functional performance, the conservative stall and lift assumptions led to a relatively low wing loading. This resulted in a larger wing area than is typical of production ultralights. The increased wing area leads to secondary effects, including higher structural weight, increased drag, and larger empennage requirements to maintain stability. Additionally, the relatively short tail moment arm requires increased tail area to achieve an adequate stability margin, contributing to the aerodynamic and structural penalties. These characteristics do not invalidate the design but do indicate opportunities for optimization in subsequent iterations.

P. Future Work

Future development efforts will focus on refining the configuration to reduce structural weight and aerodynamic drag while preserving regulatory compliance. These modifications include:

- Increasing fuselage length to extend the tail moment arm, thereby reducing required tail area
- Incorporating a higher-lift airfoil (e.g., S1223) to increase $C_{L_{max}}$ and permit a higher wing loading
- Reducing wing aspect ratio to decrease wingspan and improve manufacturability

These changes are expected to allow selection of a more aggressive design point within the W/S – W/P constraint space, resulting in improved aerodynamic efficiency and reduced structural mass. Upon completion of the refined configuration, a scaled demonstrator is planned to experimentally evaluate static stability, controllability, and low-speed performance.

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