

**A SYSTEMATIC APPROACH TO
DESIGNING LOW ALTITUDE LONG
ENDURANCE SOLAR-POWERED
UNMANNED AERIAL VEHICLES**

**UNE APPROCHE SYSTÉMATIQUE DE LA
CONCEPTION DES VÉHICULES AÉRIENS
SANS PILOTE SOLAIRES À BASSE
ALTITUDE ET LONG TEMPS DE VOL**

A Thesis Submitted to the Division of Graduate Studies
of the Royal Military College of Canada
by

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In Partial Fulfillment of the Requirements for the Degree of
Master of Applied Science in Aeronautical Engineering

July, 2026

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I'll burn that bridge when I get to it.

Acknowledgements

First and foremost I give thanks to God for the opportunity to pursue this degree and all the blessings that have followed by returning to Kingston.

I would like to express my deepest gratitude to my supervisor, Ruben Perez, for fighting tooth and nail on my behalf to get me back to RMC. Your passion and expertise has made my time at the College unforgettable and challenging in all the right ways.

I am greatly appreciative of all the other members of the Advanced Aircraft Design Lab. Your knowledge, ideas, and shenanigans are more valuable than I could ever express.

Outside of the academic world, I would also like to extend a huge thank you to my husband, family, and friends. Your love, support, prayers, and laughter have contributed immensely in maintaining my peace of mind through this work and all efforts preceding it.

Abstract

Mydland, Hayden. M.A.Sc. Royal Military College of Canada, July 2026. *A Systematic Approach to Designing Low Altitude Long Endurance Solar-Powered Unmanned Aerial Vehicles*. Supervised by Ruben E. Perez, BEng, MASc, PhD, PEng, Professor.

Long endurance solar powered flight missions are highly sensitive to the available solar power and onboard energy capacity. The location and time of year are critical factors which determine whether a specific design will be able to fly continuously for more than 24 hours. The onboard energy capacity must be sufficient to support a continuous power draw throughout the night until sufficient solar power is available in the morning to recharge the batteries. The extended endurance has practical applications including intelligence, surveillance, reconnaissance, signals relay, and environmental monitoring. Current endurance record holders and other successful solar powered Unmanned Aerial Vehicle (UAV) designs feature glider-like wings and sleek fuselage designs. There is minimal justification that a glider is the most suitable for low altitude solar UAVs. To determine the most promising UAV configuration for long endurance solar flight, this work includes a detailed parametric comparison analyzing different combinations of the vehicle configuration, the battery capacity, the number of solar cells, and the wing planform. Results show that configurations with a T-tail empennage are the most promising and that the intended payload has a significant influence on the other optimal design traits. Of the configurations with the T-tail, it is inconclusive as to which wing configuration is most promising: a clean wing or one with nacelles intersecting the span. Both wing configurations are carried forward through the UAV optimization, where each wing configuration with a T-tail is optimized for two different payload cases. From the optimization results, a clean wing and T-tail solar UAV has the greatest potential for long endurance missions for year-round operations for a sample case based in Kingston, ON. A final geometry is proposed for production featuring a 10 ft wingspan and maximum takeoff weight of 9.3 lbf. Lastly, the properties of the proposed solar UAV are connected to a trajectory and control optimization tool to determine suitable and practical flight strategies for maximizing energy margins of this solar UAV.

Keywords: aircraft optimization, conceptual design, UAV, applied aerodynamics, solar powered.

Résumé

Mydland, Hayden. M.A.Sc. Collège militaire royal du Canada, juillet 2026. *Une approche systématique de la conception des véhicules aériens sans pilote solaires à basse altitude et long temps de vol.* Thèse dirigée par Ruben E. Perez, BEng, MAsC, PhD, PEng, Professeur.

Les missions de temps de vol plus longues alimentées par l'énergie solaire sont très sensibles à l'énergie solaire disponible et à la capacité énergétique embarquée. L'endroit géographique et la période de l'année sont facteurs cruciaux qui déterminent si un design peut voler sans pause pour plus de 24 heures. La capacité énergétique embarquée doit être suffisante pour répondre à la consommation d'électricité tout au long de la nuit, jusqu'à ce que l'énergie solaire soit suffisante le lendemain matin pour recharger les batteries. Le temps de vol plus long a des applications pratiques comme le renseignement, la surveillance, la reconnaissance, le relais de signaux, et la surveillance environnementale. Les détenteurs courants des records de temps de vol et d'autres modèles de véhicules aériens sans pilote (ASP) à l'énergie solaire à réussite sont dotés d'ailes planeur et d'un fuselage profilé. Il n'y a que peu de justification qu'une aile planeur est le plus approprié pour les ASP solaires volant à basse altitude. Pour déterminer la configuration de l'ASP plus appropriée pour les missions de temps de vol longue, cette thèse inclue une comparaison paramétrique avec les combinaisons différentes du type de configuration, de la capacité des batteries, du nombre de cellules solaires, et de la forme de l'aile. Les résultats démontrent que les configurations avec un empennage en T sont les plus appropriées et que les charges utiles ont une influence considérable sur les autres caractéristiques de design. Parmi les configurations à empennage en T, il n'est pas possible de conclure quelle configuration d'aile est la plus appropriée : une aile sans nacelles ou une aile avec des nacelles qui coupent l'envergure. Les deux configurations de l'aile sont utilisées dans l'optimisation de l'ASP avec deux cas des charges utiles. L'optimisation conclue qu'une configuration sans nacelles avec un empennage en T présent le plus grand potentiel pour les missions de temps de vol long dans le cadre d'opérations tout au long de l'année pour un cas d'échantillon basé à Kingston, en Ontario. Une configuration finale est proposée pour la production avec un envergure de 10 ft et un poids maximal au décollage de 9.3 lb_f. Enfin, les caractéristiques du ASP proposé sont associées à un outil d'optimisation de trajectoire et de commande afin de déterminer des stratégies de vol appropriées et pratiques afin de maximiser les marges énergétiques de cet ASP solaire.

Mots-clés: optimisation des aéronefs, conception préliminaire, véhicule aérien sans pilote, aérodynamiques appliquées, à l'énergie solaire

Contents

Acknowledgements	iii
Abstract	iv
Résumé	v
List of Tables	x
List of Figures	xii
Nomenclature	xxi
1 Introduction	1
1.1 Motivation for the Research	1
1.2 Research Objectives	2
1.3 Thesis Layout	3
1.4 Contributions	3
2 Literature Review	4
2.1 History of Solar Flight	4
2.2 Challenges of solar-powered Flight	20
2.2.1 LALE Solar Aircraft	21
2.2.2 HALE Solar Aircraft	22
2.3 Sizing and Conceptual Design	23
2.4 Optimizing Solar UAVs	27
2.4.1 Path Planning and Trajectory Optimization	27
NEAT for Trajectory Planning	30
2.4.2 Design Optimization	32
2.5 Solar UAV Design Considerations	33
2.5.1 Current Technologies	35
2.6 Knowledge Gap	36
3 Tools and Methods	37
3.1 UAV Geometry Configuration	37
3.2 Weight Prediction Model	40
3.2.1 Propulsion Component Sizing Methodology	40
3.2.2 Propulsion Component Sizing Correlations	41
BLDC Motors	41
ESCs	44

Propellers	47
3.2.3 Lifting Surface Weight	48
3.3 Payloads	51
3.4 Aerodynamics	52
3.5 Point Mass Model	53
3.6 Solar Model	56
3.7 Energy Model	60
3.7.1 Battery Packs	60
3.7.2 Solar Cells	61
3.7.3 Propulsion	62
Method 1	62
Method 2	62
3.7.4 State of Charge	64
4 UAV Configuration Analysis and Selection	65
4.1 Objective	65
4.2 Design Variables	67
4.2.1 Configurations	67
4.2.2 Battery Packs	68
4.2.3 Solar Cells	68
Orientation	68
4.2.4 Payload	70
4.2.5 Time and Location	70
4.3 UAV Definition	70
4.3.1 Wing Geometry	71
4.3.2 Empennage	72
Configuration Compatibility	72
Sizing	72
4.4 Flight Assumptions	74
4.5 Results	76
4.5.1 Effects of Time and Location	83
4.6 Parametric Analysis Summary	88
5 Solar UAV Optimization	90
5.1 Optimization Problem	90
5.2 Flight Assumptions	98
5.3 Initial Results	99
5.3.1 Clean Configuration without Payload	99
5.3.2 Clean Configuration with Payload	104
Comparison of Clean Configuration Solutions	108
5.3.3 Nacelles without Payload	110
5.3.4 Nacelles with Payload	115
Comparison of Nacelle Configuration Solutions	119
5.4 Initial Optimization Summary	122
5.5 Refined Results	122
5.5.1 Clean Configuration without Payload	123
5.5.2 Clean Configuration with Payload	127
5.5.3 Nacelles without Payload	132

5.5.4	Nacelles with Payload	137
5.6	Final Optimization Summary	142
5.7	Proposed Geometry	143
6	NEAT Trajectory Planning	148
6.1	Fitness Function	148
6.1.1	Simulation	149
6.2	NEAT Trajectories	150
6.2.1	Constant Radius Circular Trajectory	150
6.2.2	Variable Radius Circular Trajectory	154
6.2.3	Constant Heading Trajectory	160
6.3	Summary of Trajectories	163
7	Conclusions and Recommendations	165
7.1	Conclusions	165
7.2	Recommendations for Future Work	166
	Bibliography	168
	Appendices	177
A	Solar Powered Aircraft	178
B	Additional Size and Weight Correlation Statistics	185
B.1	BLDC Motor Volume	185
B.2	Propeller Correlations	186
C	Weight Method Validation	193
D	Aerodynamics Method Validation	195
E	Parametric Analysis Solar Cell Orientation	196
F	Extended Parametric Analysis Results	198
G	Extended Solar UAV Optimization Results	223
G.1	Initial Results	223
G.1.1	Clean Configuration with Payload	223
G.1.2	Nacelles without Payload	223
G.1.3	Nacelles with Payload	226
G.2	Refined Results	229
G.2.1	Clean Configuration without Payload	229
G.2.2	Clean Configuration with Payload	231
G.2.3	Nacelles without Payload	233
G.2.4	Nacelles with Payload	235
G.2.5	Proposed Geometry	237
H	Solar UAV Optimization with Relaxed Constraints	238
H.1	Clean Configuration without Payload	238
H.2	Clean Configuration with Payload	244

H.3	Nacelles without Payload	248
H.4	Nacelles with Payload	254
H.5	Relaxed Constraint Optimization Summary	258
I	NEAT as Path Planning Validation	260

List of Tables

2.1	Zephyr record-breaking flight characteristics.	12
3.1	Statistical validity of BLDC motor correlations.	44
3.2	Statistical validity of ESC correlations.	46
3.3	Coefficients and statistical analysis for regressions in the form of Equation 3.12.	47
3.4	Payload characteristics.	52
3.5	Payload combinations for analysis.	52
3.6	MPPT electrical properties.	61
3.7	Maxeon C60 Solar Cell electrical properties [110].	61
4.1	Assumed empennage volume coefficients, taper ratios, and aspect ratios [47].	73
5.1	Variables for clean configuration optimization, where $P(i) \in [a_i, b_i]$	92
5.2	Constraints for clean configuration optimization.	93
5.4	Constraints for nacelle configuration optimization.	94
5.3	Variables for nacelle configuration optimization, where $P(i) \in [a_i, b_i]$	96
5.5	Wing airfoil ID options.	97
5.6	Propeller ID options.	97
5.7	BLDC motor ID options.	98
5.8	Clean configuration vehicle properties.	110
5.9	Nacelle configuration vehicle properties.	120
5.10	Updated constraints for the clean configuration optimization.	123
5.11	Updated constraints for nacelle configuration optimization.	123
5.12	Clean configuration without payload vehicle properties.	127
5.13	Clean configuration with payload vehicle properties.	131
5.14	Nacelle configuration without payload vehicle properties.	136
5.15	Nacelle configuration with payload vehicle properties.	141
5.16	Properties of the proposed solar-powered UAV.	145
5.17	Weights and performance of the proposed solar-powered UAV.	145
6.1	Initial conditions for each simulated flight.	149
6.2	State, control, and vehicle simulation limits.	150
6.3	Constant radius circular trajectory network weights and biases.	151
6.4	Variable radius circular trajectory network weights and biases.	155
6.5	Constant heading trajectory weights and biases.	161
6.6	Network fitness in hours for different months of the year.	163
A.1	Non-comprehensive list of flown fixed-wing solar aircraft to date.	179

C.1	Wing weight validation.	193
C.2	UAV weight validation.	194
E.1	Wing solar cell orientations at $b_{PV1} = 11$ and $b_{PV1} = 12$ for $n_{PV,wing}/2 = 21$, or 42 total.	196
G.1	Variables for the proposed clean configuration geometry.	237
H.1	Updated constraints for clean configuration optimization.	238
H.2	Updated constraints for nacelle configuration optimization.	238
H.3	Clean configuration without payload vehicle properties for different span constraints.	242
H.4	Clean configuration with payload vehicle properties for different span con- straints.	247
H.5	Nacelle configuration without payload vehicle properties for different span constraints.	250
H.6	Nacelle configuration with payload vehicle properties for different span con- straints.	258
I.1	Constants used in equation formulation.	261
I.2	Fitness values for both optimal control techniques.	262

List of Figures

2.1	Sunrise II, 1975 [6].	5
2.2	Solar Riser, 1979 [6].	5
2.3	Gossamer Penguin, 1980 [6].	6
2.4	Solar Challenger, 1981 [6].	6
2.5	Solair I, 1981 [6].	7
2.6	Sunseeker, 1990 [6].	7
2.7	Icaré 2, 1996 [6].	8
2.8	ERAST sponsored solar-powered aircraft [9].	9
2.9	Helios mishap, 2003 [9].	9
2.10	Alan Cocconi with SoLong [10].	10
2.11	SunSailor and design team [11]	10
2.12	Designs from Noth’s framework.	11
2.13	Zephyr [13].	11
2.14	SunLight Eagle, 2009 [20].	12
2.15	Solar Impulse 2 [21].	13
2.16	AtlantikSolar, 2015 [1].	13
2.17	SolarStratos [24].	14
2.18	MARAAL, 2017 [25].	14
2.19	Facebook’s Aquila [26].	15
2.20	Full-scale model by Rosales <i>et al.</i> [29].	15
2.21	Mini-Phantom, 2019 [31].	16
2.22	Solar Flyer, 2020 [32].	16
2.23	CREATEv, 2020 [33].	17
2.24	Sun UAV [35].	17
2.25	MoonFlyer, 2024 [37].	18
2.26	PHASA-35 [39].	18
2.27	Skydweller [41].	19
2.28	Kea Atmos [45].	20
2.29	NEAT encoding scheme [81].	31
3.1	Sample geometries from the UAV configurator.	37
3.2	Fuselage and nacelle geometry, not to scale.	38
3.3	Fuselage cross section before specifying height to width ratio.	39
3.4	Different sized battery packs streamlined with the flow, not to scale.	40
3.5	Visualization of BLDC motor weight prediction.	42
3.6	Visualization of BLDC motor diameter prediction.	43
3.7	Visualization of BLDC motor length prediction.	44
3.8	Statistical validation of ESC weight prediction.	45

3.9	Statistical validation of ESC volume prediction.	46
3.10	Statistical validation of 2-blade APC Propeller prediction.	48
3.11	UAV wing modelled as a cantilever beam.	49
3.12	Aircraft body axis orientation [102].	53
3.13	Angular relationships between earth, body, and velocity axes [102].	54
3.14	Forces acting on the aircraft in steady level flight.	55
3.15	Azimuth and elevation angle convention.	56
3.16	Sample solar data starting at 0700 1 July 2025 in Kingston, ON.	57
3.17	Solar vectors $\vec{n}_{PV}$ and $\vec{n}_s$ and the obliquity angle θ_s	58
3.18	Map of Canada with points of interest.	59
3.19	Solar irradiance throughout the year in Kingston, ON.	59
3.20	Solar irradiance throughout the year in Inuvik, NT.	60
4.1	SOC and power levels for a 48 h flight, showing t_{exc} and t_{cm}	66
4.2	Power levels and SOC throughout sample simulated flight of 28.9 h.	67
4.3	Configurations analyzed.	67
4.4	Possible solar cell locations.	69
4.5	Horizontal stabilizer solar cell orientations.	69
4.6	Sample wing geometries with $b_{PV1} = 12$, $n_{PV,wing}/2 = 27$	72
4.7	Static margin of each configuration, identified by n_{PV}	76
4.8	Effects of $\mathcal{R}$ on total weight, identified by n_{batt}	77
4.9	Effects of $\mathcal{R}$ on total weight for $b_{PV1} = 12$, identified by n_{batt}	77
4.10	Flight time as a function of b_{PV1} , identified by n_{batt} for zero payload.	78
4.11	Charge margin as a function of n_{PV} , identified by n_{batt} for $b_{PV1} = 12$	79
4.12	Flight time as a function of b_{PV1} , identified by n_{batt} with payload option 2.	79
4.13	Flight time as a function of n_{batt} , identified by payload option at $b_{PV1} = 12$	80
4.14	Flight time as a function of n_{batt} , identified by n_{PV} for zero payload at $b_{PV1} = 12$	80
4.15	Flight time as a function of n_{batt} , coloured by $\mathcal{R}$, identified by Configuration for no payload and $b_{PV1} = 12$	81
4.16	Optimal wing configuration based on b_{PV1} and n_{batt}	82
4.17	Flight time as a function of the number of solar cells on the horizontal stabilizer for Configurations 2, 3, 4, and 5, where $b_{PV1} = 12$ and $n_{batt} = 2$, with no payload.	83
4.18	Flight time as a function of time of year, identified by n_{batt} for zero payload in Kingston, ON.	84
4.19	Flight time as a function of time of year, identified by n_{batt} for zero payload in Inuvik, NT.	84
4.20	Flight time as a function of time of year, identified by n_{batt} for payload option 2 in Kingston, ON.	85
4.21	Flight time as a function of time of year, identified by n_{batt} for payload option 2 in Inuvik, NT.	85
4.22	Power levels and SOC for a UAV with $b_{PV1} = 12$, $n_{batt} = 3$, beginning at 1100 UTC, flown in Inuvik, with payload option 2.	86
4.23	Charge margin as a function of n_{batt} , identified by Configuration for zero payload, $b_{PV1} = 12$, month 6, flown in Inuvik.	86
4.24	SOC for each n_{batt} with a clean wing UAV, in month 9 with no payload, flown in Inuvik.	87

4.25	SOC for each n_{batt} with nacelles, in month 9 with no payload, flown in Inuvik.	87
4.26	Flight time as a function of n_{batt} , identified by n_{PV} for zero payload at $b_{PV1} = 12$, month 9, flown in Inuvik.	88
5.1	Sample simulated flight for optimization objective function definitions. . . .	90
5.2	Clean configuration span and chord variables.	94
5.3	Nacelle configuration span and chord variables.	94
5.4	Fitness convergence for the clean configuration without payload.	99
5.5	Constraints not meeting the tolerance threshold and their values for the clean configuration without payload.	100
5.6	Scatter of constraints not meeting the tolerance threshold for the clean configuration without payload.	101
5.7	Design variables for each iteration for the clean configuration without payload.	102
5.8	Converged solution for the clean configuration without payload. Axes are in feet.	103
5.9	Top view with solar panels for the clean configuration without payload. Axes are in feet.	103
5.10	Fitness convergence for the clean configuration with payload.	104
5.11	Scatter of constraints not meeting the tolerance threshold for the clean configuration with payload.	105
5.12	Design variables for each iteration for the clean configuration with payload.	106
5.13	Converged solution for the clean configuration with payload. Axes are in feet.	107
5.14	Top view with solar panels for the clean configuration with payload. Axes are in feet.	107
5.15	Converged solutions for the clean configuration, with payload in grey, without payload in blue. Axes are in feet.	108
5.16	Design variables for the clean configuration, with and without payload.	109
5.17	Fitness convergence for the nacelle configuration without payload.	111
5.18	Scatter of constraints not meeting the tolerance threshold for the nacelle configuration without payload.	111
5.19	Design variables for iterations 200 to 250 for the nacelle configuration, no payload.	113
5.20	Solution for the nacelle configuration without payload. Axes are in feet.	114
5.21	Top view with solar panels for the nacelle configuration, without payload. Axes are in feet.	114
5.22	Fitness convergence for the nacelle configuration, with payload.	115
5.23	Scatter of constraints not meeting the tolerance threshold for the nacelle configuration, with payload.	116
5.24	Design variables for iterations 200 to 250 for the nacelle configuration, with payload.	117
5.25	Solution for the nacelle configuration, with payload. Axes are in feet.	118
5.26	Top view with solar panels for the nacelle configuration, with payload. Axes are in feet.	118
5.27	Converged solutions for the nacelle configuration, with payload in grey, without payload in blue. Axes are in feet.	119
5.28	Design variables for the nacelle configuration, with and without payload.	121

5.29	Fitness convergence for the refined clean configuration without payload optimization.	123
5.30	Scatter of constraints not meeting the tolerance threshold for the refined clean configuration without payload optimization.	124
5.31	Converged solutions for the clean configuration without payload optimization. The initial result is in blue, the final solution is in grey. Axes are in feet.	125
5.32	Comparison of design variables for the clean configuration without payload optimizations.	126
5.33	Fitness convergence for the refined clean configuration, with payload optimization.	128
5.34	Scatter of constraints not meeting the tolerance threshold for the refined clean configuration with payload optimization.	129
5.35	Converged solutions for the clean configuration with payload optimization. The initial result is in blue, the final solution is in grey. Axes are in feet.	129
5.36	Comparison of design variables for the clean configuration with payload optimizations.	130
5.37	Fitness convergence for the refined nacelle configuration without payload optimization.	132
5.38	Scatter of constraints not meeting the tolerance threshold for the refined nacelle configuration, without payload optimization.	133
5.39	Converged solutions for the nacelle configuration without payload optimization. The initial result is in blue, the final solution is in grey. Axes are in feet.	134
5.40	Comparison of design variables for the nacelle configuration without payload optimizations.	135
5.41	Fitness convergence for the refined nacelle configuration, with payload optimization.	137
5.42	Scatter of constraints not meeting the tolerance threshold for the refined nacelle configuration, with payload optimization.	138
5.43	Converged solutions for the nacelle configuration with payload optimization. The initial result is in blue, the final solution is in grey. Axes are in feet.	139
5.44	Comparison of design variables for the nacelle configuration with payload optimizations.	140
5.45	Proposed geometry for a LALE solar UAV. Axes are in feet.	144
5.46	Aerodynamic coefficients of the proposed geometry, $Re_c = 142\,000$	146
5.47	Aerodynamic ratios with airspeed for the proposed geometry configured for payload option 1.	147
6.1	Sample flight with expected energy trends for the neural-controlled flight simulation.	149
6.2	NEAT constant radius circular trajectory.	151
6.3	Network for a constant radius circular trajectory.	152
6.4	States and controls for the first hour of a constant radius circular trajectory.	153
6.5	States and controls for a constant radius circular trajectory.	154
6.6	Power and SOC throughout the flight with the constant radius circular trajectory.	154
6.7	Variable radius circular trajectory path.	155

6.8	Network for the variable radius circular trajectory.	156
6.9	States and controls for the first hour of a variable radius circular trajectory.	157
6.10	States and controls for a variable radius circular trajectory.	157
6.11	Power and SOC throughout the flight with the variable radius circular trajectory.	158
6.12	Variable radius circular trajectory path, 21 December.	159
6.13	States and controls for a variable radius circular trajectory, 21 December.	159
6.14	Power and SOC throughout the flight with the variable radius circular trajectory, 21 December.	160
6.15	Constant heading trajectory path.	160
6.16	Network for the constant heading trajectory.	161
6.17	States and controls for a constant heading trajectory.	162
6.18	Power and SOC throughout the flight with the constant heading trajectory.	162
B.1	Visualization of BLDC motor volume prediction.	186
B.2	Statistical validation of 2-blade Electric APC Propeller prediction.	187
B.3	Statistical validation of 2-blade Regular APC Propeller prediction.	188
B.4	Statistical validation of 2-blade Slow Flight APC Propeller prediction.	189
B.5	Statistical validation of 2-blade Multi Rotor APC Propeller prediction.	190
B.6	Statistical validation of 2-blade Racing APC Propeller prediction.	191
B.7	Statistical validation of 2-blade Wide APC Propeller prediction.	192
D.1	Kahu drag polar.	195
F.1	Effects of $\mathcal{R}$ on total weight for $b_{PV1} = 10$, identified by n_{batt}	198
F.2	Effects of $\mathcal{R}$ on total weight for $b_{PV1} = 15$, identified by n_{batt}	199
F.3	Effects of $\mathcal{R}$ on total weight for $b_{PV1} = 17$, identified by n_{batt}	199
F.4	Charge margin as a function of n_{PV} , identified by n_{batt} with no payload for $b_{PV1} = 10$	200
F.5	Charge margin as a function of n_{PV} , identified by n_{batt} with no payload for $b_{PV1} = 15$	200
F.6	Charge margin as a function of n_{PV} , identified by n_{batt} with no payload for $b_{PV1} = 17$	201
F.7	Charge margin as a function of n_{PV} , identified by n_{batt} with payload option 2 for $b_{PV1} = 10$	201
F.8	Charge margin as a function of n_{PV} , identified by n_{batt} with payload option 2 for $b_{PV1} = 12$	202
F.9	Charge margin as a function of n_{PV} , identified by n_{batt} with payload option 2 for $b_{PV1} = 15$	202
F.10	Charge margin as a function of n_{PV} , identified by n_{batt} with payload option 2 for $b_{PV1} = 17$	203
F.11	Flight time as a function of n_{batt} , identified by payload option at $b_{PV1} = 10$	203
F.12	Flight time as a function of n_{batt} , identified by payload option at $b_{PV1} = 15$	204
F.13	Flight time as a function of n_{batt} , identified by payload option at $b_{PV1} = 17$	204
F.14	Flight time as a function of n_{batt} , identified by n_{PV} for zero payload at $b_{PV1} = 10$	205
F.15	Flight time as a function of n_{batt} , identified by n_{PV} for zero payload at $b_{PV1} = 15$	205

F.16	Flight time as a function of n_{batt} , identified by n_{PV} for zero payload at $b_{PV1} = 17$	206
F.17	Flight time as a function of n_{batt} , identified by n_{PV} for payload option 2 at $b_{PV1} = 10$	206
F.18	Flight time as a function of n_{batt} , identified by n_{PV} for payload option 2 at $b_{PV1} = 12$	207
F.19	Flight time as a function of n_{batt} , identified by n_{PV} for payload option 2 at $b_{PV1} = 15$	207
F.20	Flight time as a function of n_{batt} , identified by n_{PV} for payload option 2 at $b_{PV1} = 17$	208
F.21	Flight time as a function of n_{batt} , coloured by $\mathcal{R}$, identified by Configuration for no payload and $b_{PV1} = 10$	208
F.22	Flight time as a function of n_{batt} , coloured by $\mathcal{R}$, identified by Configuration for no payload and $b_{PV1} = 15$	209
F.23	Flight time as a function of n_{batt} , coloured by $\mathcal{R}$, identified by Configuration for no payload and $b_{PV1} = 17$	209
F.24	Flight time as a function of n_{batt} , coloured by $\mathcal{R}$, identified by Configuration for payload option 2 and $b_{PV1} = 10$	210
F.25	Flight time as a function of n_{batt} , coloured by $\mathcal{R}$, identified by Configuration for payload option 2 and $b_{PV1} = 12$	210
F.26	Flight time as a function of n_{batt} , coloured by $\mathcal{R}$, identified by Configuration for payload option 2 and $b_{PV1} = 15$	211
F.27	Flight time as a function of n_{batt} , coloured by $\mathcal{R}$, identified by Configuration for payload option 2 and $b_{PV1} = 17$	211
F.28	Flight time as a function of the number of solar cells on the horizontal stabilizer for Configurations 2, 3, 4, and 5, where $b_{PV1} = 10$ and $n_{batt} = 2$, without payload.	212
F.29	Flight time as a function of the number of solar cells on the horizontal stabilizer for Configurations 2, 3, 4, and 5, where $b_{PV1} = 15$ and $n_{batt} = 2$, without payload.	212
F.30	Flight time as a function of the number of solar cells on the horizontal stabilizer for Configurations 2, 3, 4, and 5, where $b_{PV1} = 17$ and $n_{batt} = 2$, without payload.	213
F.31	Flight time as a function of the number of solar cells on the horizontal stabilizer for Configurations 2, 3, 4, and 5, where $b_{PV1} = 10$ and $n_{batt} = 2$, with payload option 2.	213
F.32	Flight time as a function of the number of solar cells on the horizontal stabilizer for Configurations 2, 3, 4, and 5, where $b_{PV1} = 12$ and $n_{batt} = 2$, with payload option 2.	214
F.33	Flight time as a function of the number of solar cells on the horizontal stabilizer for Configurations 2, 3, 4, and 5, where $b_{PV1} = 15$ and $n_{batt} = 2$, with payload option 2.	214
F.34	Flight time as a function of the number of solar cells on the horizontal stabilizer for Configurations 2, 3, 4, and 5, where $b_{PV1} = 17$ and $n_{batt} = 2$, with payload option 2.	215
F.35	Flight time as a function of time of year, identified by n_{batt} for zero payload in Kingston, ON.	215

F.36	Flight time as a function of time of year, identified by n_{batt} for zero payload in Inuvik, NT.	216
F.37	Flight time as a function of time of year, identified by n_{batt} for payload option 2 in Kingston, ON.	216
F.38	Flight time as a function of time of year, identified by n_{batt} for payload option 2 in Inuvik, NT.	217
F.39	Charge margin as a function of n_{batt} , identified by Configuration for zero payload, $b_{PV1} = 12$, month 6, flown in Kingston.	217
F.40	Charge margin as a function of n_{batt} , identified by Configuration for payload option 2, $b_{PV1} = 12$, month 6, flown in Inuvik.	218
F.41	Charge margin as a function of n_{batt} , identified by Configuration for payload option 2, $b_{PV1} = 12$, month 6, flown in Kingston.	218
F.42	Flight time as a function of n_{batt} , identified by n_{PV} for zero payload at $b_{PV1} = 12$, month 3, flown in Inuvik.	219
F.43	Flight time as a function of n_{batt} , identified by n_{PV} for zero payload at $b_{PV1} = 12$, month 6, flown in Inuvik.	219
F.44	Flight time as a function of n_{batt} , identified by n_{PV} for zero payload at $b_{PV1} = 12$, month 12, flown in Inuvik.	220
F.45	Flight time as a function of n_{batt} , identified by n_{PV} for zero payload at $b_{PV1} = 12$, month 3, flown in Kingston.	220
F.46	Flight time as a function of n_{batt} , identified by n_{PV} for zero payload at $b_{PV1} = 12$, month 6, flown in Kingston.	221
F.47	Flight time as a function of n_{batt} , identified by n_{PV} for zero payload at $b_{PV1} = 12$, month 9, flown in Kingston.	221
F.48	Flight time as a function of n_{batt} , identified by n_{PV} for zero payload at $b_{PV1} = 12$, month 12, flown in Kingston.	222
G.1	Constraints not meeting the tolerance threshold and their values for the clean configuration with payload.	223
G.2	Constraints not meeting the tolerance threshold and their values for the nacelle configuration, no payload.	224
G.3	Scatter of constraints not meeting the tolerance threshold for the nacelle configuration, no payload.	224
G.4	Design variables for the nacelle configuration, no payload.	225
G.5	Constraints not meeting the tolerance threshold and their values for the nacelle configuration with payload.	226
G.6	Scatter of constraints not meeting the tolerance threshold for the nacelle configuration with payload.	227
G.7	Design variables for the nacelle configuration with payload.	228
G.8	Constraints not meeting the tolerance threshold and their values for the refined clean configuration without payload optimization.	229
G.9	Design variables for the refined clean configuration without payload optimization.	230
G.10	Constraints not meeting the tolerance threshold and their values for the refined clean configuration with payload optimization.	231
G.11	Design variables for the refined clean configuration with payload optimization.	232
G.12	Constraints not meeting the tolerance threshold and their values for the refined nacelle configuration without payload optimization.	233

