Gradient-Based Optimization of Solar-Regenerative High-Altitude Long-Endurance Aircraft

Taylor McDonnell* and Andrew Ning[†]
Brigham Young University, Provo, UT, 84602, USA

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In this paper we use gradient-based optimization to minimize the mass of a solar-regenerative high-altitude long-endurance (SR-HALE) flying-wing aircraft while accounting for nonlinear aeroelastic effects. We design the aircraft to fly year round at 35° latitude at 18 km above sea level and subject the aircraft to energy capture, energy storage, material failure, local buckling, stall, longitudinal stability, and coupled flight and aeroelastic stability constraints. The optimized aircraft has an aspect ratio of 27.8, a surface area of 99.1 m², and a mass of 508.8 kg. Our results suggest that thick airfoils provide greater structural efficiency than increased carbon fiber reinforced polymer (CFRP) ply thicknesses. We also perform several parameter sweeps to determine sensitivity to altitude, latitude, battery specific energy, solar efficiency, avionics and payload power requirements, and minimum design velocity.

I. Introduction

A long-envisioned goal in the aerospace community is the development of a solar-regenerative high-altitude long-endurance (SR-HALE) unmanned aerial vehicle (UAV) capable of sustaining year-round flight at high altitudes [1]. SR-HALE UAVs have the potential to provide capabilities similar to satellites in the areas of surveillance, communication, and environmental monitoring at a fraction of the cost [2]. These incentives have led many companies to pursue the development of SR-HALE aircraft (e.g., AeroVironment, Airbus, Boeing, Facebook, and Google) and have, in part, motivated many conceptual designs and/or optimizations on the subject.

Previous SR-HALE aircraft conceptual design and/or optimization studies include those performed by Cestino [3], Baldock and Mokhtarzadeh-Dehghan [4], Nickol et al. [5], Burton and Hoburg [6], Burton et al. [7], and Colas et al. [8,9]. Cestino performed a detailed preliminary design and analysis of a blended wing body SR-HALE configuration [3]. Baldock and Mokhtarzadeh-Dehghan used a simple conceptual design framework to assess the feasibility of a braced wing SR-HALE concept and found year-round SR-HALE aircraft flight to be feasible below 10° latitude [4]. Nickol et al. optimized many high-altitude pseudo-satellite (HAPS) concepts (including SR-HALE aircraft) to find the feasibility of each concept for hurricane science and communications relay missions [5]. While SR-HALE aircraft did not perform well compared to other HAPS concepts in that study (year-round flight was found to be infeasible, even with large technological advances), more recent optimizations have found the SR-HALE concept to be feasible. These studies include those performed by Burton et al. [6,7] and Colas et al. [8,9]. Burton and Hoburg used geometric programming to perform optimizations of SR-HALE aircraft which took just a few seconds each [6]. The speed of this approach allowed Burton et al. to generate many Pareto fronts which show the sensitivity of SR-HALE aircraft designs to various mission requirements and technological capabilities [7]. Colas et al., on the other hand, presented a more computationally expensive, but higher fidelity SR-HALE aircraft optimization approach in which the flexible aircraft analysis tool ASWING [10] was incorporated into a gradient-free optimization framework [8,9]. This optimization framework was then demonstrated on single-boom, twin-boom, and flying-wing SR-HALE aircraft.

In all of these studies, optimal SR-HALE aircraft designs tend towards high aspect ratio designs with low structural weight. High aspect ratio wings allow for increased aerodynamic efficiency by reducing induced drag. Lower aircraft mass allows SR-HALE aircraft to conserve energy by flying slower. As a result of this combination of features, SR-HALE aircraft tend to be flexible.

Motivated by the flexibility of SR-HALE aircraft, many re-

searchers have worked towards understanding the impact of extreme aircraft structural flexibility on aircraft performance. Patil and Taylor found that spanwise, non-uniform gusts invoke stronger structural responses in flexible flying wings than uniform gusts [11,12]. Shearer and Cesnik showed that linearized structural analysis is inappropriate for analyzing the asymmetric maneuvering of flexible aircraft [13]. Su and Cesnik concluded that flutter analyses should consider the whole vehicle's degrees of freedom, and that performing wind tunnel tests in a constrained model may not be sufficient for flexible aircraft flutter predictions [14]. Mardanpour et al. found that a non-negligible increase in a flexible aircraft's flutter speed occurs when engines are placed ahead of the wing's elastic axis [15,16]. Based on many of these findings, Cesnik et al. provided a brief critical review of many conventional structural design procedures applied to very flexible aircraft and provided three recommendations for flexible aircraft design environments [17]. First, the deformed aircraft geometry should be the baseline in weight, structural, and stability analysis. Second, coupling between aeroelasticity and flight dynamics needs to be considered. Third, transient dynamic simulations should include large nonlinear aircraft displacements.

Among the aforementioned studies focused on the conceptual design of SR-HALE aircraft, only those performed by Colas et al. incorporate the flexible aircraft design environment recommendations provided by Cesnik et al. Their approach was to combine a number of existing analysis tools into a gradient-free optimization framework. The advantage of gradient-free optimization is that it places little or no restrictions on the analysis tools which may be used within the optimization. The disadvantage of this approach, however, is that it is computationally expensive compared to gradient-based optimization and geometric programming, especially for large numbers of design variables. Geometric programming, on the other hand, while fast, requires that the objective and constraints are defined using monomials and posynomials. While this restriction allows these problems to be solved incredibly fast, it also makes incorporating higher fidelity models into the optimization framework extremely challenging. The purpose of this paper is therefore to present an SR-HALE aircraft optimization framework which accounts for the peculiarities of structurally flexible aircraft while remaining suitable for use with gradient-based optimization. Using this framework, we also perform several parameter sweeps to determine SR-HALE design sensitivities to altitude, latitude, battery specific energy, solar panel system efficiency, avionics and payload power, and minimum design velocity.

II. Methods

To demonstrate how gradient-based optimization can be used to optimize SR-HALE aircraft, we use a swept flying wing configuration, similar to Facebook's Aquila, as the baseline configuration. Details about the assumptions and models used in this study are given in the following sections.

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*PhD Candidate, Mechanical Engineering. Student AIAA.

[†]Assistant Professor, Mechanical Engineering. Senior AIAA.

Mission Requirements

The mission of the SR-HALE aircraft in this study is to sustain high altitude flight year-round carrying a communication payload. We design the aircraft to fly 18 km above sea level in order to avoid commercial and military air traffic. Additional motivation for this design altitude is the significant reduction of average wind speeds which occur around 18 km compared to lower altitudes, which reduce station keeping power requirements [18]. The payload assumptions we use in this paper are the same as used by Colas et al., who assumed a 10 kg payload with a continuous 150 W power draw [9]. We model the payload mass as a point mass placed at the midpoint of the root section chord. To avoid unnecessary design complexity, we include all payload wiring losses and efficiency terms in the 150 W power requirement.

The optimized aircraft is designed to fly year-round at latitudes up to 35°. SR-HALE aircraft able to fly up to 35° latitude can service more than 74 % of the world's population [19]. Sufficient energy must be captured by solar panels to sustain flight year-round at any location within this latitude band. This requirement is met if the aircraft can collect enough energy to offset its power requirements during any 24 hour period during the year ($E_{in} \geq E_{out}$). The most extreme 24 hour scenario within the designated latitude band is winter solstice at 35° latitude, so we assume winter solstice operating conditions in this work.

Aircraft Configuration

We parameterize the baseline swept flying wing aircraft using six spanwise stations as shown in Fig. 1 with motors and propellers located at stations 3 and 4. Geometric design variables include span (b), sweep (Λ), chord length (c), airfoil thickness ratio (t/c), twist (θ), winglet length ($l_{winglet}$), and winglet dihedral ($\psi_{winglet}$). We measure sweep from the leading edge of each airfoil section, chord lengths in the streamwise direction, and twist normal to the swept wing. Chord length, twist, and airfoil thickness ratio are specified at each of the six stations and are linearly interpolated between stations. To accommodate the aircraft payload, we constrain the root chord to be greater than that of the baseline Aquila configuration (2 m).

Figure 1: The parameterized baseline configuration in this study, defined at several spanwise stations, with properties linearly interpolated between stations. If a span length of 42 m is assumed, the aircraft configuration closely resembles Facebook's Aquila.

While the aircraft is designed to be trimmed without flap deflections during design operation ($\sum F = 0.0$ and $\sum M = 0.0$), flaps are added to the aircraft between stations 4 and 5 to allow for off-design trimmed-flight operating point evaluation and active control during gust simulations. Assumed flap derivatives are presented in Table 1. These flap derivatives were chosen based on flap deflection calculations using XFOIL at a Reynolds number of 4×10^5 with the E216 airfoil.

Aircraft Structure

We assume the aircraft has a box beam and stiffened skin configuration, as shown in Fig. 2. High modulus carbon fiber reinforced polymer

Table 1: Assumed aileron flap derivatives

$dc_l/d\delta_f$	0.08/°
$dc_d/d\delta_f$	0.001/°
$dc_m/d\delta_f$	-0.01/°

(CFRP) is used as the primary SR-HALE aircraft structural material due to the stability-constrained nature of flying wing SR-HALE aircraft. [9, 20] Assumed material properties are given in Tables 2 and 3. Both the stiffened skin and the box beam layups feature a sandwich structure in order to increase the bending stiffness of limited numbers of CFRP plies. Balanced and symmetric layups are used to minimize membrane/bending and stretching/shearing coupling, respectively [21]. CFRP fabric is placed on the outside of the aircraft to reduce impact damage and provide torsional rigidity. Unidirectional CFRP is aligned with the axial direction to increase bending stiffness.

Figure 2: Box beam and stiffened skin configuration assumed in this study.