G.13	Design variables for the refined nacelle configuration without payload optimization.	234
G.14	Constraints not meeting the tolerance threshold and their values for the refined nacelle configuration with payload optimization.	235
G.15	Design variables for the refined nacelle configuration with payload optimization.	236
H.1	Fitness convergence with relaxed constraints for the clean configuration, no payload.	239
H.2	Constraints violated and their values for the relaxed constraints for the clean configuration, no payload.	239
H.3	Scatter of constraints not meeting the tolerance threshold for the relaxed constraints for the clean configuration, no payload.	240
H.4	Solution for the clean configuration without payload with relaxed constraints shown in blue, the original vehicle shown in grey. Axes are in feet.	240
H.5	Design variables for the clean configuration without payload with relaxed constraints.	243
H.6	Fitness convergence with relaxed constraints for the clean configuration, with payload.	244
H.7	Constraints violated and their values for the relaxed constraints for the clean configuration, with payload.	244
H.8	Scatter of constraints not meeting the tolerance threshold for the relaxed constraints for the clean configuration, with payload.	245
H.9	Design variables for the clean configuration, with payload with relaxed constraints.	246
H.10	Solution for the clean configuration, with payload with relaxed constraints shown in blue, the original vehicle shown in grey. Axes are in feet.	248
H.11	Fitness convergence with relaxed constraints for the nacelle configuration, no payload.	248
H.12	Constraints violated and their values for the relaxed constraints for the nacelle configuration, no payload.	249
H.13	Scatter of constraints not meeting the tolerance threshold for the relaxed constraints for the nacelle configuration, no payload.	251
H.14	Solution for the nacelle configuration without payload with relaxed constraints shown in blue, the original vehicle shown in grey. Axes are in feet.	252
H.15	Design variables for the nacelle configurations without payload, with relaxed constraints.	253
H.16	Fitness convergence with relaxed constraints for the nacelle configuration, with payload.	254
H.17	Constraints violated and their values for the relaxed constraints for the nacelle configuration, with payload.	254
H.18	Scatter of constraints not meeting the tolerance threshold for the relaxed constraints for the nacelle configuration, with payload.	255
H.19	Design variables for the nacelle configurations with payload, with relaxed constraints.	256
H.20	Solution for the nacelle configuration with payload with relaxed constraints shown in blue, the original vehicle shown in grey. Axes are in feet.	257

I.1	Visualization of the orbit-raising problem [117].	260
I.2	Orbit-raising optimization results using trajectory optimization (data points) and NEAT (continuous curves).	262

Nomenclature

Roman Symbols

A	Area	[ft ²]
$\mathcal{R}$	Aspect ratio	[-]
b	Wingspan	[ft]
b_{PVx}	Number of solar cells on x th row of semi-span	[-]
C	Tail volume coefficient	[-]
c	Chord	[ft]
$\bar{c}$	Mean chord	[ft]
D	Drag	[lb _f]
E	Total maximum battery capacity	[Wh]
e	Oswald's efficiency	[-]
e_1	Squareness factor	[-]
g	Gravitational acceleration	[$\frac{\text{ft}}{\text{s}^2}$]
I	Current	[A]
I_0	No load current	[A]
I_s	Solar irradiance	[$\frac{\text{W}}{\text{m}^2}$]
I_{nom}	Nominal capacity	[Ah]
K	Lift induced drag factor	[-]
K_V	Motor KV constant	[$\frac{\text{rpm}}{\text{V}}$]
L	Lift	[lb _f]
ℓ	Length	[in]
M	Bending moment	[lb _f ft]
m	Mass	[g, lb _m]

m_{batt}	Number of battery cells in series	[-]
n_{batt}	Number of battery cells in parallel	[-]
n_{PV}	Number of solar cells	[-]
$\vec{n}_s$	Solar vector	[-]
n_z	Normal load factor	[g]
P	Power	[W]
Q	Torque	[lb _f ft]
R	Resistance	[Ω]
r	Turning radius	[ft]
$\bar{R}$	Rotational transformation matrix	[-]
S	Lifting surface reference area	[ft ²]
SM	Static margin	[-]
SOC	State of charge	[-]
T	Thrust	[lb _f]
t	Thickness	[in]
t	Time	[h]
T_{set}	Thrust setting	[-]
V	Velocity	[$\frac{ft}{s}$]
V	Voltage	[V]
$\bar{V}$	Volume	[in ³]
W	Weight	[lb _f]

Greek Symbols

α	Angle of attack	[deg]
η	Efficiency	[-]
η	Nondimensional span location	[-]
γ_1	Bank angle	[deg]
γ_2	Climb gradient	[deg]
γ_s	Solar elevation angle	[deg]
κ	Obliquity factor	[-]

A	Sweep angle [deg]
λ	Taper ratio [-]
ω	Rotational speed [$\frac{\text{rad}}{\text{s}}$]
Φ	Diameter [in]
ϕ	Bank angle [deg]
ψ	Azimuth angle [deg]
ρ	Density [$\frac{\text{slug}}{\text{ft}^3}$]
σ	Maximum allowable bending stress [$\frac{\text{lb}_f}{\text{ft}^2}$]
θ	Pitch angle [deg]
θ_{prop}	Propeller pitch [in]

Dimensionless Groups

C_D	Drag coefficient [-]
C_L	Lift coefficient [-]
C_m	Pitching moment coefficient [-]
C_{D_0}	Parasitic drag [-]
C_{D_i}	Lift induced drag [-]
$C_{L,\alpha}$	Lift curve slope [rad^{-1}]
$C_{l,\beta}$	Static roll coefficient [rad^{-1}]
C_{L_0}	Lift coefficient at zero angle of attack [-]
$C_{n,\beta}$	Static yaw coefficient [rad^{-1}]

Roman Subscripts

0	Takeoff
B	Body axis
b	Due to bending
$batt$	Battery
CG	Centre of gravity
cm	Charge margin
D_{min}	Minimum drag
E	Earth axis

<i>e</i>	Empty
<i>exc</i>	Excess
<i>HT</i>	Horizontal tail
<i>hub</i>	Propeller hub
<i>i</i>	Inner
<i>LE</i>	Leading edge
<i>m</i>	Motor
<i>max</i>	Maximum
<i>min</i>	Minimum
<i>mpp</i>	Maximum power point
<i>o</i>	Outer
<i>oe</i>	Operational empty
<i>p</i>	Propulsion
<i>pl</i>	Payload
<i>prop</i>	Propeller
<i>PV</i>	Solar Panel
<i>r</i>	Root
<i>s</i>	Solar
<i>shaft</i>	Propeller shaft
<i>t</i>	Tip
<i>V</i>	Velocity axis
<i>VT</i>	Vertical tail
<i>w</i>	Wing

Abbreviations

ARPA	Advanced Research Projects Agency
BLDC	Brushless Direct Current
CREATEv	Clean Renewable Energy Aerial Test Vehicle
EDT	Eastern Daylight Time (UTC-04:00)
ESC	Electronic Speed Controller

EST	Eastern Standard Time (UTC-05:00)
FAI	Fédération Aéronautique Internationale
HALE	High Altitude Long Endurance
ISR	Intelligence, Surveillance, Reconnaissance
LALE	Low Altitude Long Endurance
Li-ion	Lithium-Ion
LiPo	Lithium Polymer
MAC	Mean Aerodynamic Chord
MALE	Medium Altitude Long Endurance
MDT	Mountain Daylight Time (UTC-06:00)
MPPT	Maximum Power Point Tracker
MSL	Mean Sea-Level
MST	Mountain Standard Time (UTC-07:00)
NEAT	NeuroEvolution of Augmenting Topologies
RMSE	Root Mean Square Error
SOC	State of Charge
UAV	Unmanned Aerial Vehicle
UTC	Coordinated Universal Time

1 Introduction

1.1 Motivation for the Research

The relevance of long endurance Unmanned Aerial Vehicles (UAVs) has become increasingly prevalent with the rapid evolution of drone warfare and the rising frequency of natural disasters. Coupled with the increase in battery energy density and the efficiency of photovoltaic (or solar) cells, UAVs are an attractive solution to long endurance Intelligence, Surveillance, and Reconnaissance (ISR) missions, communications and signals relay, or environmental monitoring for several days, weeks, or months.

There are three main categories which dictate the general size and endurance of solar UAVs: Low-, Medium-, and High-Altitude Long Endurance (LALE, MALE, and HALE, respectively). Throughout this work, low-altitude refers to altitudes lower than 10 000 ft, medium-altitude refers to altitudes between 10 000 ft and 30 000 ft, and high-altitude refers to altitudes higher than 30 000 ft. The general endurance capacity of a solar-powered UAV scales with altitude, where the current record for LALE solar-powered flight is 81.4 hours and the record for HALE solar-powered flight is 67 days [1,2]. Regardless of altitude, solar-powered UAVs have vast applications for observing the development and progression of natural disasters, monitoring the spread of oil spills and containment processes, monitor oceanic thermal currents, identify sources of natural resources including mineral, oil, or geothermal power, monitor pollution and toxic gas releases, or expand cellular networks specifically in remote areas or after natural disasters [3,4]. The geographical location and time of year has a significant influence on the available solar energy. At latitudes further from the equator, the number of days that prolonged flight is possible throughout the year diminishes. Ideally, the most desirable design would be capable of prolonged flight independently of the time of year or geographical location. For a solar-powered vehicle, this is impossible at extreme latitudes where there is no daylight through the winter months.

A common feature between all solar-powered UAVs capable of prolonged flight (*i.e.*, continuous flight through one or more 24 h solar cycles) is a high aspect ratio wing. A major design challenge of HALE solar-powered UAVs are the aeroelastic effects while climbing to their cruise altitude and more intensive ground procedures for handling and launching. LALE UAVs do not require as high of aspect ratios to maintain lift at the higher atmospheric density than their HALE counterparts, thereby not suffering from aeroelastic effects to the same degree nor traversing the most weather-intensive altitudes. The endurance of low altitude vehicles is more limited by weather conditions, where overcast skies prevent the vehicle from extracting the solar energy necessary to prolong its flight time. LALE UAVs are smaller in weight and size compared to

HALE UAVs, making them more readily deployable and more flexible as local signals or communications relays or local environmental monitoring, particularly with shorter turnaround times. To be practically employed, the design process must consider possible payloads and the time of year of operation, where the optimization of the UAV should consider these factors.

There are two primary methods of optimizing a solar-powered UAV; the vehicle design itself and how the vehicle is flown. The vehicle optimization is dependent on various models including propulsion systems, aerodynamics, and weight prediction. Most aircraft analysis tools can be applied with minor adjustments to analyze small UAVs. The weight prediction has its own challenges because classical aircraft weight prediction methods are rooted in historical aircraft data for validated semi-empirical models which does not include or apply to small UAVs. The limited databases and published methods for the sizing of internal and structural components of small UAVs is currently a limiting factor in the sizing analysis and therefore optimization of the UAV design. Furthermore, the trajectory optimization of a long endurance vehicle is highly computationally expensive: if considering one day of flight and a 1 s time step, there are 86 400 discrete time steps that must maintain continuity between the states of the vehicle and satisfy numerous constraints. That optimized trajectory would be specific to a given day's solar conditions and may not be practical to implement in real flight conditions with stochastic weather events including wind gusts or cloud cover. It would be very computationally expensive to couple the vehicle and trajectory optimization into a singular problem. Current research assumes a path and optimizes the vehicle or assumes a vehicle and optimizes the path.

1.2 Research Objectives

This research aims to optimize and design a LALE solar-powered UAV for operations within Canada with the future objective of building and testing the proposed vehicle. The assumptions embedded into the analysis are indicative of currently available and accessible technologies. The overarching objective can be broken into the following specific research objectives:

- i. Develop weight prediction methods for the sizing of small electric propulsion components and the structural weight of small fixed wing UAVs;
- ii. Evaluate various UAV configurations to determine the most promising for LALE flight,
- iii. Determine the configuration trade-offs between aerodynamic efficiency and energy capacity for a suitable LALE solar-powered UAV and confirm the optimality of a glider-like wing;
- iv. Optimize the geometry, energy capacity, and propulsion system of a solar-powered UAV to maximize energetic margins and broaden the operating window throughout a year for which persistent flight can be achieved for a given latitude;
- v. Propose a geometry for a solar-powered UAV for operations within Canada; and
- vi. Evolve a neural network for the optimal control of the proposed UAV to validate optimal flight strategies for LALE flight.

1.3 Thesis Layout

This thesis is organized into six chapters that follow this introduction. Chapter 2 presents a literature review with a detailed history of aviation and methods for designing and optimizing a solar-powered UAV. The methods used for analyzing and simulating the flight of a solar-powered UAV are presented in Chapter 3, including the developed methods for sizing the weight of electric propulsion components and the wing weight of a small fixed wing UAV. Chapter 4 analyzes the effects of various geometric, payload, and electric properties to determine the most promising configuration for LALE flight. The most promising configuration from Chapter 4 is optimized in Chapter 5. The results of the optimization are used to recommend a final geometric design for building and testing in future works. The properties of the proposed vehicle are used in Chapter 6 to evolve neural networks as flight controllers as an alternative to trajectory optimization to explore flight strategies for LALE flight. Chapter 7 provides concluding arguments and recommendations for future areas of research and development on this topic.

1.4 Contributions

The contributions of this thesis include:

- i. A detailed, consolidated history of notable strides and achievements in solar-powered aviation;
- ii. Proposed weight prediction methods for brushless DC motors, small propellers, electronic speed controllers, and the wing structure of small fixed wing UAVs;
- iii. A detailed parametric analysis on the configuration selection for a solar-powered UAV;
- iv. A detailed parametric analysis showing that the optimal tradeoff between aerodynamic efficiency and energy capacity is highly dependent on payload, such that deviating from a high- $\mathcal{R}$ wing is more advantageous as power requirements increase;
- v. A proposed optimized design for a solar-powered UAV which maximizes operational capability; and
- vi. Confirmation of viable trajectories for LALE solar-powered flight.

2 Literature Review

This chapter presents research to date on solar Unmanned Aerial Vehicles (UAVs). This includes a historical review of solar flight and the challenges associated with it, followed by methods for sizing small scale electric aircraft. Methods for optimizing UAVs with the overarching objective of increasing endurance are presented next, including path planning, trajectory optimization, and vehicle optimization. Subsequently, additional considerations for solar UAV design are included. The chapter closes with identifying the knowledge gap aimed to be filled by this thesis.

2.1 History of Solar Flight

The aim of this section is to provide an overview of significant and well-recorded events and key aircraft in the history of solar flight. By no means is this summary comprehensive to every accomplishment. The intent is to discuss some of the aircraft that have contributed to the current state of the art of solar-powered flight from its inception to present day. Basic geometric details of the aircraft discussed herein and additional solar UAVs are summarized in Appendix A.

As with powered flight, solar-powered flight begins with two brothers: Roland and Robert Boucher. The Boucher brothers began experimenting with electric flight in 1970, concentrating on achieving sufficient power to practically replace internal combustion engines. Their work began with small radio-controlled models, eventually capable of performing simple aerobatic manoeuvres. With the level of battery technology, agencies were pessimistic of ever achieving battery energy densities that could match or rival gasoline. Electric flight was attractive as a means of surveillance, where the lack of exhaust made it more difficult to be picked up by infrared sensors. The Bouchers worked together aiming to design an electric aircraft capable of 12 or more hours of continuous flight to secure the interest of the Advanced Research Projects Agency (ARPA), named Sunrise I [5].

Sunrise I made history with its maiden flight on 4 November 1974. There were no batteries on board; the propulsion system was powered entirely from the solar panels on the 32 ft span wing, weighing 27.5 lb_f. Sunrise I proved capable of 3 to 4 hour flights during windy and winter conditions. Although not able to fly for 12 hours, ARPA was encouraged to authorize continued work on the design, leading to Sunrise II, shown in Figure 2.1. Reduction in solar cell weight and increase in efficiency allowed Sunrise II to be lighter at 22.5 lb_f with significant theoretical increases in performance. The Sunrise aircraft proved the ability of heavier-than-air solar-powered flight, paving the way for future designs to emerge [5].



Figure 2.1: Sunrise II, 1975 [6].

In the spring of 1979, Larry Mauro successfully piloted Solar Riser, a bi-plane glider converted to a solar bi-plane shown in Figure 2.2. This vehicle had enough battery capacity for 10 minutes of powered flight then used solar panels on the wing to charge the battery in about 3 hours back on the ground. Although this flight was crewed, it is not considered manned solar-powered flight because the propulsion system was unable to be powered by the solar array. Its longest flight was 0.5 miles at altitudes between 5 ft and 15 ft [5].



Figure 2.2: Solar Riser, 1979 [6].

Two months later, Freddie To and his team launched Solar One on its maiden flight. Solar One was originally built as a man-powered plane with the intention of crossing the English Channel. The plane required more power than one pilot could provide and was converted to solar-powered. Similar to the Solar Riser, Solar One was able to fly shorter duration flights on battery power then land and recharge the batteries with solar power. The performance after its maiden flight on 13 June 1979 was not encouraging enough to attempt crossing the Channel. The entirety of the upper wing surface was not covered with solar cells due to the high cost. The price of solar cells at this time limited the development of applications for solar power and the potential for solar-powered flight [5].

The Gossamer Penguin, shown in Figure 2.3, was the first manned solar-powered flight on 18 May 1980. It was powered directly from the solar cells and had no onboard batteries. To reduce weight, the Gossamer Penguin was piloted by 13-year-old Marshall MacCready, accomplishing a 500 ft range at 5 ft of altitude. The aircraft was initially

designed as the backup aircraft to the Gossamer Albatross, the man-powered plane pedalled across the English Channel the year prior. The designers moved on to developing the Solar Challenger, in attempt to cross the Channel on solar power alone [5].



Figure 2.3: Gossamer Penguin, 1980 [6].

The Solar Challenger, developed with the goal of crossing the English Channel using solar power alone, is shown in Figure 2.4. The aircraft has a custom airfoil design where 85% of the chord was flat to allow smooth integration of solar cells. After assembly and testing, the Solar Challenger was ready to cross the Channel. Being completely reliant on solar power, the flight would only be possible with sunny skies and reduced winds. After weeks of weather delays, Solar Challenger successfully crossed the English Channel on 7 July 1981, marking the first solar-powered plane to cross the Channel and the first international solar-powered flight at a total weight of 119 lb_f [5].



Figure 2.4: Solar Challenger, 1981 [6].

As a competitor to the Solar Challenger, Gunter Rochelt and his wife spent years hand-crafting the Solair I, shown in Figure 2.5. Particular emphasis was placed on the aerodynamics design. The plane was piloted and flew on battery and solar power simul-

taneously. When the battery ran out of capacity, Gunter relied on thermals to climb and remain aloft, allowing the solar panels to recharge the battery. Flight times of up to 5 hours were reported, featuring a 53 ft span and 440 lb_f takeoff weight [5].



Figure 2.5: Solair I, 1981 [6].

Throughout the 1980's and early 1990's, several small scale solar vehicles were flown. Most had wingspans within the range of 5 ft to 11 ft, weighing no more than 7 lb_f. Little else is recorded on these aircraft, other than basic dimensions shown in Appendix A [6].

The Sunseeker was a piloted solar-powered aircraft, shown in Figure 2.6. The design begun in 1986 and was tested as a glider in 1989. In 1990, the Sunseeker flew across the USA in 21 flights with 121 hours in the air, relying on solar power and thermals [6]. The designer, Eric Raymond, and the team at Solar Flight later developed an updated version called Sunseeker II which was the first solar and battery powered aircraft to fly across the Alps in 2009 [7]. The company later developed the Sunseeker Duo, the first two-seat solar-powered aircraft. The Sunseeker Duo's maiden flight occurred in June 2013 and it has logged several hundred flight hours since [8].



Figure 2.6: Sunseeker, 1990 [6].

In 1996, the challenge for the Ulm Berblinger aeronautical competition was to create a manned solar-powered airplane. More than 30 projects were announced but fewer arrived at the competition. Those presented as prototypes included Solair II, a continued development after Solair I by Gunter Rochelt. Only one aircraft was ready to fly at the competition; Icaré 2 from the Stuttgart University, shown in Figure 2.7. Icaré 2 won the competition and the prize money. Solair II flew two years later in May 1998 [6].



Figure 2.7: Icaré 2, 1996 [6].

Since the introduction and proven feasibility of solar-powered flight, it has been imagined that solar technology could be used for perpetual flight, long endurance communication relays, and high altitude atmospheric probes among other applications [5]. The US government provided funding to study the feasibility of HALE solar-powered flight. The first prototype, HALSOL, by AeroVironment proved the structures and aerodynamics but lacked the sophistication required for solar cell and battery technologies required for HALE solar flight. It was flight-tested with battery power to prove the concept but the solar project was terminated in 1983 [3]. As technology improved, the concept was revisited with Pathfinder, achieving its first flight in 1993. Pathfinder was adopted into the NASA Environmental Research Aircraft and Sensor Technology (ERAST) program. The ERAST program was a response to the increasing demand for HALE aircraft, supporting prospective projects for pseudo-satellites to act as communication relays and surveillance platforms. Under the ERAST program, flying-wing solar UAVs were developed in sequence, shown in Figure 2.8, increasing in wingspan with each configuration. In chronological order, the prototypes were Pathfinder, Pathfinder Plus, Centurion, and Helios, shown in Figure 2.9 [7, 9].

Helios was intended to demonstrate sustained flight at an altitude of 100 000 ft with an endurance of 24 hours, including 15 hours above 50 000 ft. In 2001, Helios sustained an altitude of 96 000 ft for 40 minutes and a maximum altitude of 96 863 ft [7]. Its 247 ft wingspan and aspect ratio of 31 was, however, susceptible to deflection and aeroelastic effects. During its second flight on 26 June 2003, the aircraft encountered turbulence causing an unexpected, persistent, high dihedral deflection, creating instability in pitch, causing the aircraft to enter an unrecoverable dive, shown in Figure 2.9. After losing control of the aircraft, it exceeded the design dynamic pressure which caused the skin

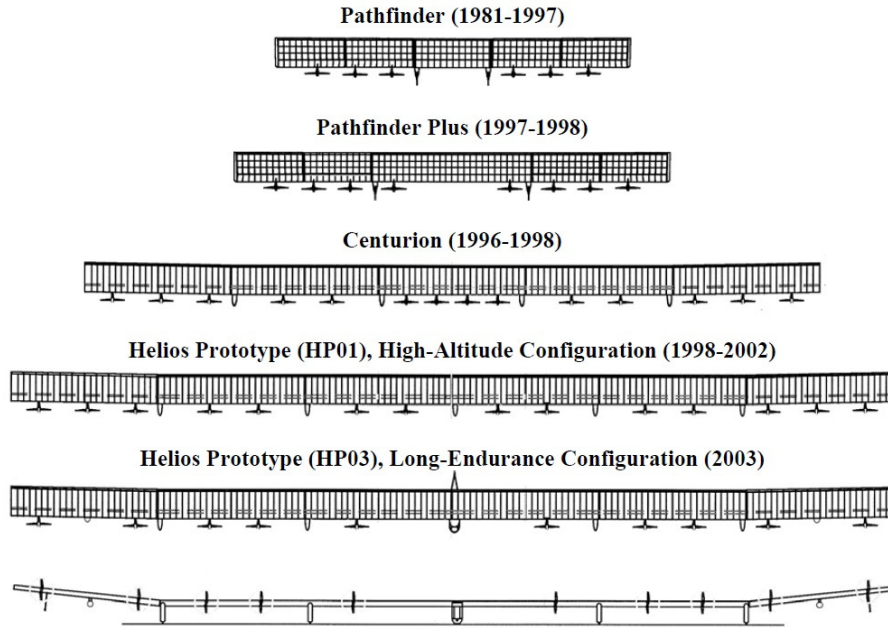
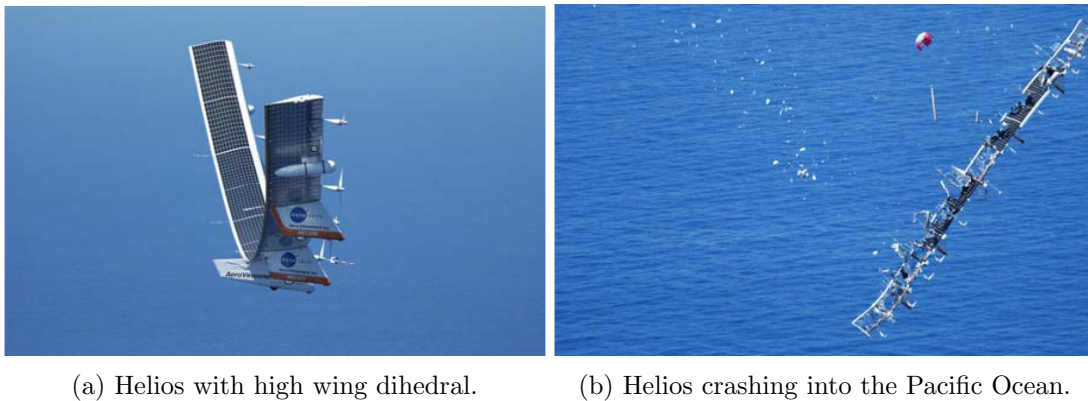


Figure 2.8: ERAST sponsored solar-powered aircraft [9].

of the wing and the embedded solar panels to separate from the structure. The aircraft crashed into the Pacific Ocean off the coast of Kauai [9]. Helios serves as a reminder of the challenges of high aspect ratio HALE aircraft, which are sensitive to aeroelastic effects. Helios did not achieve its objective of continuous flight of at least 24 hours [6].



(a) Helios with high wing dihedral.

(b) Helios crashing into the Pacific Ocean.

Figure 2.9: Helios mishap, 2003 [9].

Research and testing of smaller scale, fixed-wing solar UAVs increased with technological improvements in battery energy density and the weight and efficiency of solar cells. The reduction in cost of solar cells allowed for smaller teams and more projects to explore solar flight. Continuous solar-powered flight for more than 24 hours was realized by the SoLong solar UAV, a project undertaken by Alan Cocconi as the chairman of

AC Propulsion. Alan Cocconi and SoLong are shown in Figure 2.10. With a 15.6 ft wingspan, the SoLong achieved 48 hours 11 minutes starting on 1 June 2005 in California [10]. There were six experienced pilots that brought the SoLong to its record flight time, relying on thermals and unpowered flight to allow the batteries to recharge. The record flight was terminated due to pilot exhaustion rather than any limitations in the energy storage [7].



Figure 2.10: Alan Cocconi with SoLong [10].

In 2006, students from Technion IIT, Haifa formed a project with the intention of setting a new FAI (Fédération Aéronautique Internationale) world record for distance flown by a solar-powered UAV. To set the record, the aircraft needed to weigh less than 11 lb_f, have less than 16.1 ft² of upper surface area, and be strictly powered by solar cells, *i.e.*, no batteries onboard [7]. The final design, SunSailor, had a 13.8 ft span and weighed 7.9 lb_f [6]. Unfortunately, both aircraft that were built crashed before the record could be set [7]. SunSailor and the design team are shown in Figure 2.11.



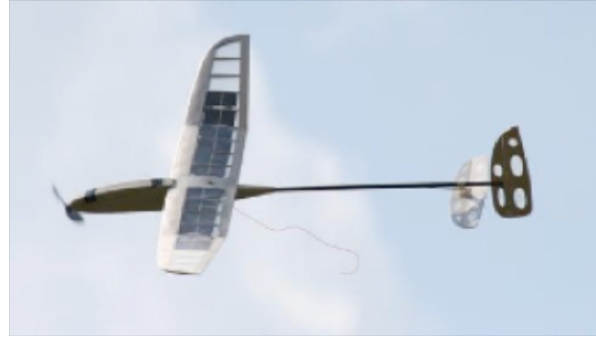
Figure 2.11: SunSailor and design team [11]

Starting in 2004, André Noth started a project to study the feasibility of a solar-powered aircraft to fly on Mars. After developing a flexible design tool for weight predictions

and sizing of UAVs, Noth designed, built, and tested Sky Sailor, shown in Figure 2.12a. Testing began in 2005 with several flights up to 10 hours. In 2008 its full potential was realized with a record endurance of 27 hours [12]. Noth’s conceptual design methodology was used to develop the Sun Surfer in 2007. The aim of this project was to increase the endurance of a small UAV by incorporating solar cells to supplement the onboard batteries. The final design of Sun Surfer had a 2.5 ft span and an empty weight of 0.256 lb_f, shown in Figure 2.12b [6, 7].



(a) André Noth and Sky Sailor [6].



(b) Sun Surfer, 2007.

Figure 2.12: Designs from Noth’s framework.

Meanwhile, the UK company QinetiQ was developing the Zephyr HALE solar UAV, shown in Figure 2.13. The first flight tests occurred in 2005, with one vehicle achieving 6 hours of flight and climbing to 26 000 ft. Since its first flights in 2005, further models of Zephyr have continuously broken endurance records. Table 2.1 summarizes the record-breaking flight times with basic geometric properties of the model. Now owned by AALTO, Zephyr is being used and further developed to expand 4G and 5G cellular networks and provide surveillance capabilities from the stratosphere [13].



Figure 2.13: Zephyr [13].

Table 2.1: Zephyr record-breaking flight characteristics.

Year	Model	Endurance	Span [ft]	Weight [lb _f]	
2005	4	6 h	39.4	37.5	[14]
2006	5	18 h	52.5	68.3	[14]
2007	6	54 h	59.1	66.1	[15]
2008	6	82 h 37 min	59.1	66.1	[15]
2010	7	14 d	73.8	116.8	[16]
2018	8	25 d 23 h 57 min	82.0	143.3	[17]
2022	8	64 d	82.0	143.3	[18]
2025	8	67 d 6 h 52 min	82.0	165.3	[2]

In the same time frame, multiple low altitude solar-powered designs were explored. Similar to the Gossamer Penguin, a human powered airframe was converted to a solar-powered UAV by Aurora Flight Science’s [19]. The SunLight Eagle, shown in Figure 2.14, was flight-tested in May 2009, collecting data on its aerodynamic performance and solar cells. Next steps included longer flights at higher altitudes, but it is unclear if these flights occurred or what any outcomes were [20].