Table 2: Assumed Material Stiffness Properties

Property	CFRP Tape	CFRP Fabric	Foam
E_1 (GPa)	175.0	85.0	0.048
E_2 (GPa)	8.0	85.0	0.048
G_{12} (GPa)	5.0	5.0	0.028
ν_{12}	0.30	0.10	0.3
ρ (kg/m ³)	1600	1600	75.0
t_{ply} (mm)	0.1	0.1	≥ 1

Spar cap widths normalized by chord ($(w/c)_{spar}$) are allowed to vary from station to station and are linearly interpolated between stations. Foam core thicknesses (t_{foam}) and spar cap location ($(x/c)_{spar}$), measured from the leading edge to the first web and normalized by the section chord length, are also exposed to the optimizer as design variables but are not varied spanwise. Foam core thicknesses are constrained to be greater than 1 mm ($t_{foam} > 1$ mm) to prevent optimal designs with unrealistically small foam core thicknesses.

Airfoil Shape and Performance

To allow structural and aerodynamic airfoil thickness trade-offs to be captured, we created a family of airfoils by modifying the thickness over chord ratio (t/c) of a baseline airfoil in XFOIL [22]. Among potential baseline airfoils, we chose to limit our selection to low Reynolds number airfoils due to the low optimal flight speeds of SR-HALE aircraft. Among these airfoils we chose to use the E216 airfoil due to its gradual stall characteristics and large drag bucket. Selected airfoils from our final airfoil family are shown in Fig. 3.

Two-dimensional lift, friction drag, profile drag, and moment coefficients are precomputed using XFOIL for a range of thicknesses, Reynolds numbers, and angles of attack. We fit cubic B-splines to the precomputed data and used these splines in the optimization. Local two-dimensional aerodynamics properties are specified with these splines using local station thickness, Reynolds number, and angle of

Table 3: Assumed Material Strength Properties

Property	CFRP Tape	CFRP Fabric
S_1^+ (MPa)	1000.0	350.0
S_1^- (MPa)	850.0	150.0
S_2^+ (MPa)	40.0	350.0
S_2^- (MPa)	200.0	150.0
S_{12} (MPa)	60.0	35.0

Figure 3: Selected airfoils from E216 airfoil family generated using XFOIL.

attack. Sample lift and drag properties, assuming a Reynolds number of 4×10^5 , are shown in Figs. 4a and 4b.

Figure 4: E216 airfoil family lift and drag coefficient variation with thickness and angle of attack at a Reynolds number of 4×10^5 .

Propulsion System

To reduce propulsion system complexity and weight, we chose to use fixed pitch propellers. Assumed propeller torque, thrust, and efficiency curves are presented in Figs. 5a to 5c. We calculated these propeller performance curves using blade element momentum theory in a coupled SR-HALE trajectory and propulsion optimization study [23]. Propeller diameter (D_p) and advance ratio (J) are available to the optimizer as design variables and are the same for each propeller.

Motor performance is estimated using Drela's first order motor model [24]. Motor torque and rotation rate is matched to propeller torque and rotation rate in order to determine combined propulsion system performance. Motor no-load current (I_0) and K_v are used to characterize each motor and are varied by the optimizer. We model the

Figure 5: Assumed propeller non-dimensional properties

mass of each motor and propeller combination as a five kilogram point mass located one meter in front of the aircraft at stations 3 and 4.

Power Requirements

Total power required is given by:

$$P_{\text{req}} = \frac{V_{\infty} D}{\eta_{\text{motor}} \eta_{\text{prop}} \eta_{\text{other}}} + P_{\text{avionics}} + P_{\text{payload}}$$

where V_{∞} is the freestream velocity, D is the total aircraft drag, η_{motor} is the motor efficiency, η_{prop} is the propeller efficiency, η_{other} is the efficiency of all other components of the propulsion system (assumed in this study to be 0.97), P_{avionics} is the power required by the avionics system, and P_{payload} is the power required by the payload. The avionics and payload combined are assumed to require 400 W of continuous power. All avionics and payload efficiency terms are included in these power requirements.

Battery Mass

Currently reported cell specific energies on Airbus's Zephyr have specific energies of 435 Whr/kg^a. After accounting for additional mass due to battery packaging, lower specific energies can be expected. We therefore chose to set battery specific energy equal to 350 Whr/kg. Additionally, since batteries degrade rapidly when allowed to fully discharge, we limit battery depth of discharge to 80 %, effectively limiting battery specific energy to 280 Whr/kg. Batteries are modeled as four point masses located at stations 3 and 4 (the same spanwise locations as the motors and propellers). The normalized chord-wise location of these point masses is fixed at the quarter chord of their respective airfoil section chord lines. Inboard and outboard battery masses (m_{batt}) may vary subject to the restriction that total available energy storage be greater than the energy required for night-long flight ($m_{\text{batt, total}} \geq m_{\text{batt, req}}$).

Energy Capture

Energy capture is provided by solar panels covering the aircraft's upper surface. We assume that solar panels have the same thickness as those used on the Solar Impulse (135 μm ^b) and have the density of silicon (2328 kg/m³). Available solar flux is determined through using

^a<https://aviationweek.com/business-aviation/amprius-developing-more-applications-its-lithium-ion-cells>
^b<https://aroundtheworld.solarimpulse.com/adventure>

the Simple Model of the Atmospheric Radiative Transfer of Sunshine (SMARTS) [25]. The aircraft starts the day facing northward and follows a steady level 3 km radius counterclockwise orbit throughout a 24-hour period. The roll necessary to maintain this orbit is determined by that of a coordinated turn:

$$\phi = \tan^{-1} \left(\frac{V^2}{gR} \right)$$

where V is cruise velocity, g is gravity, and R is the orbit radius. The yaw angle is determined by assuming the aircraft is always oriented tangent to the orbit radius.

To determine the total amount of solar capture on the upper surface of the aircraft, the aircraft is discretized spanwise and chordwise into flat panels with associated roll, pitch, yaw, and area. This discretization is performed using the shape of the aircraft at the design operating point. Panel normal vectors are compared with sun vectors throughout the day to determine the incident solar flux on each panel at each time step. This flux is numerically integrated across the 24 hour period and multiplied by panel area to determine the amount of solar energy captured by each panel. This energy capture is then reduced using a total solar panel system efficiency of 20 %. The Solar Impulse used 23 % efficient solar panels, so the solar panel system efficiency we use in this paper is quite conservative.

Flight Envelope

To ensure conservative structural constraints, we chose to use select transport aircraft design regulations as a basis for the structural sizing of the SR-HALE aircraft considered in this study. To provide a sufficient buffer for trajectory optimization and/or other maneuvering at altitude, we chose to set the maximum cruise equivalent airspeed (EAS) 5 m/s above the design airspeed. Based on transport aircraft requirements, the dive speed is set to be 125 % of the design cruise speed and flutter is constrained at airspeeds up to 110 % of the aircraft dive speed. The resulting flight envelope is shown in Fig. 6. Lettered operating points correspond to quasi-steady operating point evaluation locations where the aircraft is constrained against material failure ($\sigma \leq \sigma_{\max}$) and local buckling ($\varepsilon \leq \varepsilon_b$). Safety factors of 1.5 are prescribed to both modes of failure. Numbered operating points correspond to flutter evaluation locations, where the aircraft is constrained against failure due to aerostructural instabilities. The aircraft is also constrained against material failure and local buckling at these locations. For all operating points shown in Fig. 6 angle of attack is varied to obtain the specified load factors. For flutter evaluation locations, flap deflections are used to trim the aircraft.

Figure 6: Flight envelope of SR-HALE aircraft in this study. V_C is set to be 5 m/s EAS above the design speed. V_D is set to 1.25 V_C . Lettered operating points are tested for material failure and local buckling. Numbered operating points are tested for aerostructural stability, material failure, and local buckling.

To prevent low-speed flutter, aircraft stability is also evaluated at half of the aircraft design speed. However, since trimmed flight may be impossible at this operating point, aircraft angle of attack and flap

deflections are set to zero. Material failure, local buckling, and stability constraints are applied at this operating point.

Non-Uniform Gust Encounters

Flexible flying wing aircraft react more strongly to non-uniform gusts than uniform gusts [11, 12], so we chose to consider only non-uniform gusts. Using AC 25.341-1, gust magnitudes and profiles are given by:

$$U(x) = \frac{U_{ds}}{2} \left(1 - \cos \frac{\pi x}{H} \right)$$

where U_{ds} is the design gust velocity and H is the gust gradient distance. Gust profiles is then modified on a spanwise basis by the following function:

$$U(x, y) = U(x) \cos \left(\frac{2\pi y}{b} \right)$$

where b is the aircraft span. Three time domain simulations are performed and checked for failure each optimization iteration using the parameters given in Table 4.

Table 4: Gust Parameters

	V_{ds} (m/s, EAS)	H (meter)
Short Gust	3.712	9.1
Medium Gust	5.053	57.9
Long Gust	5.595	106.7

During gust simulations, PID control based on the aircraft pitch rate is used to manipulate the aircraft's flaps to restore steady level flight. These gains were tuned manually and fixed for the duration of the optimization. PID gains are presented in Table 5. Proportional control is also used to reduce aircraft throttle in order to maintain airspeed during gust encounters.

Table 5: Pitch rate PID gains for gust simulations

k_p	-0.4° s/rad
k_i	-0.02° /rad
k_d	$-0.1^\circ \text{ s}^2 \text{ /rad}$

Flexible Aircraft Aeroelastics

ASWING is used to model aircraft design airspeed, dive speed, and limit load flight conditions as well as for gust encounters. The main approach of ASWING is to model aircraft as interconnected nonlinear beams with lifting surface properties calculated using extended lifting line theory. [26, 27] Section structural properties are determined using PreComp [28], whereas 2D aerodynamic parameters are calculated using XFOIL as previously described. Rather than using a fixed lift curve slope for each airfoil section, we chose to calculate lift curve slopes and airfoil zero lift angle of attacks using local airfoil angle of attack, lift coefficient, and lift curve slope. Section normal twist was used as the local angle of attack, and lift coefficient and lift curve slope were obtained from the precomputed E216 family splines discussed previously. We also created a wrapper for ASWING which would allow direct function input/output (rather than file writing and reading) so the program would be more suitable for gradient-based optimization^c.