Figure 2.14: SunLight Eagle, 2009 [20].

A popular and relatively recent accomplishment in solar-powered flight was the Solar Impulse 2 flight around the world completed in 2016. The two pilots, Bertrand Piccard and André Borschberg sought to fly around the world strictly on solar power. The aircraft was not designed to “carry passengers, but to carry messages”, with the desire to get people to “question what they’ve always taken for granted” [21]. The project started with Solar Impulse 1 which completed its maiden flight in December 2009. Its longest flight was 19 hours, travelling from Madrid, Spain to Rabat, Morocco in 2012 [22]. The second model, Solar Impulse 2, shown in Figure 2.15, had a 236 ft span and a maximum weight of 5070 lb_f. Solar Impulse 2 captured global attention as it prepared for and completed its trip around the world in 17 flights. The longest flight was from Nagoya to Hawaii, flown by André Borschberg, lasting 4 days 21 hours 52 minutes in 2015. The

entire project was completed in 17 legs, with a total of 550 flight hours and multiple FAI records [21].



Figure 2.15: Solar Impulse 2 [21].

More recent efforts in solar-powered flight focussed on smaller scale, lower altitude, UAV platforms. AtlantikSolar, shown in Figure 2.16, was an 18.7 lb_f hand-launched UAV designed, built, and tested by a group from the Swiss Federal Institute of Technology in Zurich [1]. It followed work done with Sky Sailor and Sun Surfer, using the same conceptual design framework [10,12]. It was designed with the goal of maximizing perpetual flight robustness, making perpetual flight less sensitive to weather conditions that otherwise reduce flight time. With larger energetic safety margins, AtlantikSolar completed an 81 hour flight in July 2014. Although not registered with FAI, this was a global endurance record for an aircraft lighter than 110 lb_f (50 kg). AtlantikSolar is further proof of the viability of perpetual flight at altitudes and overall size significantly lower than HALE aircraft [1].

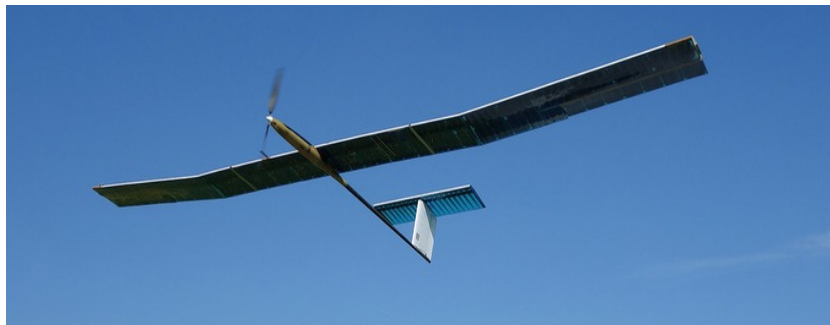


Figure 2.16: AtlantikSolar, 2015 [1].

Months after Solar Impulse 2's successful circumnavigation, another Swiss adventurer announced and revealed a solar-powered aircraft intended to reach the stratosphere with its pilot onboard. The SolarStratos, shown in Figure 2.17, was 8.5 m long, weighed 992 lb_f, and is expected to reach an altitude of 82 000 ft in future flights. The maiden

flight in 2017 at low altitudes “went off without a hitch” [23]. Several flights have followed, with a notable altitude record for manned solar-powered flight reaching 31 000 ft on 12 August in 2025. The 237 ft² of solar panels are meant to provide 24 hours of continuous flight time; however, it was unclear if the vehicle was capable of recharging the batteries in flight or if the power draw of the systems are too high to sustain longer missions [23]. Ultimately, its main objective is to reach the stratosphere, not sustain long endurance flights [24].



Figure 2.17: SolarStratos [24].

There have been several other attempts to achieve more than 24 hour continuous flight or extend UAV range and or endurance by relying on solar power. MARAAL, shown in Figure 2.18, was developed for surveillance and mapping applications. It has a theoretical endurance of 18 hours and achieved 8.5 hours during a flight test on 1 April 2017 in Kanpur. MARAAL is comparable in wingspan to AtlantikSolar but weighed nearly twice as much, explaining the difference in endurance performance [25].



Figure 2.18: MARAAL, 2017 [25].

Facebook had its own program for a high altitude solar pseudo-satellite to serve as relay stations to provide internet to remote areas. The first flight of the flying wing vehicle,

Aquila, was 90 minutes long and suffered significant structural damage upon landing in June 2016. The project's second flight occurred in May 2017, lasting 1 hour 46 minutes and landing successfully on the prepared landing site. Aquila, shown in Figure 2.19, at 140 ft had a wingspan greater than that of a Boeing 737 and was intended for missions near 60 000 ft in altitude. Aquila was grounded and the project terminated in 2018, intending to continue the pursuit of high-altitude aircraft with partners like Airbus [26–28].



Figure 2.19: Facebook's Aquila [26].

Rosales *et al.* incorporated solar cells into a fixed-wing UAV with the objective of increasing endurance as an undergraduate project, shown in Figure 2.20. The UAV had a 12.7 ft span, weighed 14.2 lb_f, and flew for 33 minutes in its first test flight in November 2018, limited by the onboard battery depleting [29, 30].



Figure 2.20: Full-scale model by Rosales *et al.* [29].

Mini-Phantom, shown in Figure 2.21, was developed to test the feasibility of sustained solar flight in low temperatures and high altitudes, specifically for applications of monitoring rare wildlife over the Tibetan plateau in winter. Flight tests in 2019 proved that contemporary battery and solar cell technology is sufficient for colder and medium operating altitudes (around 15 000 ft MSL). Theoretically, Mini-Phantom could fly for 29.3 hours in summer and 20.8 hours in winter with the same flight conditions and a takeoff time of midnight [31].



Figure 2.21: Mini-Phantom, 2019 [31].

The German-developed Solar Flyer was constructed of commercial-off-the-shelf components, aiming to sustain continuous flight. The project uses a commercial fixed-wing UAV and converts it to a solar-powered aircraft, shown in Figure 2.22. Flight testing on 31 August 2020 showed that the aircraft was capable of 8 hours of flight in non-ideal (cloudy, windy, and gusty) conditions, indicating that with better flight conditions the vehicle would be capable of flying for longer. In a shorter 1 hour flight under sunny and calmer conditions, the design proved to have a balanced energy expenditure and collection, which further supported its potential to fly longer missions [32].



Figure 2.22: Solar Flyer, 2020 [32].

Starting in 2020, under the supervision of Goetz Bramesfeld, an aircraft design team has been aiming to break the world record for the longest autonomous flight. At the inception of the project, this record was 25 days 23 hours 57 minutes set by Zephyr in 2018. The Clean Renewable Energy Aerial Test Vehicle (CREATeV), shown in Figure 2.23, was a conventional glider-like design with lightweight structures and a high

aspect ratio. They achieved their first record endurance flight of 10.5 hours, concluding that a 20% weight reduction and 3-4% gain in propulsive efficiency would be required as a minimum to attempt breaking the world endurance record [33]. Nearly a year later, with improvements to the design, CREATEv was able to fly for 18 hours using solar power and onboard batteries. In July 2022, CREATEv was able to complete a 32 hour flight [34]. CREATEv had a 20.6 ft span and weighed 26.5 lb_f. Presently, the record for longest autonomous flight belongs to Zephyr at 67 days [2]. Due to the dependency on weather conditions for LALE flight, breaking this record at low altitudes is a difficult challenge, regardless of latitude.



Figure 2.23: CREATEv, 2020 [33].

Similar to Solar Flyer, Sun was a 759-2 Phoenix 2000 RC plane modified to be fitted with solar cells on the upper wing surface, shown in Figure 2.24. The aim of the project was to increase the endurance by incorporating solar power. The aircraft was shown to have a 22.5% in-flight power saving in the use of battery stored capacity. The longest circuit flown lasted 7 min 27 s. The vehicle was not tested for maximum endurance [35].



Figure 2.24: Sun UAV [35].

On 2 June 2022, a Taiwanese team set an endurance record of 21 hours 50 minutes. Their UAV had a 16.5 ft wingspan and weighed 11 lb_f. The original objective of the project was to out-fly the 24 hour solar cycle. Reasons this was not achieved include strong winds during testing, an inability to achieve the target altitude, and a delayed

takeoff time from the original flight plan. With further reductions in weight and the installation of a smoother upper wing surface with the solar panel integration, the team expects that the 24 hour endurance target is within reach [36]. It is unclear if further design iterations or longer flight tests have been conducted.

The sixth iteration of a solar-powered UAV, MoonFlyer, shown in Figure 2.25, designed by a group from National Formosa University achieved a 25 hour 53 minute flight in Letiste Breclav, Czech Republic. The flight commenced at 0314 local time on 10 July 2024. The aircraft climbed up to approximately 5000 ft during daylight hours, allowing the UAV to glide and reduce power consumption after sunset. This flight set a record for both Taiwan and the Czech Republic for the longest unmanned aircraft flight [37].



Figure 2.25: MoonFlyer, 2024 [37].

Prismatic, a subsidiary of BAE Systems, began developing the PHASA-35 in 2016. It resembled Zephyr as a large glider-like vehicle, except it had a larger wingspan, nearly double the payload capacity, and was twice as heavy as Zephyr [4]. PHASA-35, shown in Figure 2.26, achieved stratospheric flight at 66 000 ft for at least four hours in a 24 hour mission in 2023 [38]. More recent testing had PHASA-35 landing after a 24 hour flight in a serviceable condition, making it ready for flight again two days later. The quick turnaround between flights was promising for expanding further operational capacities for the platform [39].



Figure 2.26: PHASA-35 [39].

SoftBank Corp. in Japan continued work from AeroVironment with the development of SunGlider. It was a continuation of the research and development from Pathfinder through to Helios. SunGlider was a high altitude flying wing solar-powered platform with a wingspan of 256 ft [4]. The company aimed to offer high-capacity, high-quality, stable communications with a network of UAVs as pseudo-satellites in the stratosphere, with

payloads up to 165 lb_f [40]. In 2020, Sunlider flew for 5.5 hours, reaching 62 500 ft [4]. Field tests conducted by AeroVironment and US Department of Defence in August 2024 led to further platform evolution, a step closer towards commercialization [40].

After the Solar Impulse 2 completed its flights around the globe, Skydweller was founded by Robert Miller and John Parkes. The company transformed Solar Impulse 2 into an autonomous vehicle with a payload capacity of 880 lb_f. Skydweller, also the name of the transformed aircraft, is shown in Figure 2.27. The company aimed to provide similar capabilities as the ERAST program and the current Zephyr projects, with long endurance ISR capabilities, acting as a telecommunications relay, and monitoring natural disasters such as oil spills or forest fires as a MALE platform [41]. In October 2024, Skydweller endured 22.5 hours on a test flight, terminating only to avoid a hurricane. Test-flight data indicated that the design had better energy margins than expected, making their goal of 90 day continuous flight closer to reality [42]. In the summer of 2025, Skydweller completed back-to-back 73 hour and 74 hour tests, getting closer to 90 continuous days [41]. In May 2026, Skydweller crashed due to inclement weather and depletion of battery charge in the Gulf of Mexico after an eight day test-flight. The strong winds increased the power draw of the aircraft until Skydweller was no longer able to sustain flight. All systems performed as expected and there were no technical failures reported. The ability to progress towards the commercialization of solar UAVs for telecommunications, surveillance, and observation are not impacted by this loss of aircraft [43].



Figure 2.27: Skydweller [41].

In February 2025, a New Zealand company, Kea Atmosphere, successfully completed a flight to the stratosphere with the Kea Atmos Mk1b. Kea Atmos is smaller than other stratospheric solar UAVs, weighing less than 90 lb_f and a wingspan of just 41 ft. On its first flight to the stratosphere, Kea Atmos reached an altitude of 56 284 ft, in a flight lasting 8 hours and 20 minutes. To minimize weight, it did not have landing gear and was launched from the top of a car, shown in Figure 2.28, and belly landing. The company aimed to achieve months of endurance with applications like maritime surveillance, environmental monitoring, and disaster management [44].

At the time of writing, this chapter summarizes the significant strides and progression of fixed-wing solar-powered flight. Conceptual designs and theoretical projects have not been included in this summary, which contribute to sizing and conceptual design methodology or optimization methods.



Figure 2.28: Kea Atmos [45].

These aircraft, while all fitted with solar cells, shared many common design features. The majority of solar-powered aircraft featured high aspect ratio wings, tending towards glider-like designs. The design processes emphasize maximizing aerodynamic efficiency, reducing the drag of the vehicle to reduce the power required for flight and minimizing the gliding sink rate. This is evident in the LALE solar-powered UAVs with the longest endurances, for which there are minimal differences between the configurations [1, 8, 10, 12, 21, 25, 31–33, 36, 37]. Many of the records that have been set have occurred in summer months where the daylight hours and solar irradiance is more favourable, with the designs focussed on record-setting rather than being designed for practical applications. Emphasis was also placed on using lightweight materials. More expensive and/or recent aircraft used carbon fibre, while balsa wood was a common choice for some of the older and/or less expensive aircraft. Generally, the higher the intended ceiling of the aircraft, the greater the wingspan.

Of these vehicles prioritizing aerodynamic efficiency, there is little published investigation on the tradeoff between aerodynamic efficiency and onboard energy capacity or the influence of payload capacity. Furthermore, it appears that all projects have settled on the glider-like configuration with little analysis justifying it as the best option for long endurance solar-powered missions. To the best of the author’s knowledge, the configuration selection and design process resulting in the selection of glider-like configurations is unpublished, indicating a gap in the proper design analysis of a solar UAV. This research aims to fill that gap, analyzing different configurations, the effects of energy capacity *vs* aerodynamic efficiency, the role of payload, and the influence of time of year and geographical location. Before delving into methods for design and analysis, the following sections elaborate on the challenges of solar-powered flight and the current methods for analysis and optimization.

2.2 Challenges of solar-powered Flight

This section describes some of the challenges in the design, testing, and operation of solar-powered aircraft, as discovered and confirmed through historical flight tests. Although the challenges associated with solar-powered flight considerably overlap between high altitude and low altitude platforms, they are separated into subsections for

the purposes of clarity, distinguishing which challenges apply more specifically to each operating situation.

2.2.1 LALE Solar Aircraft

First and foremost, LALE solar flight is extremely sensitive to weather systems and geographical location. The further away from equatorial latitudes, the intensity of solar irradiation decreases and the daylight hours in a day vary more significantly throughout the year, making perpetual flight more challenging. The time of year has a similar impact on the feasibility of perpetual solar flight, due to more time between sunrise and sunset in summer months and less time in winter months [31, 32]. Local and global solar UAV records have been broken during summer months, proving the intuitive connection between time of year and feasibility of solar-powered perpetual flight [1, 5, 10, 12, 33, 36, 37].

Available solar power is also a function of local atmospheric transparency. Solar-powered flight can be rendered impossible with enough cloud cover or pollution, especially if the aircraft does not have onboard batteries. This has been recorded in testing and flight delays with the Solar Challenger and Solair I, where both platforms did not have onboard batteries to support flight in cloudy conditions [5]. High winds can limit a UAVs ability to station-keep if its maximum velocity is less than the wind speed, reducing the effectiveness of a surveillance or signals-relay type platform. Weather systems have delayed takeoff times for flight tests, delaying or inhibiting the ability of a platform to achieve its mission [5, 36].

Perpetual solar-powered flight is only possible if the net energy balance of the vehicle is not negative in a 24 hour cycle. A net-zero energy balance does not provide any energetic margin for the aircraft, leaving its ability of perpetual solar-powered flight reliant on ideal flight conditions. With any turbulence, unexpected drafts, drops in efficiencies, or cloudy skies, the vehicle would not be able to fly throughout the night. Increasing net energy balance and introducing energy safety margins allows a vehicle to adapt to real flight conditions without compromising its endurance or payload capacity [1, 29, 32].

One strategy to achieve 24 hours of continuous flight is to adjust the takeoff time to early morning. In such an instance, the aircraft does not need to have adequate battery capacity to fly throughout the entire night, only until it reaches the takeoff time the next morning. Local endurance records have been achieved with this strategy; however, those designs would be unable to support perpetual solar-powered flight due to a limited battery capacity that cannot support flight throughout the night [36, 37]. Even with considerate attention to aerodynamic efficiency in design, solar panels and on-board batteries are not always adequate to maintain a net positive energy balance and sustain perpetual flight. Other strategies to reduce expended power to sustain flight include powerless gliding and climbing with thermals. These have been used in record-setting flights and have significantly increased the endurance of solar-powered aircraft [5, 8, 10]. Additional strategies for maximizing endurance will be discussed in Section 2.4.

Reducing weight is a primary consideration when designing any aircraft. In solar-powered aircraft, small variations in total weight can have dramatic effects on the en-

durance and ability to achieve perpetual solar-powered flight. Common materials used for solar UAVs include balsa wood, Kevlar, carbon and carbon fibre, foam, plywood, and clear plastics [1, 5, 25, 30, 31, 33, 36]. It is uncommon for LALE solar UAVs to have metals used for the structure, as wood and composite materials are significantly less dense, offering adequate strength for the weight and applications of the platform.

Another significant challenge is incorporating the solar cells into the vehicle itself. Of the solar aircraft that have been built, very few have not used the upper wing surface for solar cells. The integration of solar cells into the wing of the aircraft is a critical component of design and construction. If the solar cells are too inflexible and the airfoil curvature too great, the solar cells greatly reduce the aerodynamic efficiency of the wing. Some designs have overcome this with custom airfoils to have flatter upper surfaces to accommodate stiff solar cells [5]. The design of the airfoil and its drag characteristics can be the reason a given design is unable to achieve continuous solar-powered flight [46]. The drag polar, stall characteristics, and ability to incorporate solar cells must be carefully considered. Alternatively, others have opted for more flexible solar cells at the expense of solar cell efficiency and/or cost in order to maintain the airfoil curvature [25]. Other designs use plastic or epoxy coverings over the solar cells to adhere them to the wing surface or reshape the upper surface above the solar cells [5, 11, 29, 35]. If selecting a covering over solar cells, the effects of the transparency, absorptivity, reflectivity, and density on the total solar cell efficiency and weight of the aircraft must be considered. Significant reductions in power collection or increases in power draw can occur, decreasing the net energy balance and reducing the endurance of the vehicle [5, 11, 29, 35].

As long-endurance UAVs increase endurance from hours or days to weeks or months, which is typically accompanied by an increase in maximum operating altitude, the reliability and life cycle of parts becomes a limiting factor. Incorporating redundant components in case of failure of the first component will allow the vehicle to remain airborne for longer, despite adding to the weight of the vehicle. This is less of a challenge with LALE vehicles since weather systems and the size of the vehicle generally inhibit such long flights. In the case of the Skydweller, testing of mechanical and electrical components revealed that quadruple redundancy made the aircraft 99% reliable through a 30-day flight [42]. However, projected changes in weight and energy margins as the quadruple redundant system is implemented are not reported.

2.2.2 HALE Solar Aircraft

HALE solar-powered aircraft are not limited by weather systems the same way LALE aircraft are. They operate at much higher altitudes, only facing cloud cover and windy, turbulent conditions during the climb and descent stages of flight. Beyond takeoff and landing, the climb and descent stages are very critical with large risk for failure to occur [4, 9]. The lightweight and large wings are optimal for gliding in the stratosphere, but require idealized weather conditions for climb and descent so they do not disintegrate like Helios in 2003 or Zephyr 8 in 2022. There are few geographical locations with consistently mild weather systems throughout the different atmosphere layers that could support the launching or landing of a HALE pseudo-satellite, limiting the access to the stratosphere for consistent operations. PHASA-35's proven ability to land, reconfigure,

and relaunch in a short time frame was a useful feat, only if it renders the aircraft able to consistently survive the stochastic nature of the atmosphere [39].

Small-scale LALE solar UAVs do not typically have landing gear for weight reduction. They are launched by hand and land in unprepared fields and some have sacrificial material on the underside in case landing in a rougher field is needed. HALE UAVs require faster takeoffs due to the larger wings that cannot be achieved by hand-launching. Zephyr, Skydweller, and PHASA-35 had landing gear that were used for rolling takeoffs, but none were attached to the vehicle. Rather, they remained on the ground so as to not add to the weight of the vehicle [13, 39, 41]. Skydweller relies on technicians riding on electric scooters to keep the wings balanced after touch-down to prevent the lightweight wings from tipping and dragging through terrain. This is a stark contrast to PHASA-35 intentionally belly landing on runways.

Aeroelasticity becomes a greater design challenge the larger and lighter the wings become. Assuming constant material properties along the span, a rectangular wing will have lower torsional stiffness with increasing span, resulting in lower divergence airspeeds with increasing span. The same is true for the bending stiffness of the wing and the relation to stall flutter. The importance of maintaining low structural weights for the overall performance has further effects on limiting the achievable stiffness in the structure. These combined factors decrease the divergence and stall flutter airspeeds and increases the sensitivity of the vehicle to turbulent or windy flight conditions. This greatly impacts the allowable weather conditions for takeoff and landing, having the potential to delay flights and reduce operational effectiveness of fleets providing ISR, cellular networks, or other monitoring tasks. Zephyr currently holds the longest flight record, but emerging companies have the potential to provide platforms with greater payload capacities and therefore a larger range of operational capability [2, 4].

2.3 Sizing and Conceptual Design

There is a distinct difference in sizing and conceptual design for small scale, electric aircraft compared to standard aviation. This section will identify some of these differences and present methods and frameworks developed for designing UAVs.

The maximum weight of the aircraft will determine key characteristics of the design, including the size of the lifting surfaces and the thrust required of the propulsion system. The basis of sizing an aircraft at the beginning stages of conceptual design is highly reliant on historical analysis to develop a relationship between the maximum takeoff weight, W_0 , and the empty weight, W_e , of the aircraft, typically in the form of

$$\frac{W_e}{W_0} = AW_0^C \quad (2.1)$$

where A and C are constants [47]. Conventional sizing of aircraft is based on specified missions with a designated range or endurance with a given payload. These missions are used to generate weight fractions between flight segments and use the calculated fuel burn to determine, in combination with the empty weight fraction, the amount of fuel required for the mission and the maximum takeoff weight of the vehicle to accomplish this mission. Knowing W_0 , historically based empirical equations and constants

can be used to estimate the weights of specific components of an aircraft to be used for centre of gravity and stability calculations to size and locate the empennage and landing gear, which can be found in various aircraft design books such as in [47], which also outlines typical design features for given configurations or aircraft type. The design is continuously iterated, adjusting the weight, lifting surfaces, propulsion systems, onboard electronics, empennage size, and any other detail included in the conceptual design, constantly revisiting or negotiating the requirements from the RFP to produce a conceptual vehicle that satisfies the client's needs. It is important to begin simple conceptual design with multiple configurations to be able to qualitatively and quantitatively compare which configuration is best suited for the desired performance and missions before moving forward with more detailed design [47]. Detailed design has its own methods depending on the system being designed which is not being discussed in this section.

The initial stages of conceptual design of an aircraft is heavily dependent on historical data of similar aircraft. It is more difficult to conceptually design an electric UAV using similar techniques for several reasons. The same sizing method using weight fractions is not applicable to electric vehicles, as there is no change in weight of the vehicle throughout the flight. UAVs have not been developed for as long as fuel-consuming aircraft, therefore there are fewer vehicles to establish historical trends from. UAV is a broad term which includes single rotor, multi rotor, and fixed-wing platforms, as well as some that flap like insects or birds. The broad scope of UAVs and their recent emergence in history makes it difficult to establish generic weight trends for these platforms as each design can vary considerably, including the materials chosen for construction, further increasing the difficulty of establishing useful historical trends.

Typical weight estimates for a given system or component are based on basic geometric characteristics or electrical characteristics of the system being sized. Before the correlation between these characteristics and the weight can be determined, a database has to be established to create a useful range of data that the correlation can be used across. There are several databases of components found in UAVs that can be referenced for creating sizing equations. For example, Tyto Robotics has established a public database of motors, propellers, and Electronic Speed Controllers (ESCs), which contains tested performance measures of each component in the database [48]. APC Propellers has accessible files with geometric properties and performance of their propellers available at [49]. Green *et al.* presents results of tests of various solar cells, reporting their electrical characteristics and efficiencies [50]. The work in Noth's PhD and the development of SkySailor included analysis of a large set of electric motors, which he has been kind enough to share [12]. These databases can be used to establish adequate weight prediction methods for small electric UAVs, which will be discussed in Chapter 3.

The remainder of this section will be discussing established sizing methods for various UAV types. This will include methods specific to propulsion technologies, generic electric and non-electric sizing methods, multi-rotor sizing, fixed-wing sizing, and specifically solar-powered fixed-wing sizing methods.

Alulema *et al.* established and analysed a database of propulsion systems for small scale UAVs, specifically to allow for the sizing of micro, small, and MALE UAVs [51]. The

database includes lithium-based and nickel-based battery cells, Lithium Polymer (LiPo) battery packs, electric ducted fans, electric brushless motors, turbojets, turboprops, and two- and four-stroke piston engines. The resulting correlations allowing for a weight estimation of the selected propulsion system can be embedded into sizing methodologies or optimization scripts to aid in conceptual and preliminary design and analysis of small scale UAVs [51]. This paper intentionally does not include propellers or ESCs in its analysis, stating that these devices are widely available in the public domain, such as in [49], and their weight can be reasonably neglected when compared with the weight of batteries or electric motors [51].

Winslow *et al.* performs a similar analysis, generating correlations for sizing quadrotor propulsion components [52]. This article uses a smaller database than that in [51]. It produces correlations for rotor weight, battery weight, brushless motor weight, brushed motor weight, ESC weights for brushed and brushless motors, and for the airframe weight. The correlations are able to predict the weight of individual components within 10% error and the maximum takeoff weight of the vehicle within 4% error [52].

Several UAV databases have been established, many of which are used to size a variety of UAV types, not the individual components [12, 53, 54]. Noth created a large database of propulsion group components and sailplanes [12]. The database by Hann and Wallisch consists of 93 UAVs, with large variations in weight, span, and propulsion types [53]. Palmer consolidated a database of 875 fixed wing UAVs across a variety of propulsion systems, including 37 solar-powered UAVs [54]. The report by Palmer includes various correlations between aircraft mass, span, length, endurance, thrust, payload weight and type, differentiated for each propulsion type of aircraft [54]. These databases are used in further analyses to develop sizing methods, which will be discussed next.

In his PhD, Noth develops an analytical and continuous design approach for conceptual solar aircraft design. The design method is constructed to be used over a wide range of designs to include manned solar planes, solar UAVs, and solar micro UAVs [12]. Noth's method is not an iterative approach such as the method described in the beginning of this chapter and therefore does not rely on the same weight predictions. Using a database of 415 sailplanes, the weight of the airframe is established as a function of wing area and aspect ratio. Other relationships are developed to predict the mass of solar cells, batteries, and the propulsion group. Combined into a singular analytical method, this work has proved an effective tool for designing solar-powered UAVs. SkySailor, developed by Noth, achieved an endurance record of 27 hours [12]. This design framework has been used for other projects within the same academic institution, with Sun Surfer and AtlantikSolar [1, 6]. AtlantikSolar unofficially holds the record for longest flight for aircraft lighter than 110 lb_f with notable energy margins, further demonstrating the usefulness and success of the design framework [1]. A conceptual solar-powered UAV was designed by Hartney in 2012 using the Noth's design framework, with final remarks supporting solar power as a means to achieve greener aviation practices [55].

A low fidelity sizing method for rotor UAVs by Dreyer and Armanini relies on correlations from Palmer in [54] for the maximum takeoff weight [56]. The sizing method specifically applies to rotor UAVs, with vertical flight and hovering as key aspects of mission profiles used in the sizing method. The method is validated against 20 UAVs

from existing databases, including [53], proving to have mean error of 13.8% in its prediction of maximum takeoff weight [56].

Verstraete *et al.* published sizing correlations from the database and initial analyses found in [54]. The 856 fixed wing aircraft are split into propulsion specific categories: piston, battery, turbine, fuel cell, and solar-powered. The resulting correlations and visualization of the general performance of each propulsion type can be used as a tool to quickly compare and select the type best of UAV for a given application based on desired performance [57]. Although the database is extensive, only 17 of the vehicles are solar-powered. A larger database of solar-powered UAVs should be considered to create more robust correlations for sizing a solar-powered fixed wing UAV.

Leutenegger *et al.* developed a tool for conceptual design of solar aircraft which is very similar to the work found in [12]. It does not estimate structural weight from statistics and historical data. Instead, the design tool executes calculations of feasible, simplified, lightweight structures [58]. The tool also consists of a performance estimation, which considers missions at constant altitude or includes an optimization if the designed aircraft is permitted to climb to store potential energy. The tool considers latitude for performance calculations and concludes that LALE solar flight is only possible in the summer at medium to high latitudes and that latitude contributes significantly to the battery weight of the design [58].

Calculating the performance of an aircraft is a critical aspect of design, especially when using iterative design approaches. Shi *et al.* developed a performance evaluation tool for electric multicopters, allowing the user to input basic characteristics of the UAV, including a wide range of commercially available propulsion components [59]. This tool can be used for rapid comparison between components to select the best combination of motors, propellers, ESCs, and batteries for the desired performance. Similar tools for solar-powered vehicles, fixed wing or otherwise, would prove useful in validating solar-powered UAV design or coupling the performance tool with other sizing or optimization methods.

There are other sizing methods that have been developed that are useful to consider for conceptual UAV design. Kotian *et al.* developed a tool for HALE UAV design for fuel burning platforms, using conventional aircraft design methods [60]. McDonald takes the traditional aircraft design approach and introduces a coefficient to consider an aircraft of decreasing, constant, or increasing weight throughout its mission [61]. This approach expands traditional aircraft sizing methods to include alternative propulsion methods, concluding that further considerations are needed to refine how the approach can be applied to align with real aircraft data [61]. Panagiotou *et al.* develop a sizing tool for hybrid solar MALE UAVs, combining solar-powered propulsion with fuel burning combustion engines. The tool can be adapted to a purely solar-powered platform, but requires further refinements to the structural weight estimation [62].

Alternatively, some research has danced around traditional sizing methods for electric UAVs by adopting optimization methods to pick parameters such as weight or wing loading and basing the remainder of the analysis around these parameters as optimization variables. Colas *et al.* established a framework with aerodynamics, structures, load-

ing and aeroelasticity, propulsion, power collection, stability and control, and weight modules in an optimization scheme for a given mission [63,64]. The maximum takeoff weight is composed of results from the other modules, seeking to be minimized without violating design constraints. The optimization scheme is applied to several HALE UAV configurations: a conventional empennage and twin boom design in [63] and a flying wing in [64]. Retana and Rodriguez-Cortes developed an optimization method to maximize the endurance of a flying wing as a means of sizing the aircraft [65]. This method considers fewer design variables, does not fully define the geometry of the flying wing, nor does it consider aspects of higher level aircraft design [65].

2.4 Optimizing Solar UAVs

Sizing an aircraft is only the first step of design. Further analysis is required to design the wing, propulsion system, empennage, landing gear, or any other onboard system or geometrical features. An iterative design process within a design team will gradually evolve the aircraft into a more optimized design, but may never reach a true optimized form, depending on the objective of the aircraft. To achieve maximum performance, minimum weight, or any other desired criteria, optimization methods can be used in a variety of ways to achieve specific design goals.