Material Failure

Maximum axial strain and torsional shear stress are calculated as performed by ASWING, but are extended from the original ASWING framework to allow evaluation at multiple locations for each airfoil section. Classical laminate theory is then used to determine ply stresses and strains based on laminate axial strain and torsional shear stress. First-ply failure is assumed, with failure predicted by the Tsai-Wu failure theory [29]. Prior to applying this failure theory, material

^cThe code for this wrapper can be found at <https://github.com/buyflowlab/Aswing.jl>

scatter and barely visible impact damage (BVID) knockdown factors are applied. Material scatter knockdown factors are calculated for CFRP tape and fabric based on A-basis values from the Advanced General Aviation Transport Experiments for TORAY T700GC-12K-31E/#2510 Unidirectional Tape and TORAY T700SC-12K-50C/#2510 Plain Weave Fabric [30, 31]. Material scatter knockdown factors are tabulated in Table 6 for convenience. The impact of barely visible impact damage varies depending on the specific layup used and is typically determined through testing. For this paper, we chose the BVID knockdown factor to be 0.65 which approximately corresponds to a 25 mm impact damage diameter [32].

Table 6: Material Scatter Knockdown Factors

Material	k_1^+	k_1^-	k_2^+	k_2^-	k_{12}
CFRP tape	0.625	0.762	0.803	1.0	0.920
CFRP fabric	0.764	0.776	0.719	0.859	1.0

Local Buckling

To calculate buckling loads, the upper and lower surfaces of airfoil sections are broken up into several flat panels using the section's webs as shown in Fig. 7. These buckling panels, along with the airfoil section's

Figure 7: Airfoil section divided up into several flat buckling panels by the section's webs

webs are approximated by long, simply supported plates and tested for buckling.

Assuming long simply supported plates, the local buckling load is given according to Johnson [33] as:

$$N_{cr} = 2 \left(\frac{\pi}{b} \right)^2 \sqrt{D_{11}D_{22} + D_{12} + 2D_{66}}$$

where b represents the width of the plate, here taken as the chordwise length from the web location to the front (or back) of the airfoil section, and the D stiffness matrix is computed using classical laminate theory. The buckling strain is then computed as:

$$\epsilon_b = \frac{N_{cr}}{A_{11} - \frac{A_{12}^2}{A_{22}}}$$

where the A stiffness matrix is found using classical laminate theory. This buckling strain is then used as an upper bound for axial strains computed in ASWING.

Aircraft Stall

Critical section theory is used to constrain aircraft stall ($c_l \leq c_{l, \text{stall}}$) during design operation. According to critical section theory, a wing stalls when any airfoil section lift coefficient exceeds the corresponding section's maximum lift coefficient. To provide a buffer against stall during design operation, maximum lift coefficients are reduced by 90 % on the inboard portion of the wing and up to 80 % on the outboard portions of the wing as defined by the following factor:

$$G(y/b) = \begin{cases} 0.9 & 0 \leq |y/b| \leq 0.25 \\ 0.9 - 1.6(|y/b| - 0.25)^2 & 0.25 < |y/b| \leq 0.5 \end{cases}$$

Maximum lift coefficients on outboard portions of the wing are reduced further than inboard lift coefficients in order to prevent tip stall.

Longitudinal Stability

A static margin of 10% is enforced at the design airspeed (static margin $\geq 10\%$). This static margin constraint is enforced through an equivalent constraint on dC_M/dC_L about the aircraft's center of gravity. Traditional stability derivatives are based on rigid aircraft assumptions, however, static margin may still be calculated either by assuming quasi-steady aeroelastic loads or by assuming a rigid aircraft shape (after taking into account deformations due to design operating conditions). We constrain both quasi-steady and rigid aircraft static margin values at each design iteration.

General Stability/Flutter

ASWING allows for the determination of coupled flight and aerostructural stability through eigenmode analysis about quasi-steady states. We use this feature of the program in the optimization to assess the stability of the SR-HALE aircraft for airspeeds up to 110 % of the aircraft dive speed. A stable configuration is found when all eigenvalues are in the left half of the complex plane. Equivalently, a stable configuration exists when the maximum real part of all eigenvalues is negative. We use the latter constraint formulation, in which the maximum real part of all eigenvalues is constrained to be negative. However, since the maximum function is not smooth, and thus not appropriate for use with gradient-based optimization, we approximate it using the Kreisselmeier-Steinhauser (KS) constraint aggregation function [34]. This function has the following form:

$$KS(g(x)) = \frac{1}{\rho} \ln \left[\sum_{j=1}^{n_g} e^{\rho g_j(x)} \right]$$

where $g(x)$ defines a vector of constraint functions, n_g is the length of that vector, and ρ is a parameter that controls the "hardness" of the function. As ρ approaches ∞ , the "hardness" of this function increases until $\rho = \infty$ when the actual maximum is returned. An additional feature of this function is that it overestimates the maximum ($KS(g_j(x)) \geq \max(g_j(x))$) which makes it particularly well suited for the aggregation of constraints since it guarantees a conservative constraint formulation.