A primary objective of solar-powered aircraft is to prolong endurance, often aiming for perpetual flight. There are various means to achieve this. The physical characteristics of the aircraft will affect the aerodynamic characteristics and therefore the endurance. Flight conditions including flight speed, altitude, latitude, and flight path can drastically affect the power consumption of the aircraft which also impacts the endurance. There are numerous methods that have been used to optimize solar UAVs. They are presented in this section, categorized into how the aircraft is flown (path planning and trajectory optimization) and physical characteristics of the aircraft (design optimization).

2.4.1 Path Planning and Trajectory Optimization

There are various flight techniques used by pilots to reduce power output, extract energy from the airflow, and prolong flight. Perpetual flight has been envisioned for at least as long as solar flight has been possible [5]. Strategies for prolonging solar flight were considered with the design of the first solar aircraft, which have been a topic of research ever since.

Gliders and soaring birds alike take advantage of naturally occurring updrafts to climb to increase their overall energy state. Solair I, Sunseeker, and SoLong specifically report gaining altitude using thermals to increase their glide time and overall flight time [5,6,10]. Connerty establishes a thermal model and optimizes trajectories of an autonomous UAV by maximizing the total kinetic and potential energy of the aircraft for differing thermal strengths [66]. These results are not specific to solar-powered aircraft, but provide insight to the value of thermal soaring and extracting energy from naturally occurring phenomena. Wu *et al.* considers a solar UAV soaring with thermals at various times of day with different flight speeds. Results show that slower flight speeds reduce the power draw from onboard battery storage, allowing the aircraft to glide for longer and store more solar energy, prolonging flight time [67]. One cycle of

climbing 100 m in a thermal and gliding to the original altitude is able to store the most electric energy at solar noon compared to other times of day, showing that solar power in combination with thermal soaring is able to increase endurance [67].

Edwards *et al.* maximizes the net power in circular trajectories for solar UAVs [68]. The bank angle, and therefore turn radius, for circular trajectories is optimized depending on the incident angle of the sun. For low sun elevation angles near sunrise and sunset, tighter turns with a high bank angle allow the most solar energy to be stored. For high sun elevation angles, larger turns with a low bank angle are more optimal for solar energy collection [68]. To gain potential energy from thermal updrafts, a UAV would have to deviate from its optimized bank angle for solar energy collection to remain within the influence of the thermal. Results from [68] shows that it is energetically beneficial to gain total energy by using available thermals to climb then resume the optimal solar collection bank angle rather than relying purely on solar energy collection.

A similar approach to extracting energy from thermal wind gradients is found in [69] and [70]. Rather than using the vertical updrafts found in thermals, these approaches aim to extract energy from horizontal wind gradients, relatively close to the earth's surface. Bower *et al.* compares optimized trajectories for two soaring UAVs, one of which includes solar power for onboard systems or secondary means of propulsion, aiming to minimize the wind speed required for soaring [69]. Using solar power to charge a battery and power an electric motor, the solar UAV is able to reduce the minimum wind speed required for soaring by 40% compared to the non-solar UAV. The resulting trajectories show the solar UAV remaining more constant in bank angle, opting for the collection of solar power rather than relying exclusively on energy extraction from the wind gradient [69]. The stochastic nature of wind is a challenge for using optimized soaring trajectories in real flights. Trajectory optimization is too computationally expensive to be executed in real time onboard aircraft. To overcome this, Kim effectively optimizes a neurocontroller for autonomous soaring [70]. This differs from trajectory optimization as the path of the aircraft is not optimized, but the connections of the neurocontroller are evolved to respond to the vehicles position and acceleration. Theoretically, the simple neurocontroller can be embedded into an autopilot and would be capable of adapting to real time variations in wind profiles and maintain soaring [70]. Details of this optimization approach applied to a solar UAV will be discussed in Chapter 6.

Thermals are used to gain altitude with minimal or no power expenditure, allowing flight time to be extended with the subsequent glide phase. Another strategy to extending solar-powered flight is to store as much energy as possible in potential energy during daylight hours, allowing the aircraft to glide throughout the night with less or no power output without the use of thermals. Sachs *et al.* proposes that a solar UAV could fly perpetually without batteries onboard by climbing to high altitudes during the day and gliding throughout the night [71]. Without batteries, the modelled UAV would need to climb from approximately 4000 ft to approximately 70 000 ft during the day [71]. This model does not fully consider the challenges of designing an aircraft suitable for flight throughout the entirety of the troposphere into the stratosphere, as discussed in Section 2.2.2, claiming this method to be reasonable as a means to achieve perpetual flight [71].

Wu *et al.* proposes two energy strategies for solar-powered long endurance tracking of a target [72]. Both strategies include a scheduled climb to higher altitude, a non-powered glide, a powered glide, and cruising at a minimum altitude. The problem is formulated with a lower altitude limit of 50 000 ft and an upper limit of 98 000 ft. The first strategy uses a slow and steady climb until the energy balance becomes negative at sunset. The second strategy climbs quickly, then maintains a high altitude to recharge batteries until the energy balance becomes negative. After sunset, both strategies glide unpowered, then powered, then hold a low altitude through to the morning until enough charge can be gained to repeat the cycle. This formulation is specific to a target tracking application. The objective function includes a target tracking term which requires a specified target to always be within the detection range of the sensor and is dependent on the altitude of the aircraft. The unpredictability of a moving target makes it difficult to employ these results in real time, which is a similar challenge faced by dynamic soaring with wind gradients.

Wang *et al.* develops a model to maximize the final State Of Charge (SOC) after a 24 h flight for a HALE platform [73]. The optimized results are compared to a base case at different latitudes and times of year. The optimized strategy has the greatest gains at higher latitudes and further away from the summer solstice. Energy savings are found both in the non-ellipsoid trajectory as well as the gain in altitude to allow for non-powered glide. It was found that there are more energy profits by charging the batteries at a lower altitude before climbing than by climbing earlier and charging at a higher altitude.

In [74], Hosseini *et al.* also aim to maximize the final battery SOC after 24 hours of flight. The 24 hour flight is discretized into 24 points, using position, velocity, heading, flight path angle, power consumption, and SOC as the states to find optimum points for. A reduced model of the direct optimization method produces similar trajectory results and solves in less than half the time [74]. The reduced method is used in future work, optimizing the path of a surveillance solar UAV to maximize the coverage area of the sensor, where there is a trade off between altitude gain and sensor coverage [75]. Less altitude gain reduces the ability to maintain energetic safety margins in flight by shortening the unpowered glide portion of the trajectory. Path planning with a stronger emphasis on communications relay and maintaining line of sight is discussed by Padilla *et al.* [76]. With tighter constraints on altitude and geographical position, the modelled UAV is forced to adopt switch backing patterns to maximize solar energy capture rather than capitalizing on extended unpowered glide phases [76].

Similar to [72], Gao *et al.* develops an energy management strategy for the climb and descent phases of flight, aiming to reduce power consumption of a HALE solar UAV. A particle swarm optimizer is used to establish a climb speed schedule and glide power schedule for the respective phases of flight from 16 500 ft to 65 600 ft, or vice versa. The resulting climb schedule results in 68% energy savings compared to a base case and the glide power schedule results in 4.8% energy savings. The energy savings in the glide phase are accompanied by nearly 30 min of increased glide time [77]. The flight duration of the baseline or optimized climb phase is not specified.

Another approach is to consider the horizontal path of the UAV more closely, similarly to the results in [76]. Powers and Lozano-Duran find optimal trajectories at solar noon and at sunset of a small scale solar UAV [78]. Multiple solutions are found for different seed cases using a Sparse Nonlinear OPTimizer, which solves for local optima based on the initial state. When the average heading is constrained, all solutions improve the steady state energy gain rate by a factor of two [78]. The resulting optimized trajectories include straight flight, spiraling, and switch-backing. When forced to traverse parallel to incident sunset rays, the optimal trajectory is to adopt a snaking pattern [78]. These trajectories provide insight to optimal flight strategies at low altitudes that do not include drastic changes in altitude to benefit from an unpowered glide. Several projects have been concerned with maximizing energetic safety margins, including those in [1, 21, 79], which ultimately also seek to maximize the steady state energy gain rate, even if not explicitly stated.

The path planning and trajectory optimization for solar UAVs is useful in determining trends or strategies for prolonging endurance. It is difficult to optimize useful trajectories for real flight scenarios where wind and cloud cover are unpredictable and require real-time adaptation to the flight path to ensure minimum power expenditure. Additionally, discretizing a 24 h or longer flight into sufficiently small time steps to allow an autopilot to achieve the same path and states is very computationally expensive. Neural networks as a controller provide an interesting alternative to path planning, as the updated state of the aircraft can be used to drive controls with a wide range of applicable objectives in real time. Ni *et al.* explores the use of a network trained via reinforcement learning to develop a controller capable of maintaining HALE flight [80]. The trained network results in a 28% increase in the SOC of the modelled UAV after a 24 h flight and a 39 day continuous flight, which is 50% longer than the base case [80]. The resulting network consists of hundreds of nodes and several layers between the inputs and the actions. There are no proposals as to how this control could be implemented for real flight controls, which would be difficult considering the complexity of the network.

Another method of using neural networks for controls is established in [70], which has already been introduced for dynamic soaring applications. Kim uses a method of evolving neural networks and their connections called the NeuroEvolution of Augmenting Topologies (NEAT) algorithm. Although not a solar application, the implementation of a neural network as software-in-the-loop highlights both the simplicity and the feasibility of evolved neural networks as a controller for stochastic flight conditions [70]. Further details of the algorithm are included below. It is used as a means of trajectory planning in Chapter 6.

NEAT for Trajectory Planning

The NEAT algorithm was developed by Stanley and Miikkulainen [81]. It is a genetic algorithm for evolving neural networks, relying on the principle of Darwinian fitness to propagate successful traits throughout a population. The algorithm uses an encoding scheme to write the characteristics of a network in a manner akin to genetic data. The encoding scheme of the genome is shown in Figure 2.29. The nodes in a network can be sensor or input nodes, output nodes, or hidden nodes. The encoding indicates which

nodes are connected, the weight of the connection, whether the connection is enabled or disabled, and the innovation number which indicates the point in the evolutionary sequence the connection was formed [70,81].

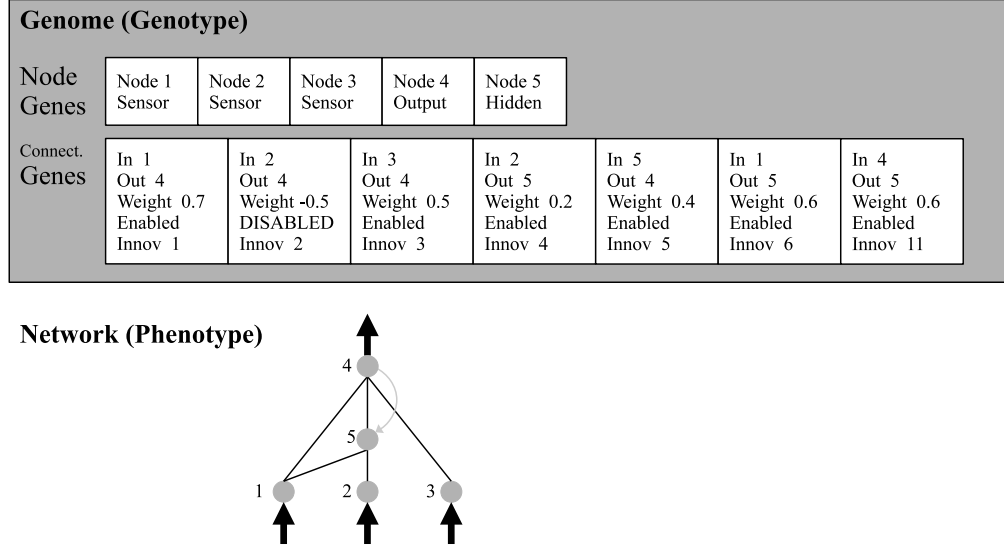


Figure 2.29: NEAT encoding scheme [81].

The algorithm randomly generates a population whose members consist of nodes and connections. The members of the population are categorized by similar network structures and shapes into species. The fitness of a member is evaluated through its interaction and performance with a defined objective function, then are ranked based on their fitness within their respective species. For the sake of trajectory planning, the fitness may be defined as distance travelled, flight time, or fuel consumed while enforcing certain boundary conditions through a penalizing contribution to the fitness value. Members with the worst fitness are terminated and the remaining members are selected for reproduction or undergo mutation. Members chosen for reproduction produce offspring via crossover, where the gene segments are randomly spliced and concatenated. Segments that do not undergo crossover are inherited from the parent with the better fitness. After the crossover, mutations have a chance to occur. Mutation processes have various operations which include the random addition or subtraction of connections or nodes to the network. The crossover and mutation of networks allows for unique and innovative networks to form while preserving characteristics associated with high performance [70].

One crossover and mutation of the population is considered one generation. The reproduction and mutation of members is repeated for several generations. The species are compared to one another through stagnation. The stagnation of a species is the number of generations that the highest fitness of that species does not increase. Once a species has stagnated for a set number of generations, it is culled from the population. An elitist mechanism preserves some of the highest performing members of that species, regardless of stagnation, allowing the best characteristics to propagate further through the population if they contribute to higher performance. This entire process is managed

by the NEAT algorithm, which populates each generation, evaluates the fitness of each member, and outputs the best performing network after a fitness threshold is reached or a set number of generations have passed. The algorithm evolves the network in a manner that improves the network as a controller in addition to the trajectory, which is highly dependent on the definition of the fitness function. Further details and a more in-depth explanation of NEAT can be found in [70, 81].

2.4.2 Design Optimization

The other aspect of optimizing a solar UAV for extending endurance is to optimize the vehicle itself. The presented literature in this section focusses on intentionally optimized conceptual designs, not designs that have been iterated and gradually optimized through flight testing campaigns.

Extending the endurance of a solar UAV is typically attempted by optimizing the aerodynamic characteristics of the vehicle. Aerodynamic efficiency of an aircraft is primarily dictated by the wing. Kontogiannis and Ekaterinaris optimize the wing of a surveillance UAV to maximize aerodynamic efficiency with the underlying objective of decreasing power consumption, increasing endurance, and improving flight performance [82]. Although not a solar-powered platform, the authors conclude that minor design adjustments could allow the optimized platform to be powered by solar energy, reaping further benefits of increasing the endurance for surveillance purposes.

Another non-solar-powered UAV wing optimization is found in [83]. Wei and Masood develop an aerodynamic analysis method for drag prediction without the use of CFD. The tool is used to optimize the wing of a UAV with airfoil characteristics, chord lengths, span, sweep, and dihedral as design variables of a cranked wing. Although the optimization is successful in reducing the average drag at multiple angles of attack, it is recognized that the optimal wing design cannot be determined without integrating into the full vehicle and analyzing the other disciplines of aircraft design [83].

A study supporting the further development of the CREATEv platform optimizes a rectangular wing planform with a newly developed airfoil for the project [46]. Design variables include wing span, chord length, and battery mass. The solar cell area is treated as a continuous parameter to allow for simpler optimizers to be employed. An upper constraint on aspect ratio has a large influence on the results, indicating that a realistic optimum may exist beyond this constraint but cannot be modelled due to limitations in the aerodynamic model employed in the analysis. This optimization accounts for increases in battery mass but does not include multidisciplinary interactions of the overall vehicle, particularly with limitations in structural modelling and solar cell discretization [46].

Colas *et al.* develops a more complete multidisciplinary design optimization (MDO) for different configurations of solar UAVs, which has already been introduced in Section 2.3 [63, 64]. The MDO framework is used both as a sizing and optimization tool, seeking to minimize weight and maximize the latitudes where perpetual flight is feasible. The framework offers a more complete optimization than those presented in [83] or [46] due to the programmed interactions between aerodynamics, structures, loads

and aeroelastic effects, propulsion, power collection and storage, stability and control, weights, and mission segments [63].

Gates *et al.* combines trajectory and vehicle optimization to optimize a solar-powered HALE platform [84]. The battery mass (and therefore energy storage capacity), trajectory, and propulsion system are all optimized independently in an iterative process, in this order, aiming to maximize the UAV's total energy. After 7 iterations, the total energy was increased by 20.7%, which effectively means a 2.4 h increase in flight time. The iterative optimization was able to achieve more than three times the increase in total energy than by optimizing the trajectory or propulsion system alone [84]. Although there are recognized improvements to be made to this analysis, it supports the need for a strongly integrated design and optimization approach to achieve the most optimal design.

2.5 Solar UAV Design Considerations

Sizing techniques and optimization methods are useful and necessary for the successful design of a solar LALE UAV. It is difficult to interpret optimization results or make design decisions without an understanding of the sensitivity of the design to a given variable. There are several important design considerations for solar LALE UAVs, which will be presented alongside design trade studies in this section. A brief summary of current materials and technologies is also included.

A simple conceptual design in Meyer *et al.* affirms the primary design and operating considerations for feasible solar-powered LALE long endurance flight [85]. Meyer *et al.* presents that the available solar power is the largest design influence, which varies with geographical location, time of year, atmospheric absorption, and solar cell efficiency. Without sufficient solar power, a solar-powered vehicle would be unable to achieve prolonged flight. The design of the wing is the second major consideration, which should be designed for high aerodynamic efficiency at low air speeds to ensure the best possible minimum thrust condition [85]. The onboard energy storage must be sufficient to provide the required thrust throughout night time operation until the batteries are able to be recharged for the next day-night cycle.

A parametric study based on the CREATEV platform demonstrated the effects of span, time of year, and battery mass on the endurance of a solar LALE UAV. Rahaman-Noronha *et al.* demonstrated the differences in best designs depending on the available solar irradiance. Regardless of time of year, a smaller aircraft with a large battery capacity is the best for increasing endurance [86]. Time of year dictates the optimal wingspan for achieving the best range. With the maximum number of batteries considered in the study, a wingspan of approximately 13 ft and 8 ft achieve the best range on the summer and winter solstices, respectively [86]. The results of this study allowed a battery and wing design to be selected to find a middle ground for operating conditions, creating operational effectiveness and increasing applications for a long endurance platform.

The work in Bower *et al.* included a sensitivity analysis of the design variables on the minimum required reference velocity for dynamic soaring. This analysis was not for the

solar UAV considered in the study; however, the sensitivity of the results are insightful for solar designs, especially if dynamic soaring is considered as a means of extracting energy to prolong flight. Generally, a larger span, lower surface area (equating to higher aspect ratio), lower weight, lower parasitic drag, and higher aerodynamic efficiency are beneficial for dynamic soaring [69]. These trends are in agreement with results of other analyses on solar UAVs.

Haider performed a parametric sweep of wingspan, aspect ratio, payload weight, payload power, cruise altitude, reduction of solar irradiation due to cloud cover, solar cell efficiency, and battery energy density for a T tail glider configuration solar UAV [87]. Results showed optimum values for minimizing weight of the UAV through each sweep. Weight of the UAV decreased with increasing aspect ratio up to 18 for the modelled UAV. Generally, span should be minimized for a given aspect ratio to have smaller, structurally lighter wings. Increasing payload mass and power consumption directly correspond to an increase in vehicle weight, as more power is required to maintain flight, which increases the required battery capacity. Decreases in collected solar energy due to cloud cover or solar cell efficiency increase the UAV weight. Increases in battery energy density decreases the UAV weight as fewer battery cells are required to provide the same total capacity [87]. The sensitivity of each variable on the UAV weight was subject to the baseline configuration and the fidelity of the model used to calculate the weight of each design point. Results for other vehicles would replicate similar trends, but likely have differences in exact values for optimums.

Dinca *et al.* more aggressively explored the extremes of designing a solar UAV, asking if aerodynamic efficiency or electrical power was the better property to optimize [88]. A traditional glider-like solar UAV configuration was compared to a flattened, wider configuration with more than triple the number of solar cells, both with the same span. The flattened configuration was heavier, consumes three times the power for its payload and propulsion, and had 67% of the range of the glider. There was no conclusive answer as to which configuration was inherently better as neither design was optimized. The primary conclusion was that trading aerodynamic efficiency for electrical power is an interesting trade off which should be further explored for future solar-powered UAVs [88]. The optimal vehicle may be dependent on the intended applications and desired payload.

The launch and landing methods for a UAV dictates geographical operating zones. Hand-launched platforms only require adequate clearing to climb to altitude, whereas platforms requiring a rolling takeoff need a suitable runway as well. Regardless of the operating altitude, permanently installed landing gear adds weight to the vehicle and reduces flight time or energetic margins. The ability to hand launch certainly influences a design towards smaller configurations when a variety of weight and span combinations could be feasible for long endurance flight [46, 88]. LALE UAVs are generally lower in aspect ratio and do not typically face aeroelastic concerns to the same extent as previously discussed with HALE UAVs. Implementing a limit on the aspect ratio or wingspan for ground handling considerations also benefits aeroelastic considerations, resulting in less fragile designs.

2.5.1 Current Technologies

This section briefly addresses some of the current technologies applicable to solar-powered flight. This thesis is not intended to analyze the electrical components of a solar UAV in detail, therefore this section provides an overview of the technologies recently employed in fixed wing solar-powered flight.

Several studies and aircraft presented in this review stress the importance of reducing the overall weight of the aircraft to reduce power consumption and to increase endurance. The first solar aircraft were commonly constructed with spruce, balsa wood, mylar sheets, and kevlar [5]. Balsa wood remains a common choice for lightweight wing and fuselage structures in several designs [1, 11, 25, 30, 31, 36]. Carbon fibre and composite materials are used in more recent platforms for their high strength and low weight properties despite the higher costs [1, 11, 21, 25, 30]. Fused deposition modelling, an additive manufacturing method commonly referred to as “3D printing”, has largely grown in popularity and practice for the development of multi-rotor and fixed wing UAVs in both commercial and hobbyist applications [89, 90]. Despite the weight savings associated with 3D printing, it has yet to be implemented in a fixed wing solar UAV. García-Gascón *et al.* proposed a specific method of structural buildup for applications in a solar UAV but did not progress beyond the conceptual stage [91]. To the best of the author’s knowledge, there have not been any fixed wing solar UAVs that have been built or tested using additive manufacturing.

The energetic density of batteries is a crucial parameter to consider when selecting energy storage for any aerial vehicle, especially those seeking long endurance. Improvements in battery technology have directly contributed to the longer flight times that have been achieved by solar-powered aircraft. LiPo and Lithium-Ion (Li-ion) batteries are popular for UAV energy storage due to their higher energy and power densities compared to alkaline, lead, or nickel alternatives [92]. Batemo has an extensive library with easy-to-use comparison features between selected cells [93].

The high cost and low efficiency of solar cells made it difficult for solar-powered flight to be researched or implemented. Some of the earlier projects were only successful because they reused solar cells from previously built solar aircraft [5]. Although solar cells are not useful overnight or in overcast conditions, the increase in efficiency and decrease in cost has allowed solar-powered flight to be explored in both large scale and small scale projects. Green *et al.* provides a summary of tested solar cells. Across various configurations and various types of photovoltaic cells, the efficiency ranges from $8.8 \pm 0.3\%$ to $47.1 \pm 2.6\%$ [50]. Typical efficiencies of solar cells used for solar LALE UAVs falls within the range of 20-26% [1, 11, 25, 31, 32, 35].

Maximum Power Point Trackers (MPPTs) are used to convert and supply correct voltages from the solar array to the battery packs [35]. MPPTs are selected by the electrical specifications of the solar array and batteries, with greater efficiencies being preferred to not lose more power through the solar to electrical power conversion.

Autopilots are crucial for long endurance flight, especially overnight. Several studies and flight techniques show that a circular trajectory, or loitering, overnight is a useful

low power mode when there is no solar power available. Autonomous modes for loitering or following pre-planned trajectories can reduce unnecessary power consumption from flight control activation that increases trim drag and also reduce the strain of the ground crew in charge of the flight.

As with all aviation, it is important to reduce weight without sacrificing structural integrity of an aircraft as much as possible to achieve the best performance. Materials and all onboard technologies should be chosen with careful considerations of the weight, power consumption, and efficiency.

2.6 Knowledge Gap

Current research on solar UAVs is targetted at increasing endurance, either to prolong electric flight or to achieve perpetual solar-powered flight. Many of the built and tested vehicles that aimed for perpetual flight were inspired by previous successful solar UAVs and opted for very similar design configurations. Conceptual and real design analyses do not compare other aircraft configurations to determine the most promising configuration for LALE flight. There is limited substantiation that a rectangular, high aspect ratio wing with a thin and long fuselage is the most promising design for a LALE solar UAV. Some of the optimization analyses and parametric studies consider basic variations to these configurations yet also neglect to analyze different aircraft configurations.

Some of the vehicles intentionally designed with larger energetic margins are motivated by increasing the robustness to unpredictable disturbances in flight including wind and cloud cover [1,21,29,32,42]. The allowable increases in payload, either in weight or power or both, are not considered as reasons to increase the energetic margins. Many projects are focussed on the realization of perpetual flight or record setting and do not consider initially designing the UAV for practical applications. There are many realized applications of LALE UAVs which have yet to be the design motivator for solar LALE UAVs.

Minimizing weight is a key aspect of designing UAVs. 3D printing has largely grown in popularity, practicality, and application for aviation given its lightweight properties and ability to produce complex geometries [89]. None of the solar-powered aircraft in this summary have been manufactured primarily out of 3D printed materials, which has significant potential for reducing weight even further than previously produced solar UAVs, allowing a further increase of energetic margins to extending the endurance.

3 Tools and Methods

This chapter defines the tools and methods used for analysis throughout this research. The main resource for the conceptual design and analysis of UAVs used in this research is the object-oriented Aircraft Conceptual Design Toolbox (*pyACDT*) which is written in Python [94]. This chapter details the models and methods used throughout the analyses in Chapters 4, 5, and 6, detailing where each chapter uses a different method when applicable. These methods include a geometry configurator, a weight prediction model, specific payloads considered, aerodynamic performance, a point mass model, a solar model, and an energy model. The geometry configurator and weight prediction models presented in this chapter have been designed specifically for this project and are detailed in Sections 3.1 and 3.2, respectively.

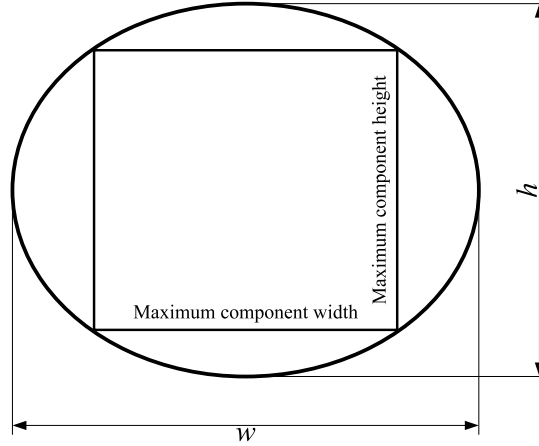
3.1 UAV Geometry Configuration

A geometry configuration tool was developed within the structure of *pyACDT* to generate the full geometry of a small fixed wing UAV. This tool takes high-level design parameters such as wing span, chord length, the number of batteries, the number and type of propellers, and the empennage configuration and returns a geometry object compatible with *pyACDT*. This tool allows for the quick generation of vehicles for rapid analysis, which is particularly useful for the parametric and optimization analyses within this research. Figure 3.1 shows some sample outputs of the configurator.

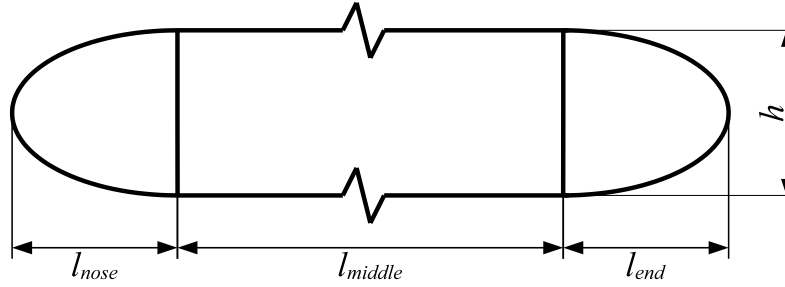


Figure 3.1: Sample geometries from the UAV configurator.

The fuselage and nacelle bodies are constructed using the same method. The internal volume of the body must be able to accommodate the volume of hardware and payload allocated to that body and the specific dimensions of those components. The frontal cross-section of the fuselage and nacelles are super-ellipsoids. The height and width are constructed to minimize the empty space around the largest dimensions of the internal components. Each body consists of a nose, middle, and end segment. Sketches of these segments are shown in Figure 3.2, which are not to scale.



(a) Cross section.



(b) Side view.

Figure 3.2: Fuselage and nacelle geometry, not to scale.

Furthermore, the curve that defines the xy cross-section is governed by Equation 3.1, where $2a$ is the total width of the body, $2b$ is the total height, and e_1 is the squareness factor.

$$1 = \left| \frac{x}{a} \right|^{e_1} + \left| \frac{y}{b} \right|^{e_1} \quad (3.1)$$

$$a = \frac{x_1}{\frac{|\cos \theta|}{\cos \theta} \cdot |\cos \theta|^{e_1}} \quad (3.2)$$

$$b = \frac{y_1}{\frac{|\sin \theta|}{\sin \theta} \cdot |\sin \theta|^{e_1}} \quad (3.3)$$

Figure 3.3 shows the fuselage cross-section minimized around an internal component with height h_c and width w_c for different e_1 . When $e_1 < 1$, the cross-section is more square. When $e_1 = 1$, the cross-section is an ellipse. The angle θ is used to define a and b of the super-ellipsoid. Physically, θ is the angle on the unit circle used to define where point $(w/2, h/2)$ intersects the super-ellipsoid if the unit circle was scaled to become the super-ellipsoid. As θ approaches $\frac{\pi}{2}$, the width of the cross-section approaches ∞ ; as θ approaches 0, the height of the cross-section approaches ∞ . Where the middle

sections have straight top and bottom surfaces in a side view, $\theta = \frac{\pi}{2}$, effectively forming a tube, relying on the fore and aft segments to define further curvature to the body. In the configurator, θ is solved to minimize the area of the cross-section that is not occupied by the internal components, simultaneously defining a default height to width ratio of the body. If the height to width ratio is defined as an input variable, the height or width will be scaled after θ is optimized and the point $(w/2, h/2)$ will no longer be intersecting the surface.

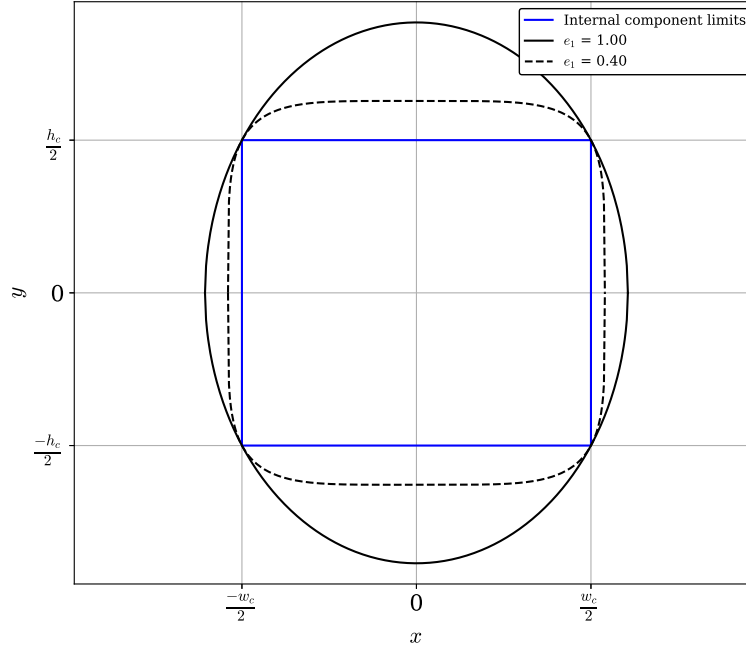


Figure 3.3: Fuselage cross section before specifying height to width ratio.

Referring back to the anatomy of the body segments in Figure 3.2b, the components within the body are stored in the middle section. The length of the middle section is the minimum of the wing root chord length or the summed length of the internal components of that body. If the length of the nose or end sections are not specified, then the length is the average of the height and width of the total cross-section.

The internal components can include an autopilot, batteries, or MPPTs. In Chapter 4, one battery pack is assumed to be a cube, where each dimension is equal in length. A battery with more cells will have larger dimensions, resulting in a larger fuselage or nacelle cross-section. In Chapter 5, the batteries are assumed to be packed as a rectangular volume where the width is defined by the number of cells in series and the height and length are defined by the dimension that is more streamline. Figure 3.4 shows samples of the battery arrangement where the flow direction indicated is along the geometric y -body-axis.

In Chapter 4, $e_1 = 1$ for all vehicles, and the height to width ratios and nose and end lengths of the fuselage and nacelles are unspecified.

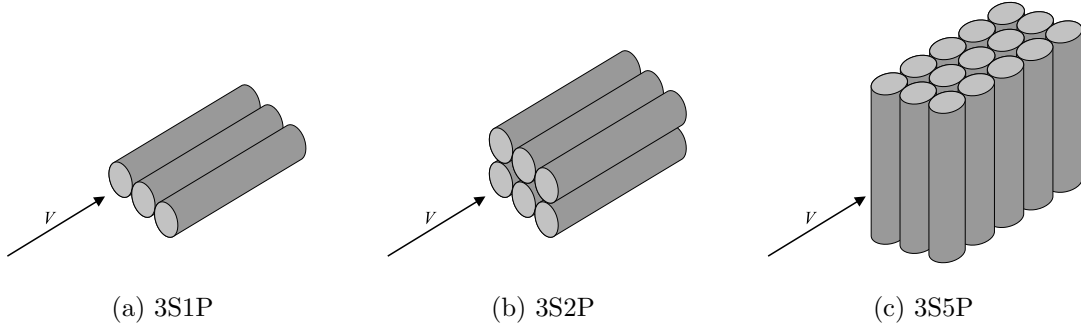


Figure 3.4: Different sized battery packs streamlined with the flow, not to scale.

In Chapter 5, e_1 , the height to width ratios for the fuselage and nacelles, and the nose length of the fuselage are used as design variables in the optimization. The end section of the fuselage is removed and the tail shape is merged from the shape of the aft middle section to a circular cross-section at the aft of the tail, with a radius defined by an optimization variable.

3.2 Weight Prediction Model

Traditional aircraft sizing methods are well established and rooted in historical trends and analysis of existing aircraft. The weight of structures, electronics, avionics, landing gear, flight control systems, environmental controls, and other components can be estimated for new designs based on the expected scale and function of the aircraft from historical correlations [47]. The initial conceptual design and sizing of small UAVs is more difficult; UAV is a broad term consisting of various configurations ranging from single- to multi-rotor to fixed wing to flapping wing platforms. There is no one-size-fits-all design approach, as historical UAVs are not documented in the same way as more-traditional manned aircraft and many are experimental or recreational with a relatively recent emergence in history, with respect to powered flight.

This section presents the development of methods for sizing the weights of various components used in UAVs. The equations for electrical and propulsive components are based on empirical relationships from databases provided by Noth and data from APC Propellers [12, 49]. This section contains sizing methods for Brushless DC (BLDC) motors, ESCs, propellers, and wing structural weight. The resulting correlations provide initial estimates on mass and basic dimensions where applicable. These correlations are incorporated into *pyACDT*, and are used to calculate the weight of a UAV in Chapters 4 and 5.

3.2.1 Propulsion Component Sizing Methodology

The resulting correlations are best-fit as power functions, as in Equation 3.4. A is a constant, X is an n -vector of design variables, P is an n -vector of exponents, there are n input variables, and Y is the output.

$$Y(X, A, P) = A \times \prod_{i=1}^n X(i)^{P(i)} \quad (3.4)$$

A and P are solved using the sum of least squares, which is a minimization of Equation 3.5, where Z is the dataset being modelled, and each $Z(i)$ corresponds to a set of $X(i)$, for N data points.

$$S = \sum_{i=1}^N [Z(i) - Y(X(i), A, P)]^2 \quad (3.5)$$

3.2.2 Propulsion Component Sizing Correlations

This section presents the resulting weight and geometric correlations for BLDC motors, ESCs, and propellers. For each component analyzed, there are multiple variables that can be used as inputs. Various combinations of input variables have been tested. The correlation with the best R^2 adjusted for the distribution of data points is presented for each relevant output for each component.

BLDC Motors

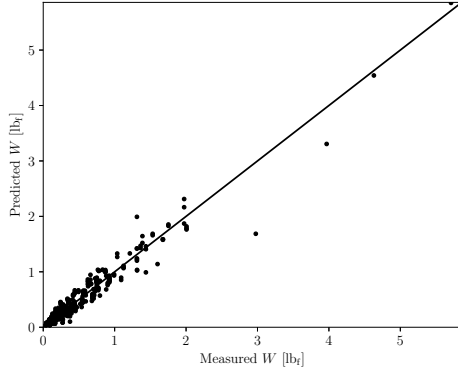
The performance of BLDC motors are generally characterized by electrical parameters which are readily available in specification sheets. The data used for the BLDC motor correlations were provided upon request from the database developed in [12]. The commonly available parameters available for BLDC motors are the KV constant, K_V measured in $\frac{\text{rpm}}{\text{V}}$; no load current, I_0 measured in amps; resistance, R measured in ohms; maximum continuous power, P_{max} measured in watts; and maximum continuous current, I_{max} measured in amps. Combinations of these five variables were considered as inputs for correlations for weight, diameter, length, and volume of BLDC motors. The equations and statistical analyses for the weight, diameter, and length are presented below. The correlations for the volume are presented in Appendix B.

The weight in lb_f of BLDC motors can be predicted using Equation 3.6. Resistance is not included in this correlation as it did not improve the statistical R^2 or error distribution from Figure 3.5c. For this correlation, $N = 563$, $RMSE = 0.1170$, and Adjusted $R^2 = 0.9453$, where RMSE is the Root Mean Square Error.

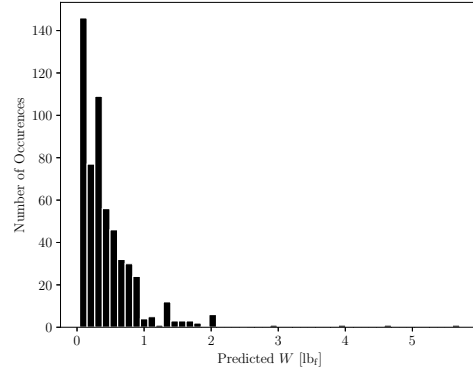
$$W [\text{lb}_f] = 8.0169 \times 10^{-3} K_V^{-0.14185} I_0^{0.24489} P_{max}^{0.94881} I_{max}^{-0.36684} \quad (3.6)$$

The entire database is used for the formulation of the empirical models, therefore any deviations from the model are modelling errors and not uncertainties. The modelling errors, or the extent to which the model does not correctly predict the property of a point within the dataset, is herein referred to as the error. Further statistical validation for the BLDC weight prediction is presented in Figure 3.5, where Figure 3.5c requires some explanation to understand its significance. The relative error was evaluated for every N data points. They are sorted from most negative to most positive and attributed a rank from $\frac{-N}{2}$ to $\frac{N}{2}$. The standard deviation is also determined for all the errors. A deviation based on the ranking and the standard deviation of the errors is attributed

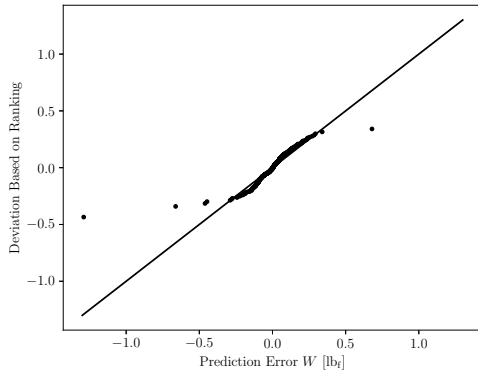
to each ranked error using an inverse normal distribution function. If the data has a normal distribution, a plot of the deviation based on ranking *vs* the relative error should be a straight line with a slope of 1 and intercept of 0, which is the solid line shown in the figure. The closer the errors follow a normal distribution, which is visualized in Figure 3.5d, the closer the data in Figure 3.5c will follow the straight line. These plots are shown for every correlation to allow a visual assessment of the applicability of each of the correlations.



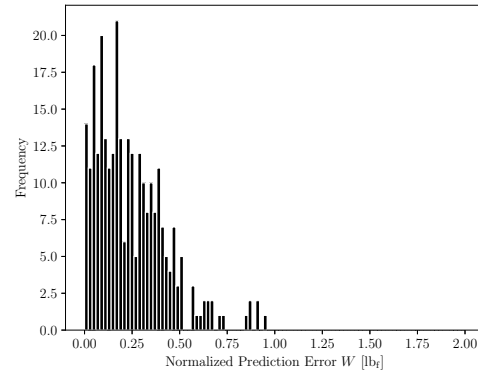
(a) Comparison of predicted to measured weight.



(b) Distribution of predicted mass.



(c) Statistical distribution of errors.



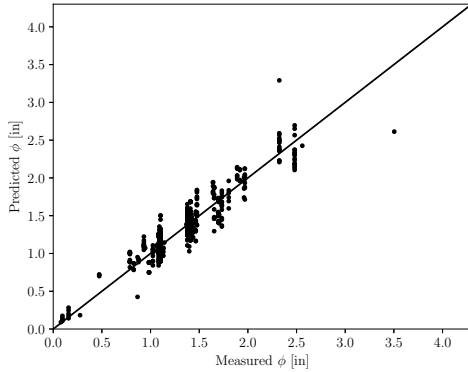
(d) Frequency distribution of errors.

Figure 3.5: Visualization of BLDC motor weight prediction.

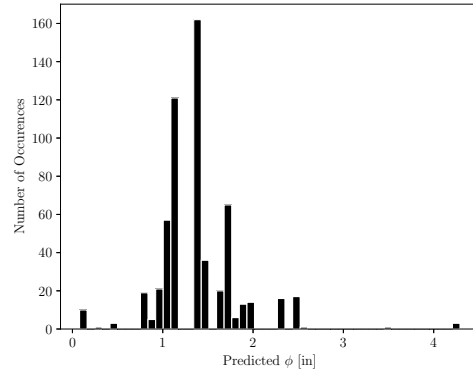
The comparison of the predicted to the measured weight in Figure 3.5a shows how the model in Equation 3.6 may over-predict or under-predict the weight of a given BLDC motor. Data points below the solid line are those that are under-predicted by the model, while data points above the line are over-predicted by the model. The distribution of predicted mass in Figure 3.5b shows where the model will be most accurate, which is where the majority of data points reside. In this case, the model is most appropriately applied to predict BLDC motor weights up to approximately 2 lb_f.

The diameter Φ in inches of BLDC motors can be predicted using Equation 3.7. R and I_{max} are not included in the correlation since they did not improve the statistical R^2 nor the error distribution from Figure 3.6. For this correlation, $N = 591$, $RMSE = 0.1834$, and Adjusted $R^2 = 0.8464$. Equation 3.7 can reasonably predict the diameter for the approximate range of $\Phi < 2.4$ in, corresponding to the majority of the data in the distribution shown in Figure 3.6b.

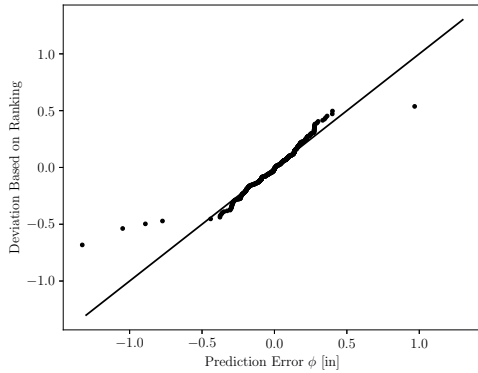
$$\Phi \text{ [in]} = 1.3348 K_V^{-0.13878} I_0^{0.08194} P_{max}^{0.16479} \quad (3.7)$$



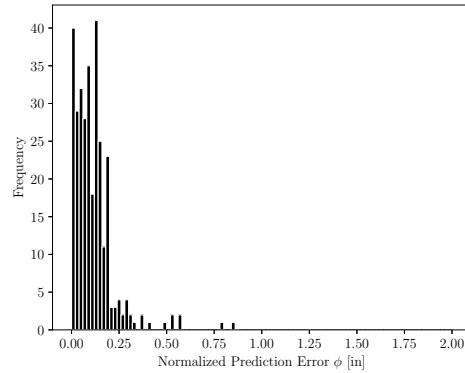
(a) Comparison of predicted to measured diameter.



(b) Distribution of predicted diameter.



(c) Statistical distribution of errors.

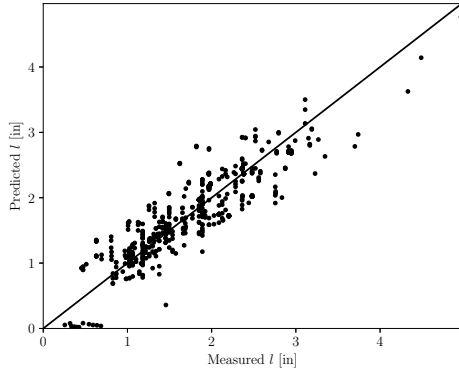


(d) Frequency distribution of errors.

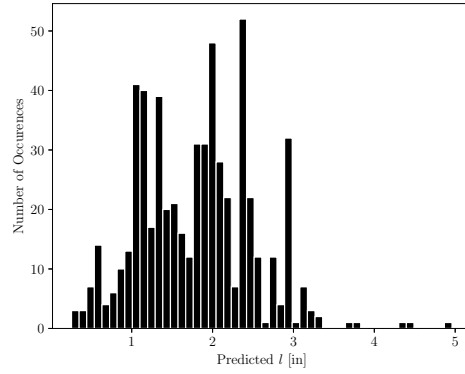
Figure 3.6: Visualization of BLDC motor diameter prediction.

The length ℓ in inches of BLDC motors can be predicted using Equation 3.8. As with volume, R and I_{max} are not included in order to obtain the best statistical R^2 and error distribution in Figure 3.7. For this correlation, $N = 586$, $RMSE = 0.3204$, and Adjusted $R^2 = 0.7891$. Equation 3.8 can reasonably predict the length for $\ell < 3.0$ in, corresponding to the data distribution shown in Figure 3.7b.

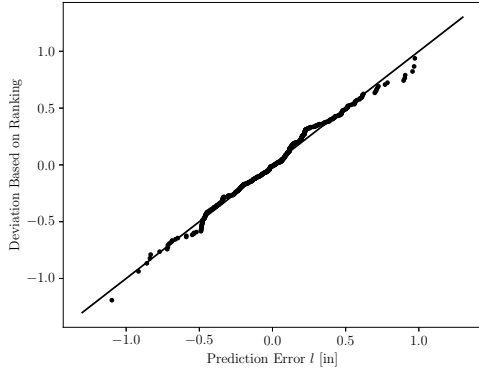
$$\ell \text{ [in]} = 1.1174 \times 10^{-1} K_V^{0.06242} I_0^{-0.03178} P_{max}^{0.37564} \quad (3.8)$$



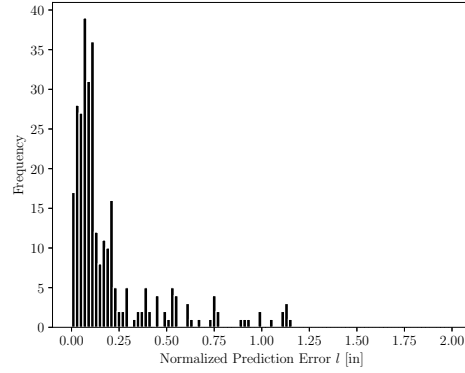
(a) Comparison of predicted to measured length.



(b) Distribution of predicted length.



(c) Statistical distribution of errors.



(d) Frequency distribution of errors.

Figure 3.7: Visualization of BLDC motor length prediction.

The statistical validity of the sizing correlations for BLDC motors is summarized in Table 3.1.

Table 3.1: Statistical validity of BLDC motor correlations.

Output	N	$RMSE$	R^2	Adjusted R^2
W	563	0.1170	0.9456	0.9453
Φ	591	0.1834	0.8472	0.8464
ℓ	586	0.3204	0.7902	0.7891

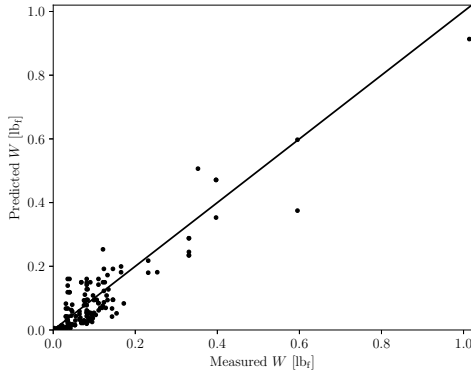
ESCs

The ESC serves as an interface between the motor and the autopilot, converting control signals from the autopilot into the power delivered to the motor, often with a receiver

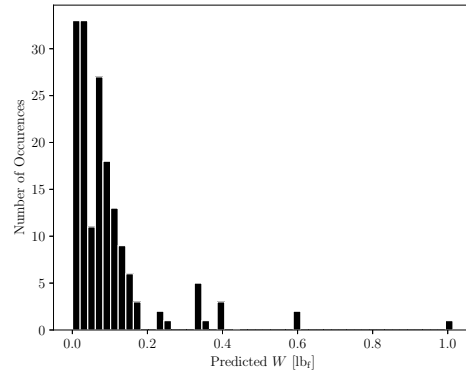
between the two components. The data used for the ESC correlations were provided upon request from the database developed in [12]. The commonly available parameters for ESCs are the maximum continuous current, I_{max} measured in amps; maximum continuous power, P_{max} measured in watts; and voltage, V measured in volts. Correlations have been developed for the mass and volume for ESCs. The equations and statistical analysis are presented in this section with a summary of N , $RMSE$, R^2 , and Adjusted R^2 for both correlations in Table 3.2.

The weight in lb_f of ESCs can be predicted using Equation 3.9. I_{max} and V are not included in order to obtain the best statistical R^2 and error distribution in Figure 3.8. For this correlation, $N = 168$, $RMSE = 0.0221$, and Adjusted $R^2 = 0.8378$. Equation 3.9 can reasonably predict the weight for $W < 0.4 \text{ lb}_f$, corresponding to the data in the distribution shown in Figure 3.8b.

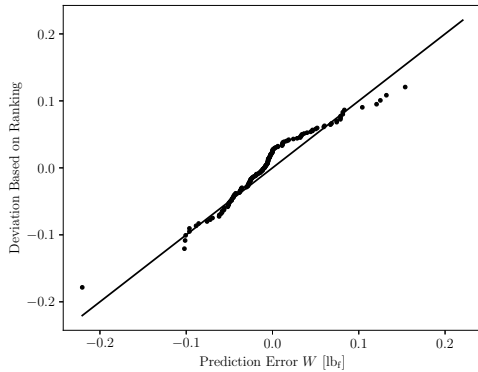
$$W [\text{lb}_f] = 3.2484 \times 10^{-5} P_{max}^{1.04733} \quad (3.9)$$



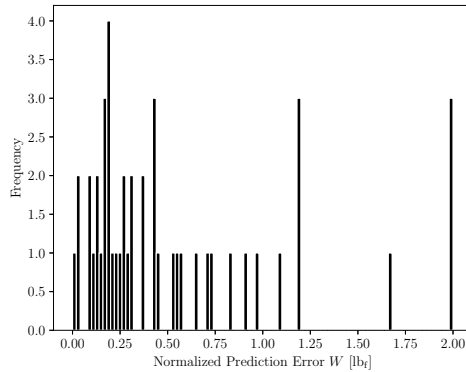
(a) Comparison of predicted to measured weight.



(b) Distribution of predicted weight.



(c) Statistical distribution of errors.

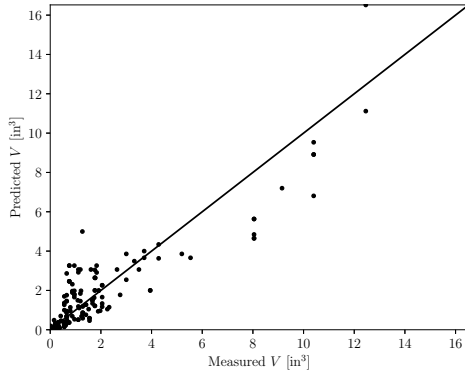


(d) Frequency distribution of errors.

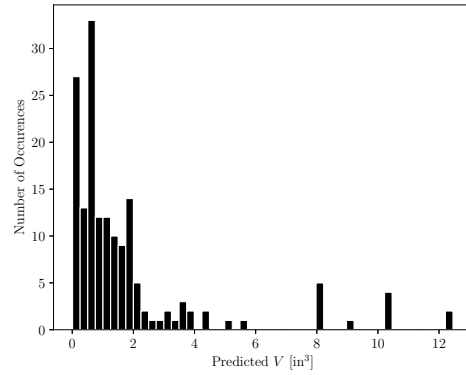
Figure 3.8: Statistical validation of ESC weight prediction.

The volume $\bar{V}$ in cubic inches of ESCs can be predicted using Equation 3.10. I_{max} and V are not included in order to obtain the best statistical R^2 and error distribution in Figure 3.9. For this correlation, $N = 163$, $RMSE = 1.0956$, and Adjusted $R^2 = 0.8038$. Equation 3.10 can reasonably predict the volume for $\bar{V} < \text{in}^3$.

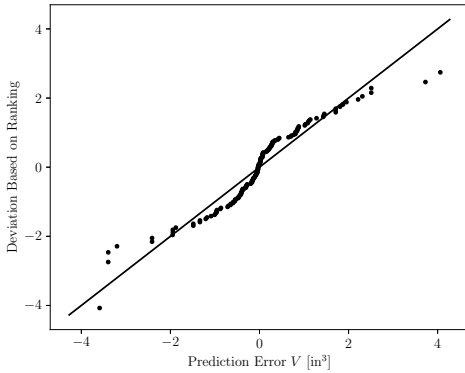
$$\bar{V} [\text{in}^3] = 1.17821 \times 10^{-3} P_{max}^{0.97616} \quad (3.10)$$



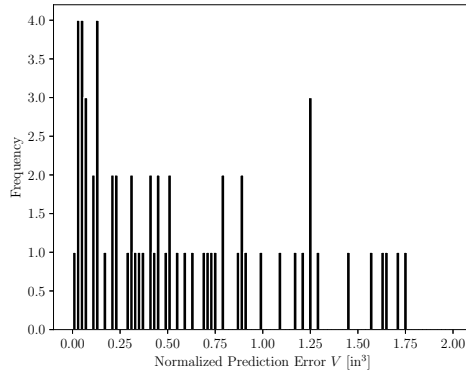
(a) Comparison of predicted to measured volume.



(b) Distribution of predicted volume.



(c) Statistical distribution of errors.



(d) Frequency distribution of errors.

Figure 3.9: Statistical validation of ESC volume prediction.

Table 3.2: Statistical validity of ESC correlations.

Output	N	$RMSE$	R^2	Adjusted R^2
W	168	0.0488	0.8388	0.8378
$\bar{V}$	163	1.0956	0.8050	0.8038

Propellers

Pressure differentials across propeller blades generate thrust and/or lift for UAVs. They are commonly characterized by the pitch and diameter of the blade. The pitch of the blade is the distance the propeller travels, without slipping, in one full rotation through an incompressible medium. Pitch is selected for the desired balance between acceleration and maximum speed. The data used for the propeller weight correlations are taken from APC Propellers [49]. This analysis presents weight correlations for 2-blade propellers of various types, including electric, regular, multi-rotor, slow flight, racing, and wide. The available geometric data includes the propeller diameter Φ_{prop} , pitch θ_{prop} , hub diameter Φ_{hub} , hub thickness t_{hub} , and shaft diameter Φ_{shaft} . All dimensions are in inches.

The weight of a small propeller can be predicted using Equation 3.11, where weight is in lb_f . The database for this correlation has 179 electric, 162 regular, 55 slow flight, 32 multi rotor, 27 racing, 18 wide, and 18 that combine multiple of these characteristics. Equation 3.11 is biased accordingly to each type of propeller. For this correlation, $N = 491$, $RMSE = 0.0113$, and Adjusted $R^2 = 0.9867$. Equation 3.11 can reasonably predict the weight for $W < 0.6 \text{ lb}_f$, in accordance with the data distribution in 3.10b, but will be more accurate at lower weights where the database is more biased.

$$W [\text{lb}_f] = 7.2966 \times 10^{-3} \theta_{prop}^{-0.05770} \Phi_{prop}^{1.42827} \Phi_{hub}^{0.89561} t_{hub}^{0.71261} \Phi_{shaft}^{0.31898} \quad (3.11)$$

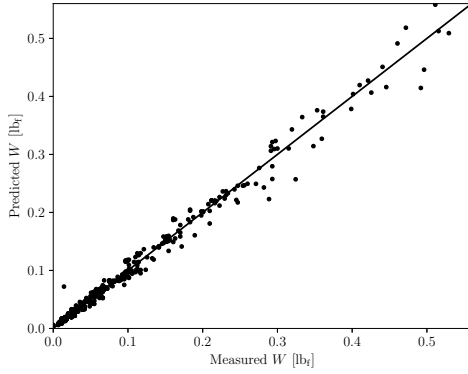
If the type of blade used for a design is known, the hub and shaft characteristics can be removed from Equation 3.11. The resulting equation is of the form

$$W [\text{lb}_f] = A \theta_{prop}^B \Phi_{prop}^C \quad (3.12)$$

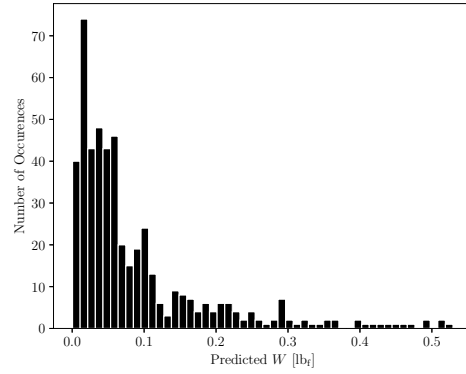
where the constants A , B , C are summarized in Table 3.3 along with statistical analysis for each propeller type. When using a regression for a specific type of propeller, N should be considered for the validity of that regression in conjunction with Adjusted R^2 . Correlations with a smaller N may result in larger errors. Equation 3.12 should only be used if the type of propeller is known, otherwise Equation 3.11 should be used. The graphical statistical representation for each type of propeller in Table 3.3 is shown in Appendix B.

Table 3.3: Coefficients and statistical analysis for regressions in the form of Equation 3.12.

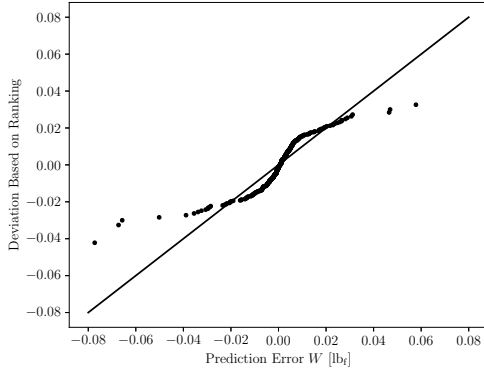
Type	A	B	C	N	$RMSE$	R^2	Adjusted R^2
Electric	6.3367×10^{-5}	0.04090	2.69864	179	0.0119	0.9825	0.9823
Regular	1.6467×10^{-4}	0.09020	2.52476	163	0.0130	0.9840	0.9838
Slow Flight	6.3114×10^{-5}	0.10742	2.51445	56	0.0030	0.9283	0.9256
Multi Rotor	1.6735×10^{-4}	-0.21952	2.42979	32	0.0034	0.9836	0.9825
Racing	2.7408×10^{-4}	0.16194	2.23641	27	0.0103	0.9866	0.9855
Wide	8.6017×10^{-5}	0.12487	2.73329	21	0.0310	0.9510	0.9456



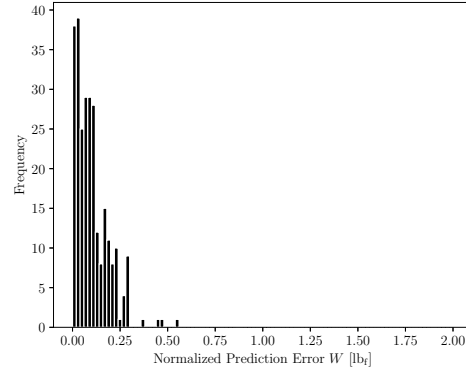
(a) Comparison of predicted to measured weight.



(b) Distribution of predicted weight.



(c) Statistical distribution of errors.



(d) Frequency distribution of errors.

Figure 3.10: Statistical validation of 2-blade APC Propeller prediction.

3.2.3 Lifting Surface Weight

This section presents the derivation of the lifting surface weight equation using a fully-stressed beam formulation. Consider a wing with a lift distribution and point load, as shown in Figure 3.11, where the wing structural box is assumed to be symmetric about the neutral axis. In the application of this equation in Chapter 5, the point load on the wing is a nacelle housing an MPPT and battery pack. The weight of this point load is dominated by the battery weight, thus is denoted by the subscript *batt* for its respective weight and location.