To decrease computational expense, only a limited subset of eigenvalues corresponding to the least stable modes for each considered flight condition are computed using the Arnoldi iteration. This approach still allows for a smooth constraint formulation when used with the KS constraint aggregation function, because as the stability of an eigenvalue increases relative to other eigenvalues its influence on the stability constraint decreases rapidly until it can be safely ignored. As a byproduct of this feature of the KS function, regions of the complex plane may be ignored through artificially increasing the stability of these modes. We use this technique in this paper to reduce the impact of eigenvalues with unrealistically high frequency components ($\text{Im}(\lambda) > 20 \text{ Hz}$) and neglect low magnitude eigenvalues ($|\lambda| < 1.0$), which likely correspond to modes which can controlled using active control methods.

Additional Mass and Drag

A mass and drag markup at each design iteration is applied to account for the additional mass and drag of components which are not explicitly modeled. Components which add to aircraft mass which are not explicitly modeled include but are not limited to: avionics, maximum power point tracker, and landing gear mass. Additional drag could result from surface roughness, landing gear drag, and/or additional protuberance drag. We assume that these miscellaneous factors account for 15 % of the aircraft's mass and 10 % of the aircraft's drag. Additional aircraft mass is placed at the undeformed aircraft's center of gravity.

Optimization Problem

Fully assembled, the optimization problem can be represented as:

$$\begin{aligned}
 &\text{minimize:} && m_{\text{total}} \\
 &\text{with respect to:} && b, \Lambda, c, t/c, \theta, l_{\text{winglet}}, \psi_{\text{winglet}}, l_{\text{foam}}, (x/c)_{\text{spar}}, \\
 &&& (w/c)_{\text{spar}}, m_{\text{batt}}, V_{\infty}, D_p, J, K_v, I_0 \\
 &\text{subject to:} && E_{\text{in}} \geq E_{\text{out}} \\
 &&& m_{\text{batt},\text{total}} \geq m_{\text{batt},\text{req}} \\
 &&& c_{\text{root}} \geq 2 \text{ m} \\
 &&& \sigma \leq \sigma_{\text{max}} \\
 &&& \varepsilon \leq \varepsilon_b \\
 &&& c_l \leq c_{l,\text{stall}} \\
 &&& \text{static margin} \geq 10\% \\
 &&& KS(\text{real}(\lambda)) \leq 0.0 \\
 &&& \sum F = 0.0 \\
 &&& \sum M = 0.0
 \end{aligned}$$

Total aircraft mass m_{total} is minimized since aircraft mass is roughly proportional to total aircraft cost. In total, the optimization problem involves 39 design variables. Gradients were obtained for gradient-based optimization using the finite difference method.

III. Results

Optimal Design

The optimized aircraft is shown in Fig. 8. Final design variables are presented in Tables 7 to 9. The aspect ratio of the final aircraft is 27.8, which is similar to the 28.4 aspect ratio of the flying wing aircraft optimized by Colas et al. The mass distribution of the optimized aircraft, as shown in Fig. 9 is also similar to the results found by Colas et al., with a large percentage of the 508.8 kg aircraft mass being used for energy storage.

Figure 8: Top and rear view of the optimal design during design operation. While the wing is designed without dihedral, a small amount of dihedral is present during cruise operating flight due to the aeroelastic deformation of the aircraft.

Table 7: Final Design Variables

Parameter	Value
Span (m)	52.49
Leading Edge Sweep (°)	19.42
Winglet Length (m)	2.07
Winglet Dihedral (°)	58.4
Skin Foam Thickness (mm)	1
Spar Foam Thickness (mm)	1
Web Foam Thickness (mm)	1
Distance to First Web (x/c)	0.040
Design Velocity (m/s)	26.5
Battery Mass at Station 3 (kg)	59.06
Battery Mass at Station 4 (kg)	53.27
Propeller Diameter (m)	3.10
Advance Ratio (V/nD)	0.917
Motor Kv (rpm/V)	1.32
Motor No-Load Current (A)	0.1

Table 8: Final Spanwise Varying Design Variable (Stations 1-3)

Parameter	Station 1	Station 2	Station 3
Chord (m)	3.101	2.781	2.545
Airfoil Thickness Ratio (t/c)	0.171	0.180	0.180
Twist (°)	8.71	8.48	8.07
Spar Width (w/c)	0.451	0.422	0.411

Table 9: Final Spanwise Varying Design Variable (Stations 4-6)

Parameter	Station 4	Station 5	Station 6
Chord (m)	1.978	0.576	0.238
Airfoil Thickness Ratio (t/c)	0.133	0.118	0.105
Twist (°)	5.71	3.35	0.88
Spar Width (w/c)	0.366	0.087	0.010

The design relies on large airfoil thickness ratios and spar widths to provide sufficient structural strength to prevent material failure, buckling, and flutter. These spanwise varying parameters are particularly large towards the center of the aircraft where greater aerodynamic loads occur. While this preference for greater structural strength on the in-board portions of the wing is not unexpected, there does appear to be a preference for increased structural efficiency over aerodynamic efficiency. This is evidenced by the presence of airfoil thickness-to-chord ratios which are up to 73 % thicker than the original thickness of the E216 airfoil (10.4 %), despite the greater lift-to-drag ratios of thinner airfoil sections.