The bending moment, M , related to bending stress, σ , is denoted as Equation 3.13, where A is the total area of stressed material and t is the thickness.

$$M = 2\sigma \frac{A}{2} \frac{t}{2} \quad (3.13)$$

The shear stresses can be neglected because the bending stress dominates the stress-state, which is especially true at high $\mathcal{R}$. Additionally, neglecting shear stresses in this

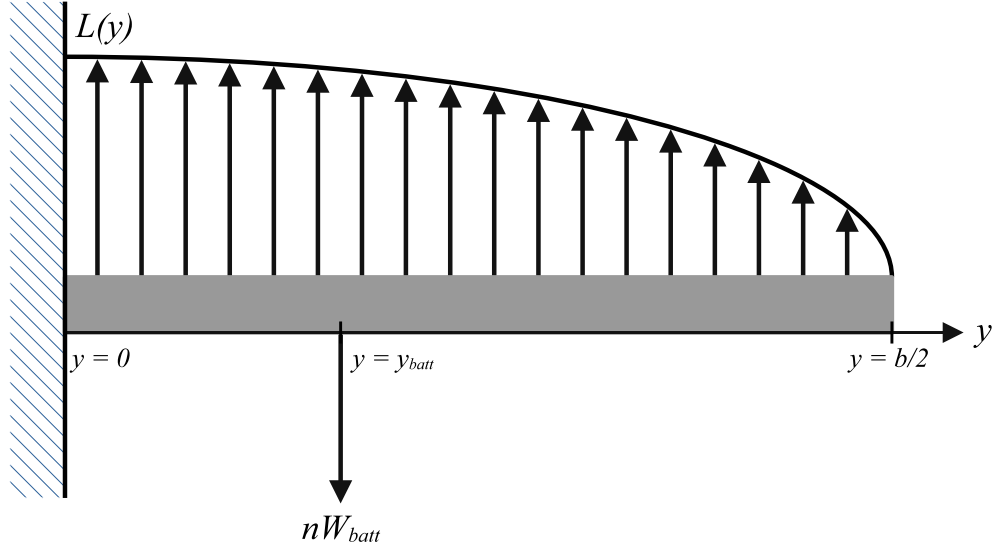


Figure 3.11: UAV wing modelled as a cantilever beam.

formulation is more conservative, resulting in a higher required structural weight to support the bending moment. Assuming bending stress to be the design limitation for wing structures, the required structural area along the semi-span is given by:

$$A(y) = \frac{2M(y)}{t(y)\sigma} \quad (3.14)$$

The structural weight of the whole wing due to bending, $W_{w,b}$, can be expressed as:

$$W_{w,b} = 2 \int_0^{b/2} \rho A(y) dy \quad (3.15)$$

where ρ is the density of the stressed material.

Substituting Equation 3.14 into Equation 3.15 and simplifying gives:

$$W_{w,b} = 4 \frac{\rho}{\sigma} \int_0^{b/2} \frac{M(y)}{t(y)} dy \quad (3.16)$$

Multiplying by the chord length as $c(y)/c(y)$ and simplifying leads to:

$$W_{w,b} = 4 \frac{\rho}{\sigma} \int_0^{b/2} \frac{2M(y)}{\frac{t(y)}{c(y)} c(y)} dy \quad (3.17)$$

Assuming a constant t/c along the span gives:

$$W_{w,b} = 4 \frac{\rho}{\sigma \frac{t}{c}} \int_0^{b/2} \frac{2M(y)}{c(y)} dy \quad (3.18)$$

Assuming a trapezoidal wing, the chord can be expressed as:

$$c(\eta) = c_r[1 - \eta(1 - \lambda)] \quad (3.19)$$

where:

$$\eta = \frac{y}{b/2} \quad (3.20)$$

$$c_r = \frac{2S}{b(1+\lambda)} \quad (3.21)$$

c_r is the root chord, λ is the taper ratio, S is the planform area, and b is the span.

Substituting into Equation 3.19, the chord becomes:

$$c(\eta) = \frac{2S}{b(1+\lambda)}[1 - \eta(1 - \lambda)] \quad (3.22)$$

Assuming an elliptical lift distribution as per lifting line theory for $-\frac{b}{2} \leq y \leq \frac{b}{2}$, the lift distribution can be expressed as:

$$L(y) = \frac{4n_z W}{b\pi} \sqrt{1 - \left(\frac{2y}{b}\right)^2} \quad (3.23)$$

where n_z is the g -loading and W is the total weight of the aircraft.

The bending moment distribution along the span may be derived as a double integral of the lift distribution along the wing, which requires accounting for boundary conditions at the wing tip to fully define the shear and moment distributions. This double integral is less conservative, as the boundary conditions introduces constant terms which decrease the required structural weight compared to the following formulation. The total bending moment at any point on the wing is defined by Equation 3.24.

$$M(y) = \begin{cases} \int_y^{b/2} L(y)ydy & \text{if } y > y_{batt} \\ \int_y^{b/2} L(y)ydy - \int_y^{y_{batt}} nW_{batt}dy & \text{if } y \leq y_{batt} \end{cases} \quad (3.24)$$

Substituting and expanding Equation 3.24 into Equation 3.18 results in Equation 3.25.

$$W_{w,b} = \frac{4\rho}{\sigma_c^{\frac{t}{c}}} \left[\int_0^{b/2} \frac{1}{c(y)} \int_y^{b/2} L(y)ydydy - \int_0^{y_{batt}} \frac{nW_{batt}}{c(y)} (y_{batt} - y)dy \right] \quad (3.25)$$

Further substitution, integration, and simplification leads to:

$$W_{w,b} = \frac{\rho}{\sigma} \frac{n_z R b (1 + \lambda)}{\frac{t}{c}} \left[\int_0^1 \frac{W}{3\pi} \left(\frac{(1 - \eta^2)^{3/2}}{1 - \eta(1 - \lambda)} \right) d\eta - \int_0^{\eta_{batt}} \frac{W_{batt}}{2} \left(\frac{\eta_{batt} - \eta}{1 - \eta(1 - \lambda)} \right) d\eta \right] \quad (3.26)$$

There are some placements and weights of the point load that changes the magnitude of the bending moment across the span. To account for this, the structural weight to reduce bending is better depicted as Equation Set 3.27.

$$\begin{aligned}
W_{w,b} &= k_w \frac{\rho}{\sigma} \frac{n_z R b (1 + \lambda)}{\frac{t}{c}} \left[\int_0^{\eta_{batt}} I_1 d\eta + \int_{\eta_{batt}}^1 I_2 d\eta \right] \\
I_1 &= \left| \frac{W}{3\pi} \left(\frac{(1 - \eta^2)^{3/2}}{1 - \eta(1 - \lambda)} \right) - \frac{W_{batt}}{2} \left(\frac{\eta_{batt} - \eta}{1 - \eta(1 - \lambda)} \right) \right| \\
I_2 &= \left| \frac{W}{3\pi} \left(\frac{(1 - \eta^2)^{3/2}}{1 - \eta(1 - \lambda)} \right) \right|
\end{aligned} \tag{3.27}$$

There is not a simple closed form solution for the integral in Equation 3.27. With a known taper ratio, the integral can be solved numerically and the weight of the wing calculated afterwards. For cases where there is no point load, Equation 3.27 is applied where $\eta_{batt} = 0$. The wing weight is dependent on the total weight of the aircraft, thus an iterative approach is required when using the correlation as part of a conceptual sizing tool. In validating the method against wing weights used in the author's lab group, presented in Appendix C, it was found that a scaling term k_w is required to match the calculated weight to the measured weight. $k_w = 100$ is used as a correction, which is used throughout the analysis in this thesis. The structures are assumed to be 3D-printed out of light-weight PLA (polylactic acid) with a carbon fibre core, where $\frac{\rho}{\sigma} = 4.58 \times 10^{-5} \text{ ft}^{-1}$. Note that Chapter 4 neglects the effects of point loads on the wing, whereas Chapter 5 includes them.

3.3 Payloads

This subsection details the payloads considered in Chapters 4 and 5, which have a significant impact on the overall performance and endurance of any electrically powered vehicle.

In this analysis, there are two effects of payload which are crucial to consider: weight and power consumption. The endurance of a solar-powered vehicle is directly linked to the power consumption of the vehicle which is discussed in Section 3.7. There are three different payload conditions defined, with options for a camera, GPS, and telemetry antenna. Table 3.4 summarizes the weight and continuous power draw for different payload components. Table 3.5 summarizes the payload combinations used to analyze the effects of payload. Option 0 is the baseline for the analysis, which is used for the majority of the results in Chapter 4. Option 1 consists of the lower weight and power consumption items listed in Table 3.4 to include one camera, one GPS, and one telemetry radio whereas option 2 consists of the higher weight and power consumption items.

Although each individual payload item would have an aerodynamic drag penalty, it is assumed that there are no external changes to the vehicle with different payload options. The potential ΔC_D for the payload would be the same across all analyzed UAVs, resulting in the same difference in power draw for all UAVs. The total flight time and performance of each UAV would shift by the same percent, but the overall trends and results would not change. For the sake of simplicity, the parasitic drag from external payloads are neglected and only the effects of the added weight and power draw are considered.

Table 3.4: Payload characteristics.

	P [W]	m [g]	Option	
SIYI A8 Camera	5	95	1	[95]
SIYI A2 Camera	2	85	2	[96]
MG F10 GPS without antenna	0.15	6	1	[97]
MG F10 GPS with antenna	0.275	21	2	[97]
LR900-A Telemetry	0.3	4	1	[98]
LR900-F Telemetry	1	8	2	[98]

Table 3.5: Payload combinations for analysis.

Option	P [W]	m [g]
0	0	0
1	2.45	95
2	6.275	124

3.4 Aerodynamics

The aerodynamics of each vehicle are analyzed using the aerodynamics module of *py-ACDT*. The module is explained in further detail by Jansen in [99], with the basics outlined in this subsection. The total drag consists of parasitic drag and induced drag. The parasitic drag is calculated by a detailed build-up method with various corrections and empirical methods which account for lifting surfaces and bodies of revolution, *i.e.* the fuselage and nacelles where applicable [100, 101].

As part of the drag build-up method, the parasitic drag, C_{D_0} , can be related to skin friction coefficients, C_f , based on the surface roughness and wetted area. Regardless of turbulent or laminar flow, the skin friction coefficient is related to the inverse of the Reynold's Number, Re , such that increasing Re results in a decrease in C_f . Parasitic drag is directly proportional to C_f , therefore an increase in Re results in a decrease in C_{D_0} [47].

The induced drag is estimated by combining two leading edge suction methods, estimating the ratio between the theoretical maximum and actual leading edge suction force, S , to calculate the Oswald efficiency factor, e , to calculate the lift induced drag factor K . There is a minor deviation from the method defined in [99] in the expressions of S . In this analysis, the two expressions for S which are averaged together are:

$$S_1 = 0.918 + 0.02388 \log K_s \quad (3.28)$$

$$S_2 = 0.974 - 0.0976 \exp^{-0.456 K_s} \quad (3.29)$$

where the parameter K_s is given by:

$$K_s = \frac{\mathcal{R}\lambda}{\cos \Lambda_{LE}} \quad (3.30)$$

where λ is the taper ratio, $\mathcal{R}$ is the aspect ratio, and Λ_{LE} is the sweep of the leading

edge. The Oswald's efficiency is then given by:

$$e = \frac{1.0}{S + (1 - S) \frac{\pi \mathcal{R}}{C_{L,\alpha}}} \quad (3.31)$$

and is used to calculate K with:

$$K = \frac{1}{\pi \mathcal{R} e} \quad (3.32)$$

The lift induced drag coefficient is given by $C_{D_i} = K C_L^2$. The parametric analysis discovered there is a physical limitation for this method, where it is only valid for $\mathcal{R} \times \lambda \leq 11.8$. As a result, some of the geometries at large spans with elevated values of $\mathcal{R} \times \lambda$ were unable to be analyzed aerodynamically and not considered, which explains the fewer data points in the results as the span increases. A validation of the aerodynamic analysis is included in Appendix D.

3.5 Point Mass Model

The dynamic and kinematic model used for each flight simulation is a point-mass system with three degrees of freedom. The earth axis is modelled as the fixed frame with the origin located at mean-sea level, denoted by subscript E . The UAV body axis, denoted by subscript B , is shown in Figure 3.12 with the rotational angles used to translate between earth and fixed body axes. Figure 3.13 shows the relationship between the earth axis and the aircraft's velocity axis, denoted as subscript V .

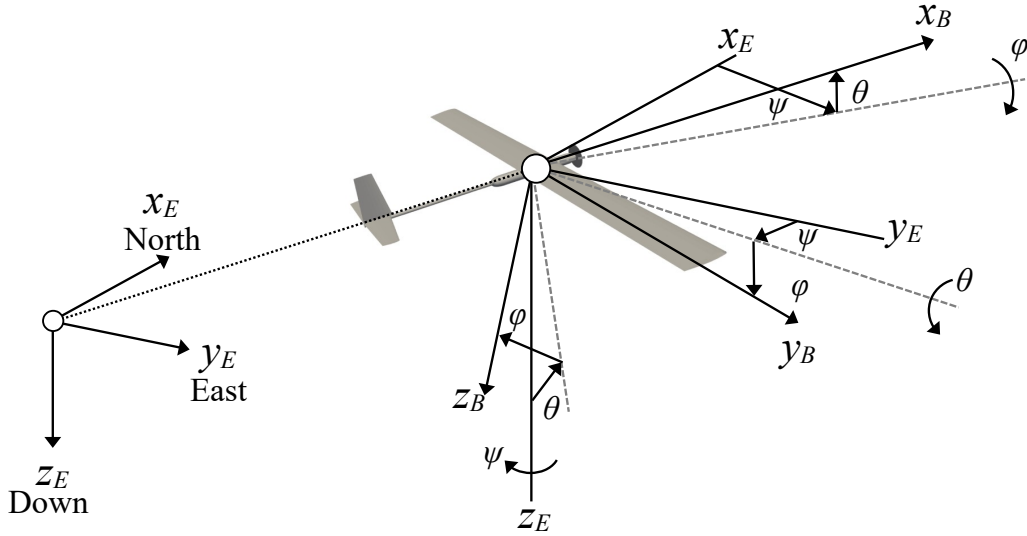


Figure 3.12: Aircraft body axis orientation [102].

The equations of motion are specific to a set of axes. The dynamics of the aircraft are expressed in body axis but must consider forces and velocities defined in the earth axis. For a vector in the body axis, $\vec{F}_B$, the rotational transformation to convert to a vector in the earth axis, $\vec{F}_E$, is defined in Equation 3.33. The order in which the rotations are applied is significant.

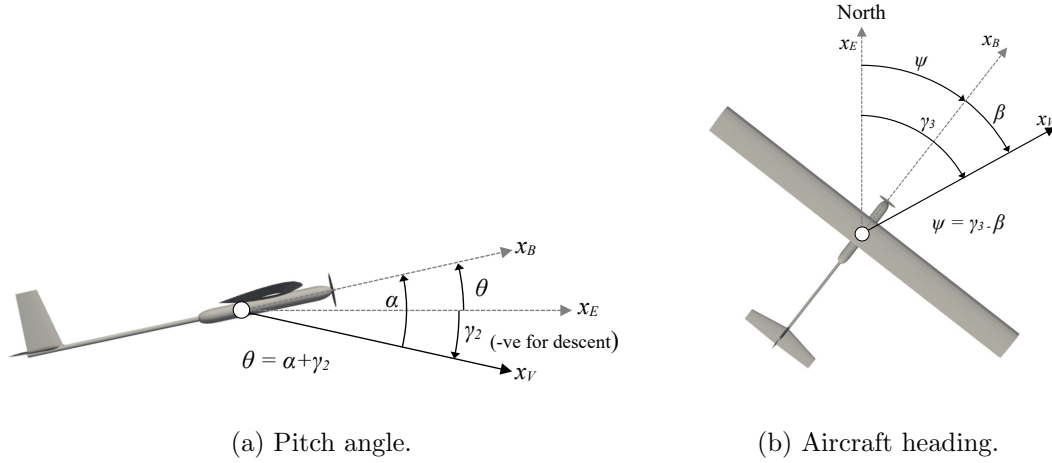


Figure 3.13: Angular relationships between earth, body, and velocity axes (subscripts E , B , and V respectively) [102].

$$\vec{F}_E = \begin{bmatrix} \cos \psi & -\sin \psi & 0 \\ \sin \psi & \cos \psi & 0 \\ 0 & 0 & 1 \end{bmatrix} \begin{bmatrix} \cos \theta & 0 & \sin \theta \\ 0 & 1 & 0 \\ -\sin \theta & 0 & \cos \theta \end{bmatrix} \begin{bmatrix} 1 & 0 & 0 \\ 0 & \cos \phi & -\sin \phi \\ 0 & \sin \phi & \cos \phi \end{bmatrix} \vec{F}_B \quad (3.33)$$

The total rotational transformation matrix from body to earth axis, $\bar{R}_B$, simplifies into Equation 3.34 [102]. The transformation from earth to body axis would use the inverse, such that $\bar{R}_E = [\bar{R}_B]^{-1}$.

$$\bar{R}_B = \begin{bmatrix} \cos \psi \cos \theta & \cos \psi \sin \theta \sin \phi - \sin \psi \cos \phi & \cos \psi \sin \theta \cos \phi + \sin \psi \sin \phi \\ \sin \psi \cos \theta & \sin \psi \sin \theta \sin \phi + \cos \psi \cos \phi & \sin \psi \sin \theta \cos \phi - \cos \psi \sin \phi \\ -\sin \theta & \cos \theta \sin \phi & \cos \theta \cos \phi \end{bmatrix} \quad (3.34)$$

The general performance equations of motion are not used directly in the numerical flight simulations in the parametric analysis or optimization, but are introduced now for applications in Chapter 6. The performance equations of motion in the velocity axis are presented in Equation Set 3.35 [102]:

$$\begin{aligned} m\dot{V} + \dot{m}V &= -D + [T \cos(\alpha + \tau_1) \cos \beta - D_m] - W \sin \gamma_2 \\ mV\dot{\gamma}_3 \cos \gamma_2 &= Y \cos \gamma_1 + L \sin \gamma_1 + T[-\cos(\alpha + \tau_1) \sin \beta \cos \gamma_1 + \sin(\alpha + \tau_1) \sin \gamma_1] \\ -mV\dot{\gamma}_2 &= Y \sin \gamma_1 - L \cos \gamma_1 + T[-\cos(\alpha + \tau_1) \sin \beta \sin \gamma_1 - \sin(\alpha + \tau_1) \cos \gamma_1] \\ &\quad + W \cos \gamma_2 \end{aligned} \quad (3.35)$$

where D is the drag force, L is the lift force, T is the thrust produced by the engine which is vectored vertically in pitch by τ_1 , D_m is the momentum drag of the propulsion system, Y is the side force, W is the weight, m is the mass, V is the true airspeed, α is the angle of attack, β is the side slip angle, γ_1 is the bank angle also denoted as ϕ . It is assumed that any thrust vectoring in the y_B -axis is symmetrical, constant, and therefore negligible, thus not included in this formulation.

Assuming constant mass, symmetric flight with no side force or side slip, a negligible angle between the thrust vector and the velocity vector, and momentum drag accounted for in the net thrust T , Equation 3.35 simplifies to Equation 3.36.

$$\begin{aligned}\frac{dV}{dt} &= \frac{T \cos \alpha - D}{m} - g \sin \gamma_2 \\ \frac{d\psi}{dt} &= \frac{(T \sin \alpha + L) \sin \phi}{mV \cos \gamma_2} \\ \frac{d\gamma_2}{dt} &= \frac{(T \sin \alpha + L) \cos \phi}{mV} - \frac{g \cos \gamma_2}{V}\end{aligned}\tag{3.36}$$

The aerodynamic, propulsive, and gravitational forces acting on the aircraft in accordance with the assumptions used for developing Equation 3.36 are shown in Figure 3.14.

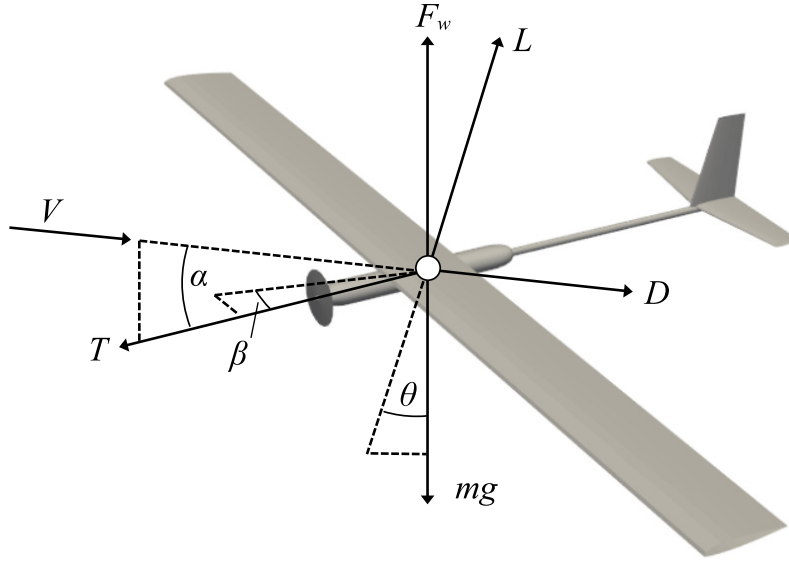


Figure 3.14: Forces acting on the aircraft in steady level flight.

The lift and drag forces are defined as:

$$L = \frac{1}{2} \rho V^2 S C_L \tag{3.37}$$

$$D = \frac{1}{2} \rho V^2 S C_D \tag{3.38}$$

where the respective non-dimensionalized coefficients are:

$$C_L = C_{L,\alpha} \alpha + C_{L_0} \tag{3.39}$$

$$C_D = C_{D_0} + K C_L^2 \tag{3.40}$$

The constants in Equation 3.39 and 3.40 are calculated by the aerodynamic module in *pyACDT*, further detailed in Section 3.4.

The kinematic equations in the earth axis are expressed as:

$$\begin{aligned}\dot{x} &= V \cos \gamma_2 \cos \psi \\ \dot{y} &= V \cos \gamma_2 \sin \psi \\ \dot{h} &= -\dot{z} = V \sin \gamma_2\end{aligned}\tag{3.41}$$

where h refers to altitude, defined as positive in the opposite direction to the z -axis. The time derivative of each variable is indicated with the dot.

3.6 Solar Model

Solar irradiance is dependent on geographical location and time of year. Studies show that the most promising design for prolonged solar flight also varies with location and time of year [58, 62, 86]. The position of the sun is modelled with an uncertainty of $\pm 0.0003^\circ$ in elevation and azimuth angles for the years 2000 to 6000 [103]. The solar irradiance model is developed from data spanning 8 years, using variables such as elevation angle, cloud fraction, and ozone and water vapour quantities [104]. Both models are used to determine the position of the sun and irradiance levels for a given geographical location and time of year through an open-source module, *pySolar*, which combines both of these models [105].

The solar azimuth angle, ψ_s , is defined as $\psi_s = 0$ being north, increasing positive to the east, bounded as $0 \leq \psi_s < 360$ deg. The solar elevation angle, γ_s , is defined as zero at the horizon, positive above the horizon, bounded by $-90 \leq \gamma_s \leq 90$ deg, shown in Figure 3.15. The outputs of *pySolar* are shown in Figure 3.16 with a start time of 0700 on 1 July 2025. The available solar irradiance is shown in Figure 3.16a. The azimuth and elevation angles are shown in Figure 3.16b.

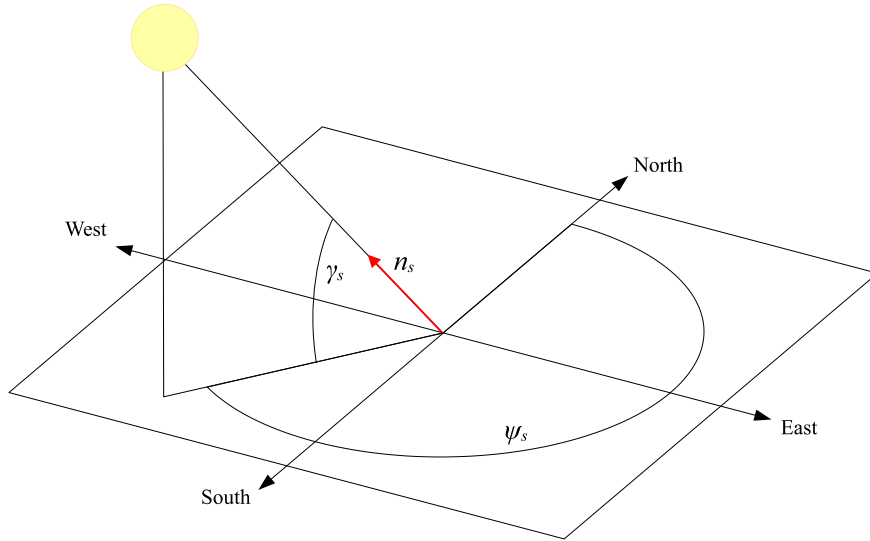


Figure 3.15: Azimuth and elevation angle convention.

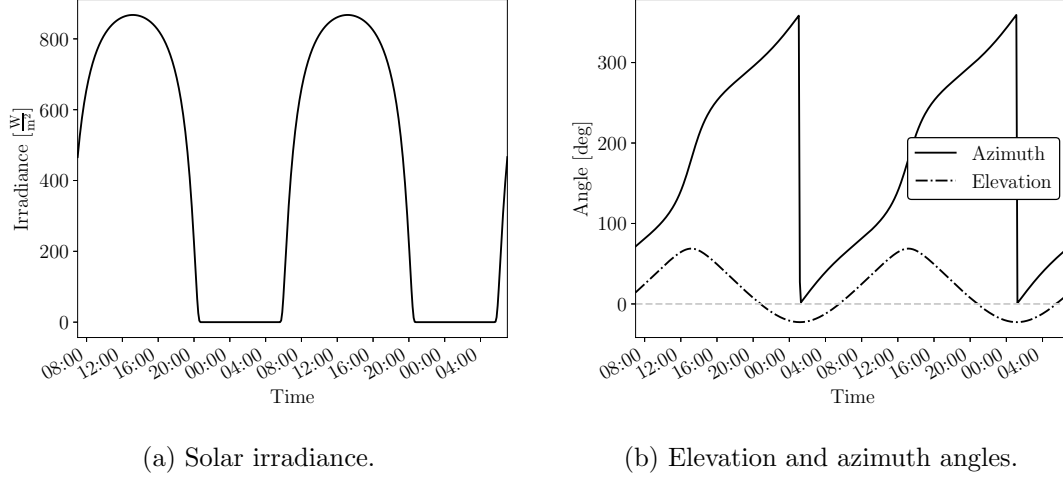


Figure 3.16: Sample solar data starting at 0700 1 July 2025 in Kingston, ON.

The solar vector, $\vec{n}_s$, shown in Figure 3.15, is defined by polar coordinates from a fixed point on the ground. The normalized solar vector in earth coordinates shown in Figure 3.17 is defined in Equation 3.42. Due to the large distance between the sun and the earth and the relatively short distance between the earth and the aircraft, it is assumed that the difference in the solar vector between the sun and specified earth coordinates and the vector between the sun and the aircraft in flight is negligible. The output azimuth and elevation angles are assumed to be the same for the aircraft throughout its simulated flight, regardless of its position within its trajectory.

$$\vec{n}_s = \begin{Bmatrix} n_x \\ n_y \\ n_z \end{Bmatrix}_e = \begin{Bmatrix} \cos \gamma_s \cos \psi_s \\ \cos \gamma_s \sin \psi_s \\ -\sin \gamma_s \end{Bmatrix}_e \quad (3.42)$$

Note that the z -axis in the earth axis is defined as positive down, therefore $n_{z,s} < 0$ during daylight hours. For each UAV being modelled in Chapter 4 and the vehicle used in Chapter 6, the solar panel normal vector in the body axis is assumed to be $\vec{n}_{PV,B} = \{0, 0, -1\}$. This assumption neglects the alignment of the solar cells to the sun based on the airfoil curvature and installation angle of the wing. To account for some of these effects, the efficiency of the panels is reduced from the published maximum efficiency from 24% to 22% in Chapters 4 and 6. The normal vector for each solar panel on the vehicles analyzed in Chapter 5 consider the placement on the airfoil, angle of attack, wing dihedral, and incidence angle, where the efficiency is assumed to be 23%.

The degree to which the solar and panel normal vectors are aligned is defined as the obliquity factor, κ . The solar panel normal is converted to the earth axis by $\vec{n}_{PV,E} = [R_B]\vec{n}_{PV,B}$, and is used in Equation 3.43 to determine the obliquity factor based on the angle θ_s between the vectors as shown in Figure 3.17. κ approaches 1 and θ_s approaches 0 as the vectors become more aligned. A larger obliquity factor means higher solar energy capture.

$$\kappa = \cos \theta_s = \vec{n}_{s,E} \cdot \vec{n}_{PV,E} \quad (3.43)$$

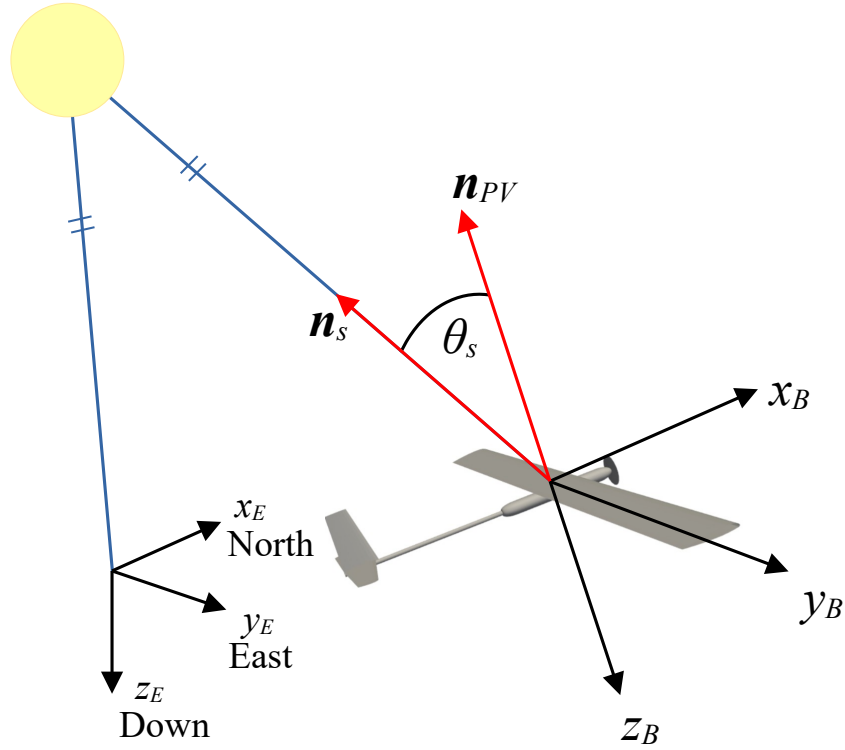


Figure 3.17: Solar vectors $\vec{n}_{PV}$ and $\vec{n}_s$ and the obliquity angle θ_s .

When considering the effects of geographical location and time of year, the variation in solar irradiance and elevation angle has a significant effect on the total flight time. Two locations are considered within this analysis: Kingston, ON and Inuvik, NT. Kingston is selected as the baseline for the majority of the analysis, with the ultimate objective of building and testing a solar UAV at the Royal Military College of Canada. Inuvik is selected for its military significance as a forward operating location for the Canadian Armed Forces and geo-political influences aimed at increasing presence and surveillance in the Canadian Arctic. A map with these locations is shown in Figure 3.18. The specific coordinates used are $(44.2717, -76.4369)$ and $(68.3589, -133.6936)$ for Kingston and Inuvik, respectively.

Figures 3.19 and 3.20 shows the irradiance for the 21st day of each month in Kingston, ON and Inuvik, NT, respectively. Note that both figures have the same vertical scale, which emphasizes the differences in irradiance, with the time shown in UTC. The months are listed numerically in the legend, where 1 corresponds to January and 12 to December. There is further justification to the selection of these locations in Chapter 4.

The summer in Inuvik has a similar maximum irradiance as Kingston in the winter but the hours of daylight vary drastically. There is more consistency in the irradiance and daylight hours in Kingston than in Inuvik. Inuvik has a high enough latitude that the sun does not fully set in summer months and does not rise in winter months. In the winter, it would not be possible to use solar power to increase the endurance, whereas

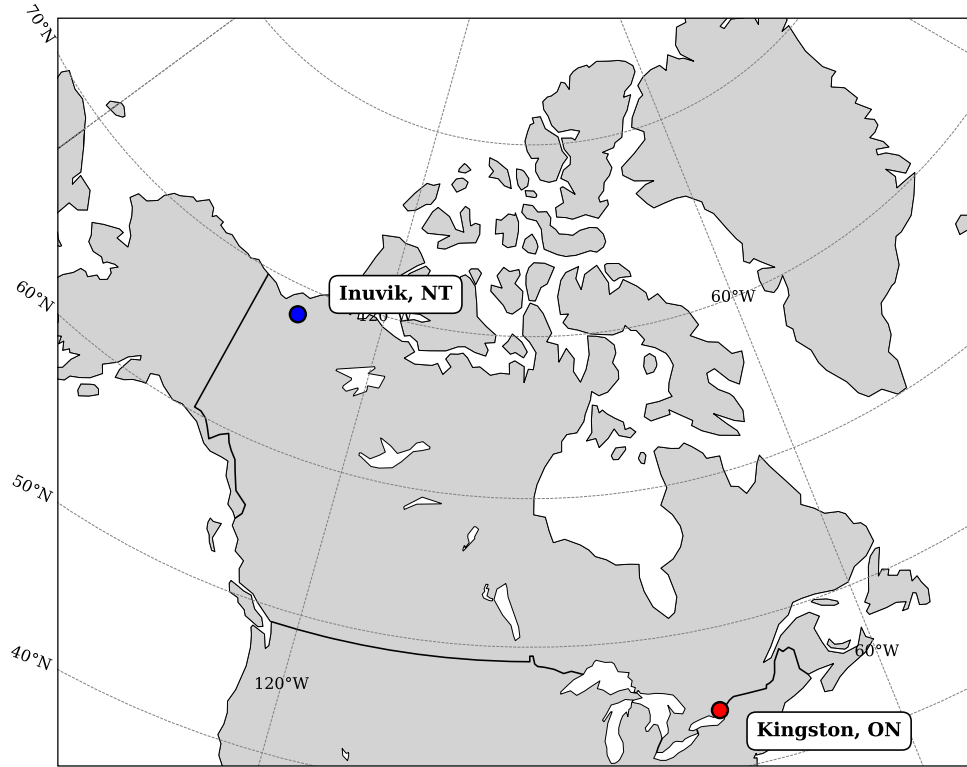


Figure 3.18: Map of Canada with points of interest.

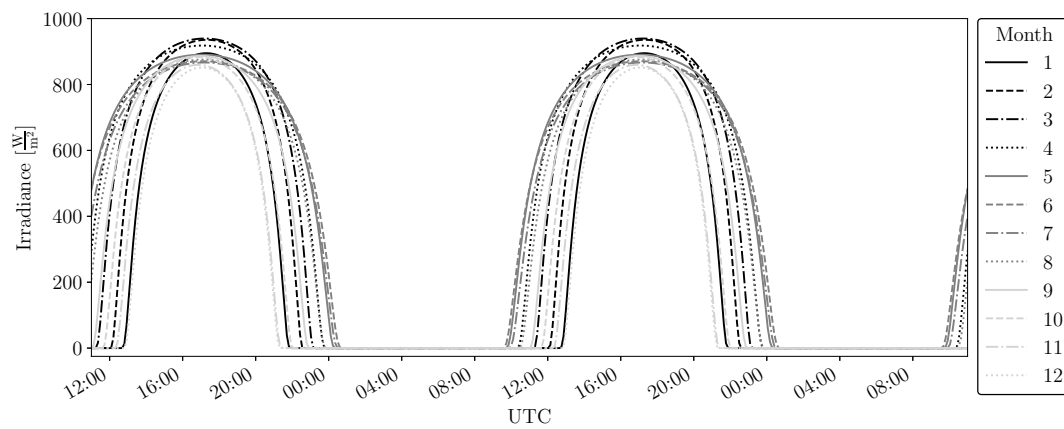


Figure 3.19: Solar irradiance throughout the year in Kingston, ON.

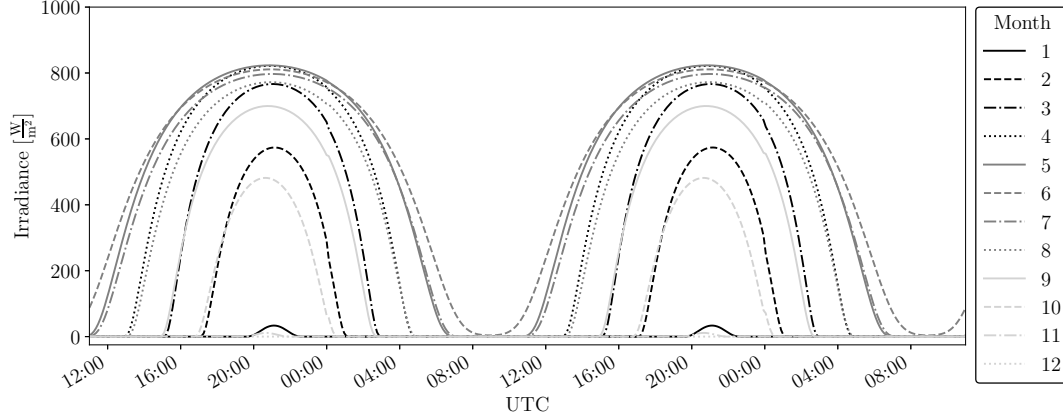


Figure 3.20: Solar irradiance throughout the year in Inuvik, NT.

the summer could almost be flown exclusively on solar power if there were minimal disruptions to the irradiance received by the vehicle. Throughout this thesis, the effects of weather on the available solar irradiance are neglected and the irradiance shown in Figures 3.19 and 3.20 are assumed to be the true irradiance at ground level.

3.7 Energy Model

There are several assumptions that are made for the electrical properties of the solar UAVs analyzed. This section details those assumptions, followed by equations used to model the energy balance throughout the simulated flight of a given vehicle. With the exception of the number of solar cells and number of batteries in parallel, the electrical properties are assumed to be the same for all vehicles throughout this research.

3.7.1 Battery Packs

The electrical properties of the solar UAV are assumed to be similar to other UAVs that have been built and flown within the author's lab group. To use similar autopilots, BLDC motors, and ESCs, each battery pack is fixed at three cells in series ($m_{batt} = 3$) to define the operating voltage.

This analysis assumes that two MPPTs are required to accommodate the solar arrays. Each MPPT is connected to a battery pack and a solar array all in series. The two circuits with MPPTs are connected in parallel to each other, maintaining the operating voltage and effectively doubling the energy capacity of the vehicle. Both battery packs are assumed to be identical, with a variable number of cells in series, n_{batt} . The analysis assumes the same properties as the Molicell 21700 Li-ion cells, which have a nominal capacity of $I_{nom} = 4.2$ Ah and a voltage range of 2.5 V to 4.2 V [93]. The nominal voltage is $V_{nom} = 3.7$ V. The total battery capacity, E is defined in Equation 3.44 [106].

$$E = I_{nom} V_{nom} n_{batt} m_{batt} n_{pack} \quad (3.44)$$

3.7.2 Solar Cells

The size and layout of the solar array must be compatible with the selected MPPT. Several solar UAVs have custom MPPTs, providing greater flexibility for the electrical properties of the solar array and orientation [1, 12, 107]. This project aims to use technology available on the market, therefore the analysis selects the number of solar panels to be compatible with the 6SBB MPPT from Packet Digital, designed specifically for integration with ArduPilot. Its electrical properties are summarized in Table 3.6, whose data sheet was provided upon request from [108]. The GVB-8 Lithium 14.2 Volt MPPT was considered, whose properties are side by side those of the 6SBB. The 6SBB MPPT is a better choice due to its lighter weight and higher maximum power. The MPPT efficiency is assumed to be 97% throughout the analyses in this thesis.

Table 3.6: MPPT electrical properties.

		GVB-8	6SBB
Panel Max Power Voltage	[V]	13	30
Min Battery Voltage for Operation	[V]	9.5	8
Max Input Current	[A]	8	8
Max Power	[W]	105	240
Weight	[g]	185	30
Reference		[109]	[108]

The selected solar cells are the Maxeon Gen 3 C60 cells. These are selected for their flexibility, cost, and efficiency, which are also used in [1, 25, 35]. Primary characteristics of the solar array are summarized in Table 3.7.

Table 3.7: Maxeon C60 Solar Cell electrical properties [110].

		Min	Max
V_{mpp}	[V]	0.616	0.626
I_{mpp}	[A]	6.01	6.09
V_{oc}	[V]	0.724	0.734
Efficiency	[%]	24	24.4

The solar cells must be connected in a manner that is compatible with the MPPT properties. Considering the maximum input current of the MPPT, the solar array cannot have more than one cell in parallel. When solar irradiance is available, there needs to be more voltage across the solar circuit than across the batteries in order to charge the batteries. Considering the change in voltage with temperature, $-0.236\%/^{\circ}\text{C}$, and a temperature range from -20°C to 40°C , one MPPT can accommodate 17 to 36 solar cells in series.

The power collected from the solar array is dependent on the solar irradiance, I_s , the surface area of the solar array, S_{PV} , the efficiency of the solar cells, η_{PV} , the efficiency of the MPPT, η_{MPPT} , and the obliquity factor, κ . The collected solar power of the

aircraft is expressed as Equation 3.45.

$$P_s = \kappa I_s S_{PV} \eta_{PV} \eta_{MPPT} \quad (3.45)$$

3.7.3 Propulsion

Each UAV is propeller driven. There are two methods used to determine the propulsive power. Method 1 is used in Chapters 4 and 6, method 2 is used in Chapter 5. In both methods, the thrust produced must be in equilibrium with the drag of the aircraft to maintain steady level flight.

Method 1

The power consumed by the propulsion system, P_p , is defined by Equation 3.46, where V is the true airspeed, η_m is the efficiency of the motor, η_{prop} is the propeller efficiency, and in steady level flight it is assumed that $T = D$ from the force balance in Figure 3.14.

$$P_p = \frac{TV}{\eta_m \eta_{prop}} \quad (3.46)$$

For a well-designed electric propulsion system, the combined efficiency of a motor and propeller optimized for a specific application or flight condition should be around 85% [12]. For the parametric comparison, it is assumed that $\eta_m \times \eta_{prop} = 0.85$ for every UAV in the analysis, despite having the same nominal motor and propeller combination. For some designs, this may be an optimistic efficiency depending on the airspeed and required thrust. Smaller propellers may not be able to achieve this combined efficiency, regardless of the motor efficiency. This assumed propulsive efficiency is used on the assumption that a propeller and motor combination exists that can provide 85% efficiency for the cruising airspeed of a given vehicle. Real efficiency values will be a function of the airspeed and power setting, which is accounted for in Method 2.

Method 2

The power consumed by the propulsion system in Method 2 is more sophisticated than in Method 1. This method considers the electrical properties of a BLDC motor and ESC and the performance of a given propeller.

The power balance of the shaft of the BLDC motor is defined as

$$P_{shaft} = P_{electric} - P_{copper} - P_{no-load} \quad (3.47)$$

where

$$P_{electric} = V_{max} I \quad (3.48)$$

$$P_{copper} = R_m I^2 \quad (3.49)$$

$$P_{no-load} = V I_0 \quad (3.50)$$

$$I_0 = I_{0ref} \sqrt{\frac{V}{V_{ref}}} \quad (3.51)$$

where the copper losses account for heat wasted by the internal resistance of the motor windings and the no-load losses account for the no-load current circulating through the motor without an applied load on the shaft. V is the voltage, I is the current, R_m is the internal resistance, I_0 is the no-load current, and $I_{0_{ref}}$ is the manufacturers reference no-load current at V_{ref} .

For a given thrust setting T_{set} , the voltage applied to the motor is given by Equation 3.52. $P_{no-load}$, I_0 , and $P_{shaft_{max}}$ are calculated with this V .

$$V = V_{max} T_{set} \quad (3.52)$$

Equation 3.47 can be rearranged in terms of current, I , in Equation 3.53 to solve for the current required for a given shaft power.

$$I = \frac{V_{max} \pm \sqrt{V_{max}^2 - 4R_m(P_{no-load} + P_{shaft})}}{2R_m} \quad (3.53)$$

The maximum possible shaft power at a given voltage is defined where the discriminant of Equation 3.53 = 0, given as:

$$P_{shaft_{max}} = \frac{V^2}{4R_m} - P_{no-load} \quad (3.54)$$

The shaft power is also defined as

$$P_{shaft} = Q(\omega)\omega \quad (3.55)$$

where $Q(\omega)$ is the torque produced and ω is the angular speed. ω is assumed to decrease linearly with the motor current, defined as

$$\omega = K_V(V - R_m I) \quad (3.56)$$

Once the motor current is known, the efficiency, η , and torque can be calculated:

$$\eta = \frac{P_{shaft}}{P_{electric}} \quad (3.57)$$

$$Q = \frac{P_{shaft}}{\omega} \quad (3.58)$$

Then the rotational speed of the motor is matched to the rotational speed of the propeller. Given the propeller data from [49] for a given propeller, the rotational speed has an associated thrust and power. In steady level flight, the thrust setting is iterated so that $T = D$. The resulting $P_{electric}$ is considered as the propulsive power, P_p , which includes an assumed constant ESC power whereby

$$P_p = P_{electric} + \frac{I^2}{R_{ESC}} \quad (3.59)$$

3.7.4 State of Charge

The battery's SOC is dependent on the power consumed by the propulsion system, payload, or other onboard systems. The rate of change of the SOC is defined in Equation 3.60. P_{pl} is the continuous power draw of the payload, where assumed values are previously defined in Table 3.5. Recall that E is the maximum battery capacity of the aircraft, defined in Equation 3.44. Equation 3.60 assumes a constant charge and discharge, which is based on the battery model in [106]. The model is nearly linear when applied between a SOC of 20% and 90%, for an effective capacity of 70%. This work considers the SOC between 30% and 95%, which lies outside of the original model bounds but has a smaller effective capacity thus does not have negative implications on the analysis contained herein. The lower bound on the SOC serves as a safety factor to protect other electrical components from brown-out. The larger the power draw of the payload or propulsion system, the faster the battery will discharge. Alternatively, the more solar power is available and collected, the slower the battery will discharge or the faster it will charge. This is included as a state in the flight simulations along with the performance equations of motion.

$$\frac{dSOC}{dt} = \frac{P_s - P_p - P_{pl}}{E} \quad (3.60)$$

The analysis methods and modelled described within this chapter are used to calculate the weight and aerodynamic properties of a given geometrical description of a fixed-wing solar UAV. The point mass model, solar model, and energy model are used to simulate the flight of the UAV. The following chapters adopt these methods for their respective analyses, which include a parametric comparison of different configurations, the optimization of a solar UAV, and a modified trajectory optimization.

4 UAV Configuration Analysis and Selection

This chapter presents a parametric comparison of different solar UAVs to determine the most promising configuration for LALE solar-powered flight. The study primarily seeks to analyze the trade off between aerodynamic efficiency and onboard energy capacity. A further aim of this chapter is to validate if a glider-like wing is appropriate for long endurance missions and if that remains true for applications requiring larger payloads, both in weight and power consumption.

Sections of this chapter present the defined objective, design variables, main geometry and sizing methods, and flight assumptions, drawing from the methods and models described in Chapter 3. The results section analyzes which configuration is most optimum for LALE solar-powered flight, and some of the effects of time of year and latitude.

4.1 Objective

The general objective of a solar UAV is to maximize the flight time, aiming for multiple cycles of 24 hours. With this approach, there is no way to distinguish between a design that nearly depletes its SOC before recharging and achieves continuous 24 hour flight cycles and a design with larger energy margins and a higher minimum SOC within a 24 hour flight cycle. A vehicle with larger energy margins would be less susceptible to unpredictable disturbances such as wind or cloud cover, thereby being the better vehicle for prolonged solar-powered flight.

With the overarching goal of maximizing flight time, various objective functions have been proposed that do not explicitly maximize flight time [74, 76, 78, 84]. While these objectives are useful for their respective research methods, the dependency on a fixed mission or flight time is not suitable for the various configurations analyzed in this study. Rather, this analysis considers the definition of excess time, t_{exc} , and charge margin, t_{cm} in a similar manner to Oettershagan *et al.* [1]. The definitions of both are shown in Figure 4.1, which plots the solar irradiance and SOC. Excess time is a useful measure of the excess energy before the batteries begin to recharge but does not directly increase the robustness to decreases in collected solar power. Excess time is necessary for increases in power consumption for all times throughout the flight when the battery SOC is not at its maximum, while t_{cm} is necessary to provide robustness to cloud cover [1]. Practical designs will need adequate margins for both t_{exc} and t_{cm} . Optimizing for t_{exc} has the same effect of maximizing the minimum SOC.

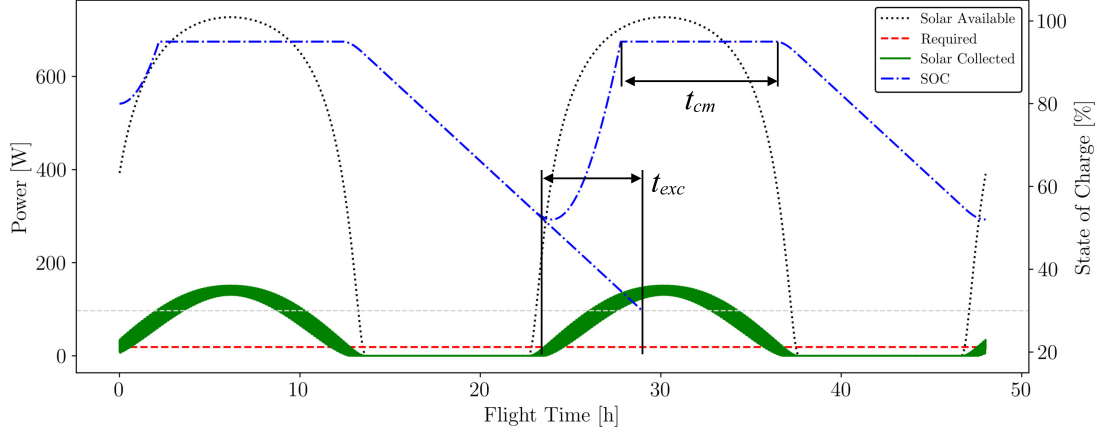


Figure 4.1: SOC and power levels for a 48 h flight, showing t_{exc} and t_{cm} .

The excess time is directly related to the power draw of the vehicle, such that a lower power draw will result in a longer excess time. The charge margin is related to both the power draw and the collected solar power. Decreasing the power draw will increase both the excess time and the charge margin. The charge margin is also increased by having more solar panels, provided that the added weight and drag penalties to the power draw are smaller in magnitude than the increase in collected solar power. Maximizing the charge margin requires a low power draw, therefore it is not expected that there would be a trade-off between t_{cm} and t_{exc} by optimizing one or the other.

This chapter adopts a similar method to maximizing t_{exc} . Each flight is simulated to begin after sunrise with $SOC = 80\%$ to account for the battery charge used to takeoff and climb, as if the vehicle was launched around sunrise. After the sun sets, it is assumed to not rise again so that there is zero irradiance for the remainder of the flight. The flight concludes when the SOC discharges to 30%. The total flight time, t , is the basis of comparison between each UAV, where a longer flight time corresponds to a better configuration. Equation 4.1 summarizes the objective for this chapter, where the design variables are further defined and explained in Section 4.2.

$$\begin{aligned} \max. \quad & f(X) = t \\ \text{w.r.t} \quad & X = \{\text{configuration}, n_{batt}, n_{PV}, \text{PV orientation}, \text{payload}\}^T \end{aligned} \quad (4.1)$$

A sample trace of the SOC, power required, solar power available, and solar power collected is shown in Figure 4.2. The flights assume a constant circular trajectory, which is explained in further detail in Section 4.4 and justified in Chapter 6. The circular trajectory causes the solar panels to cycle from more to less aligned with the incident solar vector, resulting in an oscillatory solar power collected. Note that the solar power is constant at 0 W after the sun sets due to the assumption that the sun does not rise again.

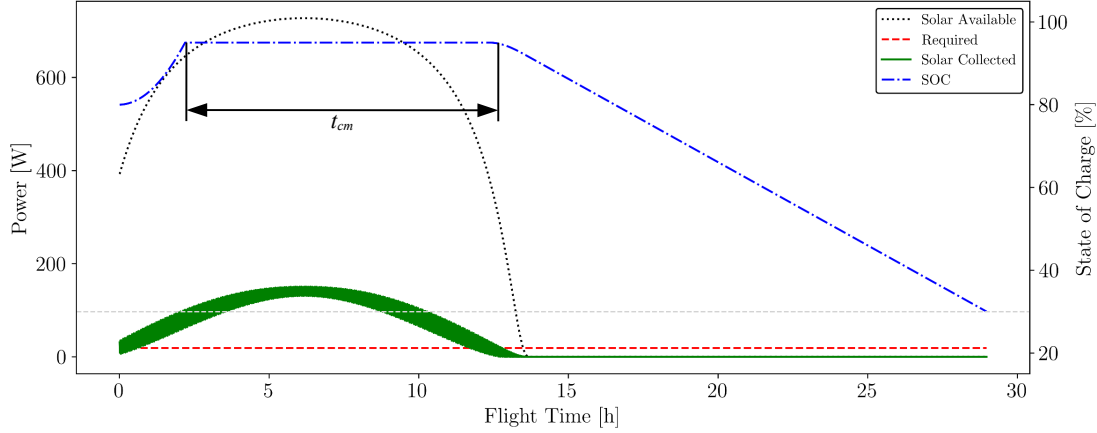


Figure 4.2: Power levels and SOC throughout sample simulated flight of 28.9 h.

4.2 Design Variables

This section presents the different design variables used throughout the parametric analysis. These include the wing and empennage configuration, the storage location of the battery packs and MPPTs, the total number of solar panels, the layout of the solar panels, the number of battery cells in parallel in each battery pack, and the payload of the vehicle. The effects of time of year and latitude will be considered after the selection of a configuration for optimization to reduce the computational expense of repeating the analysis in two additional dimensions.

4.2.1 Configurations

By including the storage location of the battery packs and MPPTs into the number of configurations, there are seven total configurations used in this analysis. Figure 4.3 shows a sample of each, where the differences are found in the battery placement and empennage configuration.

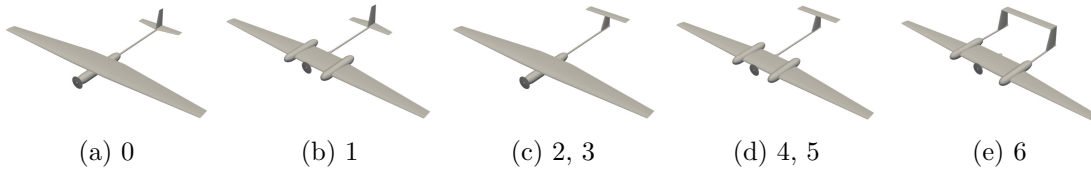


Figure 4.3: Configurations analyzed.

Configuration 0 is a clean-wing conventional tail design with batteries and MPPTs stored in the fuselage. Configuration 1 is a conventional tail design with the batteries and MPPTs stored in nacelles on the wing. Configuration 2 is a clean-wing T-tail design with batteries and MPPTs stored in the fuselage. Configuration 3 is the same as Configuration 2, except it has solar panels embedded in the horizontal stabilizer as well as the wing. Configuration 4 has a T-tail with the batteries and MPPTs stored in

nacelles on the wing. Configuration 5 is the same as Configuration 4, except it has solar panels embedded in the horizontal stabilizer as well as the wing. Configuration 6 is a twin boom configuration with nacelles on the wing for battery and MPPT storage; there are solar panels embedded in the horizontal stabilizer. There is no configuration pair for Configuration 6 without solar panels on the horizontal stabilizer. It is recognized that the added drag from the second tail boom and vertical stabilizer is viable only if the benefits of solar cells on the horizontal stabilizer outweigh the required power penalties.

4.2.2 Battery Packs

In this chapter, the battery packs are assumed to be cube volumes within the fuselage or nacelles according to the respective configuration. Recall that there are two battery packs in a UAV, with m_{batt} cells in series and n_{batt} cells in parallel. The cells in series are fixed at $m_{batt} = 3$. The cells in parallel are a variable in the parametric analysis, where $n_{batt} \in \{1, 2, \dots, 10\}$, which effectively increase the battery capacity, previously defined in Equation 3.44.

4.2.3 Solar Cells

The solar arrays are defined as identical for each battery-MPPT-solar cell circuit. The number of cells and the orientation of the cells are both considered design variables. The number of solar cells considered for the parametric analysis are such that $n_{PV} \in \{36, 42, 48, 54\}$. Recall that one MPPT can accommodate 17 to 36 solar cells. This set of n_{PV} is selected to span the range of possible n_{PV} without significantly increasing the number of UAV configurations to be analyzed.

Orientation

There are several different orientations that the solar panels can be arranged on the wing for a given n_{PV} . All possible orientations are considered such that the leading edge will always have the most number of solar panels. The analysis considers panel layouts of up to 4 panels along the root chord, where each row downstream will not have more panels than the row upstream of it. For configurations with solar cells on the horizontal stabilizer, the total number of solar cells is the sum of cells on the wing, $n_{PV,wing}$, and cells on the horizontal stabilizer, $n_{PV,HT}$. This does not bias the designs towards higher $\mathcal{R}$ wings. It enforces the leading edge to have the maximum number of panels along the wingspan and requires the leading edge to not be swept.

Consider $n_{PV} = 42$ with no solar cells on the horizontal stabilizer. There are 21 cells available for half the wing, where there are several orientations possible. Figure 4.4 shows the possible solar cell positions for half the wing when there are 12 panels along the semi-span. The squares outlined in black indicate that there must be a solar panel in those spots, while the squares outlined in grey indicate that there may be a solar panel in those spots.

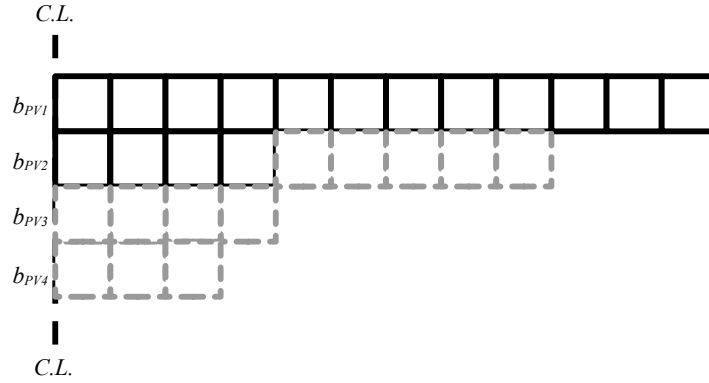


Figure 4.4: Possible solar cell locations for $n_{PV} = 42$, no cells on the horizontal stabilizer, and a semi span of 12 solar cells.

For each row of cells along the semi-span of the wing, labelled b_{PV1} , b_{PV2} , b_{PV3} , and b_{PV4} , the following constraints must always be true:

$$b_{PV1} \geq b_{PV2} \geq b_{PV3} \geq b_{PV4}$$

$$2 \times (b_{PV1} + b_{PV2} + b_{PV3} + b_{PV4}) = n_{PV,wing}$$

And, if b_{PV2} , b_{PV3} , or b_{PV4} are non-zero, they must be ≥ 2 . There will be no gaps between the solar cells in the span-wise direction. The trailing edge row is the aft-most non-zero row of solar cells. For example, if $b_{PV4} = 0$, then b_{PV3} is considered the trailing edge row, b_{PVTE} . All possible combinations of cell orientations are used as inputs for the parametric analysis. The process of generating all possible combinations of solar arrays following these outlined rules is described in Appendix E. The wing geometry is then defined around the solar array, as discussed in Section 4.3.1.

There is significantly less variability considered for the placement of solar cells on the horizontal stabilizer. Figure 4.5 shows the layout options for $n_{PV,HT} \in \{2, 4, 6\}$. To maintain constant circuit properties for each MPPT-battery-solar cell circuit and to not increase the variable size of n_{PV} , only an even number of solar cells are considered for the horizontal stabilizer. A preliminary analysis considered up to two rows of panels on the horizontal stabilizer, which resulted in large stabilizing surfaces and unreasonable static margins. As such, only one row is considered in this analysis.

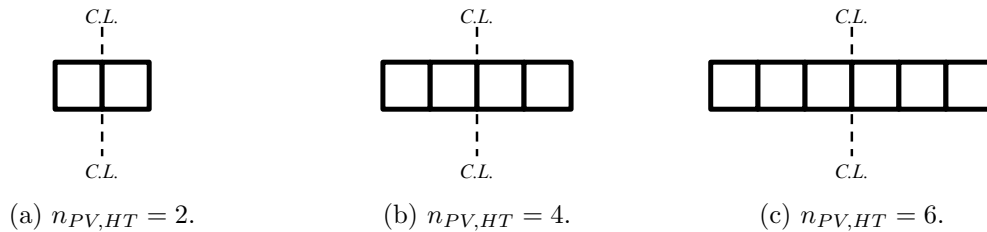


Figure 4.5: Horizontal stabilizer solar cell orientations.

Not all solar cell orientations are compatible with all configurations. The location and inclusion of nacelles is dependent on the orientation of the solar cells. The twin boom in Configuration 6 is dependent on the distance between the nacelles and the solar cell orientation on the horizontal stabilizer. These dependencies, along with other geometric properties defined by the design variables, are discussed in Section 4.3.

4.2.4 Payload

There are a total of 25 560 theoretical UAVs analyzed, which consists of every possible combination of configuration, n_{batt} , n_{PV} , and solar panel layouts. Payload is considered as an additional design variable, where all 25 560 UAVs were analyzed three times, where each analysis uses a different payload case. These payload options were defined in Section 3.3.

4.2.5 Time and Location

After determining the most promising configuration, the effects of time of year and geographical location are considered. Kingston, ON and Inuvik, NT were considered for the 21st day of each month throughout the year. There are significant variations in solar irradiance throughout the year, with sunrise and sunset occurring at a different time in each month at both locations. The initial analysis with all configurations simulates the flight beginning at 0700 local to Kingston. This allows for the charge margin and flight time to be directly compared to each other, as there is the same time between sunrise and launch for every vehicle. Changing the time of year makes it far more difficult to create a standard between all months that allows for the flight time and charge margin to be compared across different months, especially considering the variations in daylight hours in a day. The data collected for one month can be compared to that month in that location but cannot be directly compared to a different month or location. Instead, it is the trends at each month and location that are of significance to determine which vehicle characteristics perform better depending on location or time of year. To create consistency between months, all flights will be simulated with a start time of 1100 UTC, which is the same time as 0700 EDT, 0600 EST, 0500 MDT, and 0400 MST.