Active inequality constraints at the end of the optimization consisted of energy balance, battery mass, material failure, local buckling, static margin, and general stability constraints. Active bound constraints at the end of the optimization consisted of foam thicknesses, wingtip spar width, and motor no-load current, which were all constrained by their respective lower bounds. Since material failure, local buckling, and general stability constraints were all active at the end of the optimization, no structural constraint appears to dominate the optimization, thus highlighting the importance of including all the structural constraints modeled in this optimization.

Deflection and aeroelastic twist were small for all considered operating conditions as shown in Figs. 10 and 11. This is in stark contrast to solar aircraft prototypes like the HELIOS which exhibited large amounts of deflection, even during design operating conditions. The most plausible explanation for the lack of significant aeroelastic deflection and twist is the stringent transport aircraft structural constraints applied to the design. Relaxed design requirements would likely lead to an optimal design with greater structural flexibility.

To test whether the optimizer would react favorably to larger/smaller ply thicknesses, we introduced tape and fabric CFRP ply thicknesses as design variables to the optimizer and re-optimized the design. Tape ply thicknesses changed very little (final tape ply thicknesses were 0.099 mm). Fabric ply thicknesses, however, decreased

Figure 9: Mass distribution of optimal design

Figure 10: Wing deflection of optimized design

Figure 11: Aeroelastic twist of optimized design

dramatically from 0.1 mm to 0.008 mm. Total aircraft mass also dramatically decreased by 56.8 % from 508.8 kg to 219.7 kg. Curiously, as shown in Tables 10 and 11, airfoil thicknesses and spar widths mostly increased, which shows that using large airfoil thickness-to-chord ratios and spar widths to provide sufficient structural strength results in greater overall SR-HALE aircraft design performance than increasing ply thicknesses. While obtaining fabric ply thicknesses of this magnitude for actual design is improbable, these results suggest that thinner CFRP plies are in general more appropriate for SR-HALE aircraft development than thicker CFRP plies. Increased SR-HALE aircraft performance therefore provides significant motivation for the development and standardization of thin ply composites.

Table 10: Ply Thickness Study Final Design Variables (Stations 1-3)

Parameter	Station 1	Station 2	Station 3
Chord (m)	2.00	1.84	1.84
Airfoil Thickness Ratio (t/c)	0.196	0.212	0.207
Twist ($^\circ$)	8.18	8.79	6.77
Spar Width (w/c)	0.469	0.437	0.403

Parameter Sweeps

In order to calculate the sensitivity of the design to technology and mission requirement changes, we performed several parameter sweeps in which the optimal design was re-optimized while varying design altitude, design latitude limit, battery specific energy, solar panel system efficiency, avionics and payload power requirements, or minimum design velocity. The results of these parameter sweeps are presented in Figs. 12 to 17.

As seen in Fig. 12, decreases in design altitude allow lighter aircraft to be built. Increases in air density allow the aircraft to fly slower, reducing power requirements. Reduced power requirements then allow

Table 11: Ply Thickness Study Final Design Variables (Stations 4-6)

Parameter	Station 4	Station 5	Station 6
Chord (m)	1.15	0.63	0.26
Airfoil Thickness Ratio (t/c)	0.144	0.120	0.100
Twist ($^\circ$)	6.54	3.14	0.45
Spar Width (w/c)	0.647	0.098	0.025

the aircraft's size to decrease because less solar panel area and battery mass are necessary for sustained flight. Airspace restrictions, may however, necessitate flying at higher altitudes, despite the benefits of flying at lower altitudes. Additionally, station keeping constraints may require that aircraft fly faster than would otherwise be optimal, which would reduce and/or eliminate the benefit of flying at lower altitudes.

Figure 12: Sensitivity of optimized aircraft mass to design altitude

Figure 13 shows that the optimized aircraft is very sensitive to design latitude limit choices at high latitudes, but less sensitive at latitudes below 25°. The reduction in design sensitivity at latitudes below 25° is caused by inactive energy capture constraints, in other words, the aircraft surface area and shape is no longer constrained by solar capture constraints at latitudes below 25°. The absence of active solar capture constraints, does not render the design insensitive to latitude at lower latitudes, however, because latitude decreases energy storage requirements because of shorter winter solstice night lengths at lower latitudes.

Figure 13: Sensitivity of optimized aircraft mass to design latitude limit

The design's sensitivity to battery specific energy is very high when battery specific energies are set to 350 Whr/kg, but becomes less sensitive at battery specific energies above 400 Whr/kg, as seen in Fig. 14. Despite the decrease in size of the aircraft due to battery specific energy increases, the aerodynamic performance of the aircraft remained relatively unchanged, since aircraft aspect ratio, lift coefficients, and drag coefficients remained within 5 %, 2.5 %, and 1.5 % of their original values for all considered battery specific energies.

Figure 14: Sensitivity of optimized aircraft mass to battery specific energy

Figure 15 shows that the design is very sensitive to solar panel system efficiencies below 20 %, but not sensitive at solar panel system efficiencies above 20 %. Above 22.5 % solar panel system efficiencies, the design is not at all sensitive to this parameter and the optimal design does not change. The energy storage and structural constraints present in the optimal design with efficiencies of 22.5 % prevent the aircraft's surface area from decreasing further as efficiencies are increased. It may therefore be possible to reduce aircraft costs by using less solar panels and/or lower efficiency solar panels. Changes in the other mission requirements may, however, cause the design to be sensitive at higher efficiencies.

Figure 15: Sensitivity of optimized aircraft mass to total solar panel efficiency

We found the design to be very sensitive to avionics and payload power requirements for every set of power requirements we considered, as shown in Fig. 16. As power requirements increase, both additional battery mass and surface area are necessary to satisfy energy storage and capture constraints. Small changes in payload power requirements, may therefore cause large changes in the mass of optimal SR-HALE aircraft designs. So while the results of this study suggest that highly efficient solar panels may not be necessary, reducing payload power requirements seems to be an efficient way of enabling optimal designs with lower mass.

The last parameter sweep we performed was an investigation into the impact of minimum design velocities on optimal aircraft mass. The purpose of a minimum design velocity constraint would be to ensure that aircraft can successfully station keep at the design altitude in the presence of wind. As seen in Fig. 17, we found the design to be extremely sensitive to increases in design velocity. Specifically a 3.5 m/s increase in true airspeed requirements at altitude from 26.5 m/s to 29.0 m/s resulted in an aircraft design mass increase of 210.7 kg. Since high penalties exist for increasing the minimum design velocity of SR-HALE aircraft, station keeping constraints may be easier to satisfy by

Figure 16: Sensitivity of optimized aircraft mass to combined avionics and payload power requirements

changing the design altitude rather than by increasing minimum design velocity.

Figure 17: Sensitivity of optimized aircraft mass to minimum design velocity

IV. Conclusions

In this paper we presented an SR-HALE aircraft optimization framework which models and constrains the nonlinear aeroelasticity of flexible aircraft while remaining suitable for use with gradient-based optimization. We incorporated nonlinear aeroelasticity into the optimization framework by using the existing flexible aircraft analysis tool ASWING, and constructed energy capture, energy storage, material failure, local buckling, stall, longitudinal stability, and coupled flight and aeroelastic mode stability constraints that are suitable for use with gradient-based optimization. We then used this framework to minimize the mass of a flying wing SR-HALE aircraft defined by six spanwise stations subject to structural constraints based on transport category aircraft design requirements. We also ensured that the final design was conservative through mass and drag markups.

The optimized aircraft's aspect ratio and surface area were 27.8 m² and 99.1 m² respectively and the weight was 508.8 kg. Despite the high aspect ratio design, the aircraft exhibited very little aeroelastic deflection and twist, likely due to the conservative nature of the imposed transport aircraft category structural constraints. Sufficient structural strength was provided to the design through varying airfoil thicknesses, spar widths, and chord lengths. Inboard airfoil section thicknesses were particularly large, with optimal airfoil thickness over chord ratios as high as 0.180. When CFRP ply thicknesses were included as design variables, fabric ply thicknesses dramatically decreased and even larger airfoil thickness over chord ratios were found. Thick airfoils therefore appear to provide greater structural efficiency than increased CFRP ply thicknesses.

Finally, we performed several parameter sweeps to determine the sensitivity of optimized SR-HALE aircraft designs to various mission requirements and technological capabilities. Investigated parameters include design altitude, design latitude, battery specific energy, solar panel system efficiency, combined avionics and payload power requirements, and minimum design velocities. We found the sensitivity of the design to all of these parameters to be high, except for in the case of solar panel system efficiency increases. In fact, solar panel efficiency increases above 22.5 % were found to have no impact on the optimal design.

Future work includes investigating the applicability of the design features and trends found within this paper to other configurations and modeling the impact of active aeroelastic control on the design of SR-HALE aircraft. Work also needs to be done to investigate how optimal SR-HALE aircraft designs change when coupled with trajectory optimizations. Future SR-HALE aircraft optimizations would also benefit from improved methods of modeling and constraining coupled flight and aerostructural modes appropriate for gradient-based optimization. A major limitation of the current approach is that it applies an equivalent constraint to all flight and aeroelastic modes. In practical design, it may be advisable to apply different safety factors to different modes and/or allow certain aeroelastic modes to be controlled through active aeroelastic control.

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