4.3 UAV Definition

A UAV configurator has been developed to generate the geometries of a fixed-wing UAV with basic properties as input. The configurator takes wing characteristics, empennage characteristics, the inclusion of nacelles, the size of battery packs, and the inclusion of MPPTs as inputs, then outputs geometry in a format usable by *pyACDT*.

This section explains the geometric definition of a UAV given the input variables for the parametric comparison and some of the inner workings of the developed UAV configurator. The compatibility of configurations and solar cell orientations will be discussed. The geometry definition is necessary for the application of *pyACDT* for the sizing, aerodynamics, and stability of the vehicle before moving on to the basis of comparison in the next section.

4.3.1 Wing Geometry

This subsection discusses the wing geometry and configuration compatibility based on the placement and orientation of the solar cells.

The orientation of the solar cells on the wings is used to define the wing geometry. If $b_{PV1} = b_{PVTE}$, there is only one wing segment, the wing is rectangular, and there are no nacelles. Otherwise, the semi-span of the inner wing is composed of b_{PVTE} solar cells, with the outer wing segment having $b_{PV1} - b_{PVTE}$ cells along its length. The inner wing segment has a constant chord length and zero taper, independent of nacelles on the wing.

Nacelles may be included if $b_{PVTE} \leq 4$. The taper, λ , of the outer wing segment is defined to minimize the surface area without solar cells without cutting through any area covered by panels. The leading and trailing edges are straight, with zero sweep on the leading edge. In some cases, the outer wing tip chord is increased to prevent solar cells from being cut through by the trailing edge or to ensure that $\lambda \geq 0.4$ to help prevent tip stall and maintain the assumed and ideal elliptical lift distribution. Where nacelles are included, there may be a discontinuity between the inner and outer wing properties at the tip and root, respectively. With nacelles, the outer wing root chord will not be larger than the chord of the inner wing.

Figure 4.6 shows four sample wing geometries for different solar cell orientations, with the trailing edge of the outer wing segment shown in red. Figures 4.6a and 4.6b show how the taper of the outer wing changes with the inclusion of nacelles, where Figure 4.6b has a small wing surface area. Figures 4.6c and 4.6d show how ℓ_i and ℓ_j influence the taper. Where $\ell_i < \ell_j$, the outer wing is tapered as in Figure 4.6c. Where $\ell_i \geq \ell_j$, the outer wing is untapered, as in Figure 4.6d. $\ell_i = \ell_j$ is untapered to reduce the stresses in the solar cells when they are installed in the wing, as they are required to bend to fit the curvature of the airfoil in the chord-wise direction.

Figures 4.6a, 4.6b, and 4.6d demonstrate that the tip chord may be longer than the number of panels in the chord-wise direction. If $\lambda < 0.4$ after constructing the trailing edges in Figure 4.6, the tip chord will be extended so that $\lambda = 0.4$.

The parametric comparison of UAVs varies the span, chord, and taper of the inner and outer wing segments. Other wing characteristics such as dihedral, airfoil properties, incidence angle, and geometric twist remain constant for each vehicle in the analysis. These variables will be considered in the optimization stage in Chapter 5, but offer too much variability in aerodynamic performance to consider for the parametric comparison. Flight control surfaces have not been part of the solar cell orientation considerations for this study.

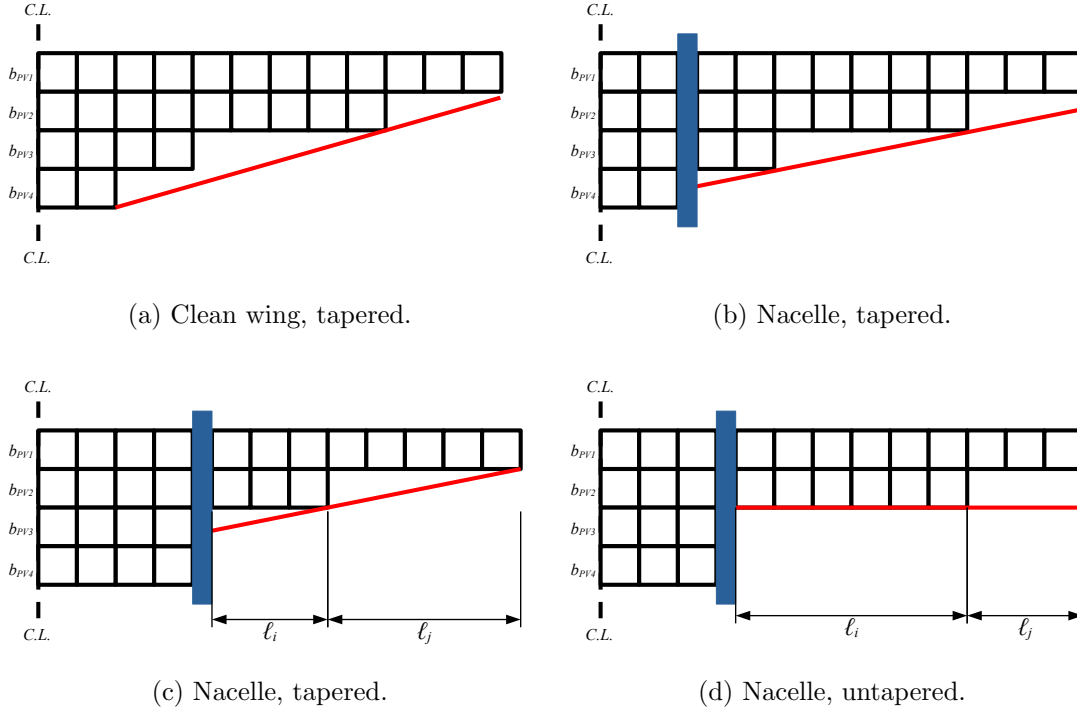


Figure 4.6: Sample wing geometries with $b_{PV1} = 12$, $n_{PV,wing}/2 = 27$.

4.3.2 Empennage

This subsection discusses design considerations, variable compatibility, and the sizing of the empennage of each UAV for the parametric comparison.

Configuration Compatibility

The solar cell orientation and inclusion of nacelles dictates which empennage can be included. The conventional tail and T-tail are compatible with all cell orientations and possibilities of nacelles. The T-tail has the option for solar cells embedded on the horizontal stabilizer, regardless of nacelles being included or not.

Configuration 6, the twin boom, has more stringent requirements. Nacelles must be included and there must be solar cells on the horizontal stabilizer. The vertical stabilizers are not angled towards or away from the centre line of the aircraft and are in line with the nacelles. As a result, the combinations of horizontal stabilizer solar cell orientations in Figure 4.5 can only be considered if the span of the inner wing segment is equal to or greater than the span of the cells on the horizontal stabilizer.

Sizing

Proper sizing of the empennage of any aircraft is an intensive iterative process which considers the interdependence of the centre of gravity and static and dynamic stability characteristics of an aircraft. There is no method to instantly and correctly design the

empennage of an aircraft, but there are methods to approximate the size of the surfaces required based on historical data.

This analysis sizes the empennage using volume coefficients, as described by Raymer in [47]. The governing equations are set out in Equation 4.2, where S is the surface area, c is the respective surface volume coefficient, $\bar{c}_{wing}$ is the mean aerodynamic chord of the wing, and L is the moment arm from the aerodynamic centre of the wing to the aerodynamic centre of the respective lifting surface.

$$S_{VT} = \frac{C_{VT} b_{wing} S_{wing}}{L_{VT}} \quad (4.2)$$

$$S_{HT} = \frac{C_{HT} \bar{c}_{wing} S_{wing}}{L_{HT}} \quad (4.3)$$

For the conventional and T-tail empennages, the tail boom is assumed to be 3.0 ft long which is similar in size to other small LALE solar UAVs. The aerodynamic centre of the wing and empennage surfaces are assumed to be at the quarter chord of each surface. The moment arm between aerodynamic centres is assumed to be 65% of the total length of the aircraft as per the guidance in [47]. With the calculated surface areas of each stabilizer, the chord length and span is defined by assumed taper and aspect ratios. These assumed values are summarized in Table 4.1, which are in within the range of typical values found in Raymer [47] for sailplanes. Configurations with the conventional tail all have these geometric properties.

Table 4.1: Assumed empennage volume coefficients, taper ratios, and aspect ratios [47].

	Horizontal	Vertical
C	0.5	0.02
λ	0.6	0.6
$\mathcal{R}$	5.0	2.0

The T-tail follows a slightly different method of generating λ and $\mathcal{R}$. The volume coefficients are used to determine the required surface areas. The areas are then reduced by 5% to account for the increased effectiveness from end-plate effects on the vertical stabilizer and being in cleaner flow, further from the down-wash of the wing on the horizontal stabilizer. The horizontal stabilizer has the same $\mathcal{R}$ as in Table 4.1, but $\lambda_{HT} = 1$ to account for the ability to embed solar cells on the surface. The tip chord of the vertical stabilizer is the same as the chord of the horizontal stabilizer, and the remaining geometry is defined by λ_{VT} as per Table 4.1. For Configurations 3 and 5 with solar cells on the horizontal stabilizer, $\mathcal{R}_{HT}$ and S_{HT} will be adjusted to accommodate the solar cell orientation if they are otherwise too restrictive.

The twin-boom follows a similar logic to the T-tail. S_{VT} is the total area of both vertical stabilizers, therefore the tail boom length is shorter than the conventional and T-tail configurations at 2.5 ft to result in larger S_{VT} and S_{HT} . The span of the horizontal stabilizer is set by the distance between each nacelle. $\lambda_{HT} = 1$ and $\mathcal{R}_{HT}$ is not a fixed value, allowing the chord to vary to allow for S_{HT} to occur. If the chord for S_{HT} does not allow for the required solar cells to fit on the surface, S_{HT} is increased. The tip

chord of the vertical stabilizer is the same as the chord of the horizontal stabilizer. λ_{VT} is in accordance with Table 4.1 and $\mathcal{R}_{VT}$ is subject to the span required to achieve the total S_{VT} .

If the surface area of the horizontal stabilizer must be increased to accommodate the number of solar cells on the surface, the moment arm is recalculated using the tail coefficients and the new area. The vertical stabilizer area is recalculated using the new moment arm. If the resized geometry results in a negative tail length or a vertical stabilizer root chord larger than the tail length, that individual geometry is considered failed and is not part of the analysis.

This singular resizing iteration helps prevent unreasonably high static margins. The primary focus of the parametric analysis is to compare the aerodynamic performance and energy capacity of each aircraft. Section 4.5 shows that this sizing method results in statically stable vehicles in pitch, which is considered sufficient for this stage of analysis. Stability derivatives will be considered in the optimization stage of the design in Chapter 5.

There are some geometries which result in very low $\mathcal{R}_{VT}$ due to the chord length being defined by the horizontal stabilizer and λ_{VT} . For the T-tail and twin boom configurations, where $\mathcal{R} < 0.5$, λ_{VT} is redefined to $\lambda_{VT} = 0.9$ to increase the span of the vertical stabilizer. Otherwise, many of the resulting geometries for Configurations 3, 5, and 6 have very long root chords and short spans, which is unrealistic in any aircraft.

This section defined geometric properties and relationships for the wings, empennage, and fuselage and nacelles for the parametric analysis. Considering the design variables from Section 4.2 and how the geometry is adjusted with each variable, there are a total of 25 560 theoretical UAVs for a given payload to be analyzed and compared.

4.4 Flight Assumptions

This section outlines the specific assumptions made for the simulated flight for each UAV in the parametric comparison. It is recognized that the trajectory of the vehicle has a significant influence on a vehicle's endurance. The flight assumptions are chosen to reflect probable flight conditions. The results of the analysis may be influenced by these assumptions, favouring configurations that are better suited to the assumed flight conditions.

Several trajectory optimization studies for solar-powered aircraft result in circular or near circular trajectories, depending on initial conditions and whether or not wind effects are considered [68, 78, 80, 84]. A circular trajectory results in higher solar capture when the aircraft is banked to align the solar panels to the incident rays of the sun and decreased capture when the vehicle is banked away from the sun. The bank angle, and therefore turning radius, of the vehicle can be optimized for the time of day to further increase the average solar power capture in one rotation of the vehicle [68]. The parametric analysis assumes a circular trajectory with a turning radius of 180 ft, approximately the radius of two neighbouring soccer fields, at an altitude of 2000 ft MSL.

This turning radius is not optimized for any particular configuration, but is reflective of a potential flight test location in Kingston, ON for flight testing in future stages of this project specifically for maintaining visual line of sight.

Assuming that $T = D$, it is important to fly at the minimum power speed, $V_{P_{min}}$. The minimum power speed is defined by Equation 4.4. Note that $V_{P_{min}}$ is slower than the minimum drag speed by a factor of $3^{-1/4}$.

$$V_{P_{min}} = \sqrt{\frac{2L}{\rho S} \sqrt{\frac{K}{3C_{D_0}}}} \quad (4.4)$$

The bank angle required to maintain a set turning radius at $V_{P_{min}}$, r , is defined by Equation 4.5, where g is the gravitational constant.

$$\phi = \arcsin\left(\frac{V_{P_{min}}^2}{rg}\right) \quad (4.5)$$

In a turn, $L = W/\cos\phi$. Without increasing the airspeed, the aircraft will lose altitude if it maintains its turn at $V_{P_{min}}$. To maintain altitude, the aircraft must increase its airspeed. The minimum power speed to maintain altitude in a turn is given by Equation Set 4.6, where n_z is the g -loading experienced in a level turn since $n_z = 1/\cos\phi$.

$$\begin{aligned} V_{P_{min},turn} &= \sqrt{\frac{2}{\rho S} \frac{W}{\cos\phi} \sqrt{\frac{K}{3C_{D_0}}}} \\ V_{P_{min},turn} &= \sqrt{\frac{2W}{\rho S} \sqrt{\frac{K}{3C_{D_0}}}} \sqrt{n_z} \\ V_{P_{min},turn} &= V_{P_{min}} \sqrt{n_z} \end{aligned} \quad (4.6)$$

The rate of turn is:

$$\dot{\psi} = \frac{V_{P_{min},turn}}{r} \quad (4.7)$$

In summary, each simulated flight begins at 0700 on 1 July 2025 with $SOC = 0.8$ as a representation of the energy used for takeoff and climb. The vehicle flies in a circular trajectory of 180 ft at 2000 ft MSL at its minimum power speed in a turn. The weight and aerodynamic characteristics are calculated from the aircraft's geometric definition, which dictates the power required to maintain flight. The flight is governed by the simplified states in Equation Set 4.8, which ends when the battery reaches $SOC = 0.3$. After the sun sets, it is assumed that it does not rise again the next morning to determine the total flight time as demonstrated in Figure 4.2.

$$\begin{aligned} \dot{\psi} &= \frac{V_{P_{min},turn}}{r} \\ \dot{x} &= V_{P_{min},turn} \cos\psi \\ \dot{y} &= V_{P_{min},turn} \sin\psi \\ \dot{SOC} &= \frac{P_s - P_p - P_{pl}}{E} \end{aligned} \quad (4.8)$$

4.5 Results

Based on modifying the configuration, the number of batteries in parallel, and the number and orientation of the solar cells, there are 25 560 distinct theoretical UAVs generated. Of these, 814 are unfeasible designs and were unable to be analyzed due to limitations in the aerodynamic and weight sizing analyses. This section presents the results of the parametric comparison. Many of the results are shown for a specific set parameter, *i.e.*, $b_{PV1} = 12$, with the same results at different b_{PV1} shown in Appendix F.

First and foremost, the static margin is plotted in Figure 4.7 by configuration. The markers indicate the total number of solar panels, n_{PV} , and are offset on the x -axis to better visualize the data at each configuration. Offsetting the data along the x -axis will reoccur in other plots for variables that are integers. Figure 4.7 shows the effectiveness of the empennage sizing method for producing statically stable vehicles in pitch. Initial empennage sizing did not include a resizing of the moment arms if the horizontal surface was required to increase to accommodate the solar cells which resulted in static margins up to 82% of the mean aerodynamic chord. Although some of the resulting UAVs through the analysis have large static margins, the sizing method is consistent, high manoeuvrability is not a requirement, and the final design will be properly sized through the optimization with stability derivatives in consideration. Although the dynamic flight responses are less of a concern with UAVs in contrast to commercial aviation vehicles and stronger pitch responses can be tolerated, it is desirable to have handling qualities that require less control surface deflections to minimize the active and sustained actuation power draw. In the case of a solar vehicle, all sources of power draw will contribute to reducing the SOC and therefore flight time of the vehicle.

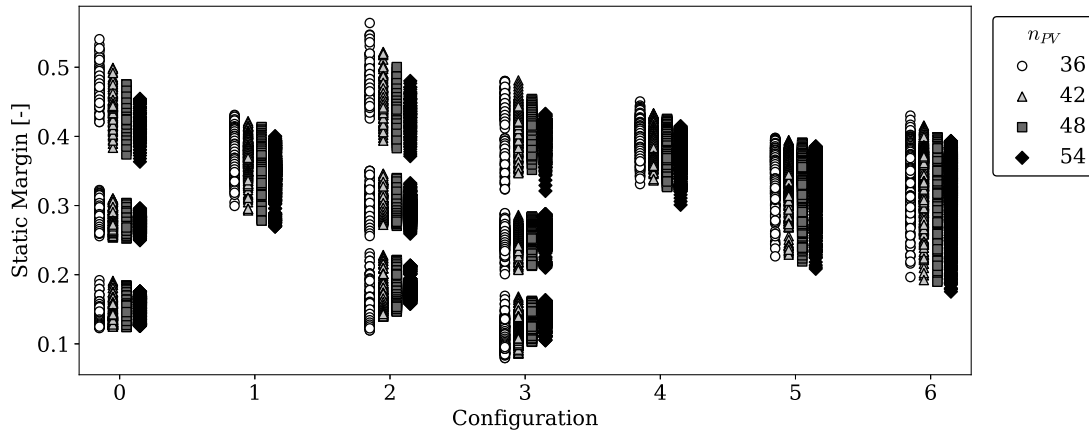


Figure 4.7: Static margin of each configuration, identified by n_{PV} .

Figure 4.8 shows the weight of the UAV as a function of aspect ratio, with the markers indicating the number of batteries in parallel, n_{batt} . At a fixed $\mathcal{AR}$, the weight is a function n_{batt} . At a fixed n_{batt} , the weight appears to be a function of $\mathcal{AR}$, which follows the weight sizing method in Section 3.2.3. A larger wingspan, which corresponds to a larger $\mathcal{AR}$ in this analysis, requires more structural weight to support the lift and weight of the vehicle. The vehicle with the longest total flight time is indicated in red.

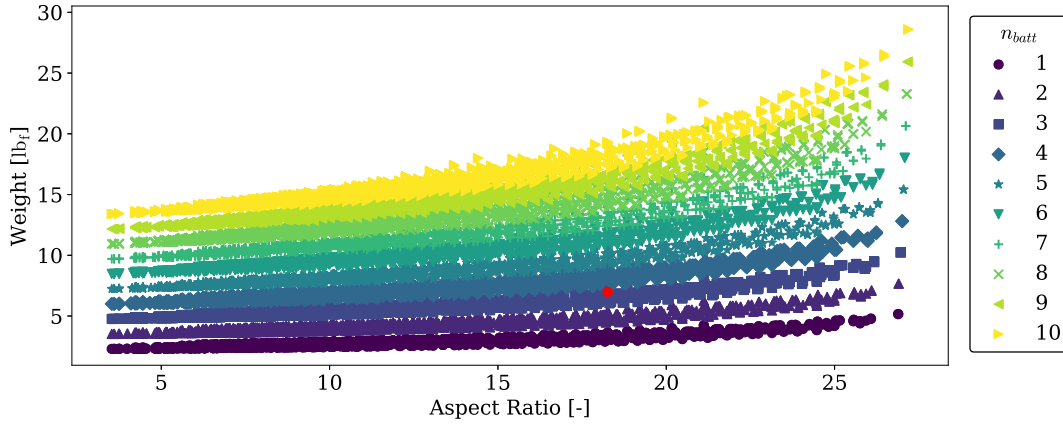


Figure 4.8: Effects of $\mathcal{R}$ on total weight, identified by n_{batt} .

Figure 4.9 plots a subset of the data in Figure 4.8, with configurations where $b_{PV1} = 12$, *i.e.*, 12 solar cells on the leading edge of the semi span. Variations in wingspan in this vehicle are found for configurations with nacelles that increase with increasing n_{batt} , which varies between 10.6 ft and 11.7 ft. Figure 4.9 shows that the weight is more accurately driven by n_{batt} and the wingspan, rather than $\mathcal{R}$. The same trend is shown for other b_{PV1} in Figures F.1 through F.3 in Appendix F. For a given clean wing configuration, the increases in n_{batt} increase the weight which appear as vertical lines in Figure 4.9. For a given nacelle configuration, increasing n_{batt} increases the diameter of the nacelles which effectively increases the wingspan and therefore the aspect ratio. The increase in n_{batt} for a nacelle configuration appears as an increase in weight and aspect ratio, most evident in the two configurations with the highest aspect ratios, which appear to shift right (increasing in aspect ratio) away from the remainder of the data.

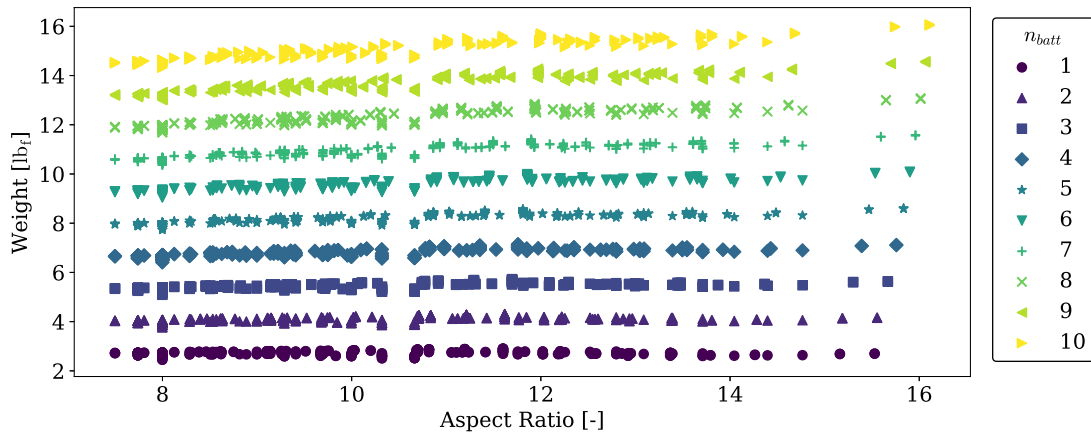


Figure 4.9: Effects of $\mathcal{R}$ on total weight for $b_{PV1} = 12$, identified by n_{batt} .

Figure 4.10 shows the flight time as a function of the number of panels along the leading edge of the semi-span of the wing b_{PV1} , with markers identifying the number of batteries

in parallel and expanded along the x -axis. With increasing panels along the semi-span, the flight time increases to a maximum of 32.4 h with $b_{PV1} = 20$ and $n_{batt} = 3$. For $b_{PV1} \leq 16$, $n_{batt} = 2$ is optimum. For all larger wingspans, $n_{batt} = 3$ is optimum.

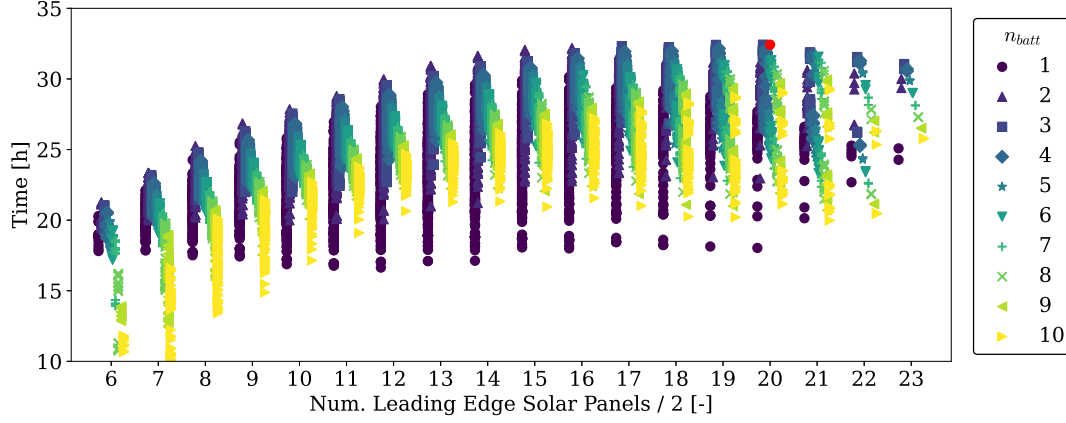


Figure 4.10: Flight time as a function of b_{PV1} , identified by n_{batt} for zero payload.

The maximum flight time corresponds to a UAV with a wingspan of 17.7 ft and $\mathcal{R} = 18$, shown in red in Figures 4.8 and 4.10, presents aeroelastic and ground handling challenges. It is desirable to construct a UAV that can be hand-launched and is easily transported. A wingspan around 10 ft is considered the upper limit to abide by these limitations, which corresponds to a b_{PV1} of 11 or 12, depending on n_{batt} and the size of any nacelles. To better understand the aerodynamic and battery capacity tradeoffs at a desirable design point, the remainder of the results will show trends for the UAV configurations where $b_{PV1} = 12$ unless otherwise specified.

The charge margin, defined in Figure 4.2, is important to provide robustness to disturbances in available solar power throughout the day. The charge margin is properly defined by the second solar cycle, where it is not affected by the initial SOC at the beginning of the flight. Figure 4.11 shows the relationships of n_{PV} and n_{batt} on the charge margin, where the charge margin is taken to be the time at maximum SOC in the first and only cycle for the simulated flight. The markers indicate n_{batt} and they are offset on the x -axis. At lower n_{batt} , the number of solar cells has little influence on the charge margin. With more solar cells, the batteries are able to charge faster and remain at maximum charge for longer. Therefore, as n_{PV} increases, the charge margin's sensitivity to n_{batt} decreases; for a given n_{batt} , the charge margin increases with increasing n_{PV} . This trend is consistent across other wingspans, shown in Figures F.4 to F.6 in Appendix F. The same trend occurs when payload is considered, where the decrease in charge margin is due to the higher power consumption of the payload, resulting in a longer time to charge the batteries, as per Figures F.7 through F.10. Configuration has little to no influence on the charge margin.

Figure 4.12 shows the same plot as Figure 4.10, except with payload option 2 as detailed in Table 3.5. The maximum flight time at each b_{PV1} occurs at a higher n_{batt} than

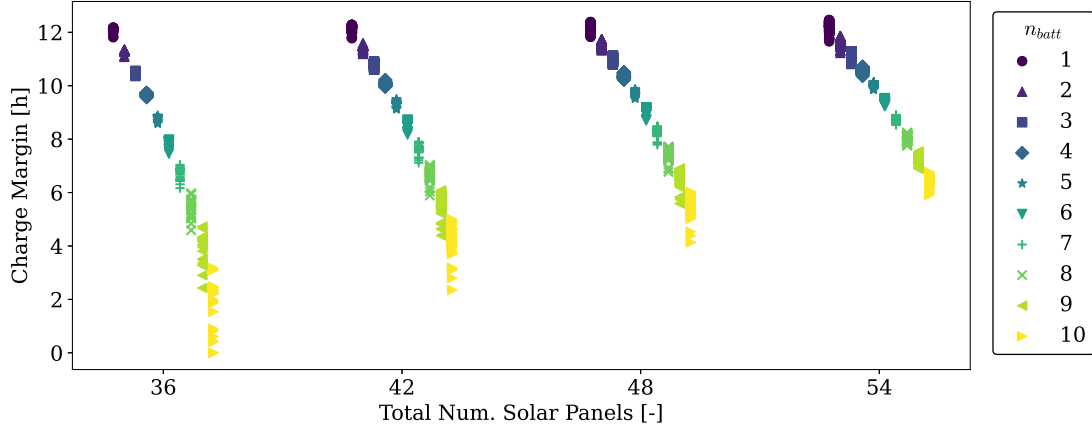


Figure 4.11: Charge margin as a function of n_{PV} , identified by n_{batt} for $b_{PV1} = 12$.

compared to no payload. For reference, the maximum flight time with payload option 2 corresponds to a wingspan of 17.7 ft, $\mathcal{R}$ is lower at 13.7, and n_{batt} increases to 8. For b_{PV1} , the optimum n_{batt} increases from 2 to 6 by adding payload. Adding payload increases the optimum n_{batt} for which increasing n_{PV} increases the charge margin. The more payload is added and the power draw becomes larger, more solar energy needs to be collected to maintain the energy balance and prolong overnight flight. Similarly, larger energy capacity and solar cell area may prove essential for prolonging flight time during non-summer months where the daylight hours are shorter and the solar irradiance is weaker.

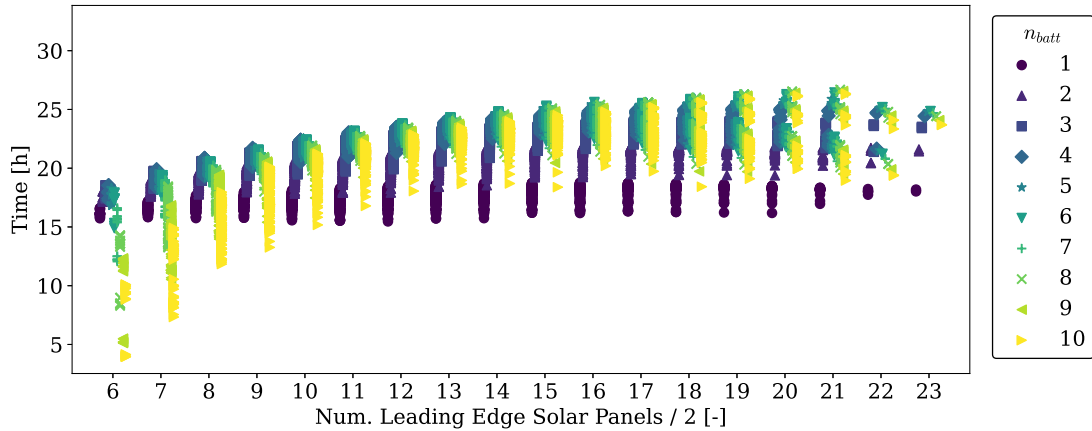


Figure 4.12: Flight time as a function of b_{PV1} , identified by n_{batt} with payload option 2.

Figure 4.13 further shows the effects of payload on the optimal n_{batt} . The flight time is shown as a function of n_{batt} , where the markers indicate the payload option and are expanded along the x -axis. Increasing the power consumption of the payload increases the optimum n_{batt} for LALE solar-powered flight, which is consistent at different wingspans, shown in Figures F.11 through F.13.

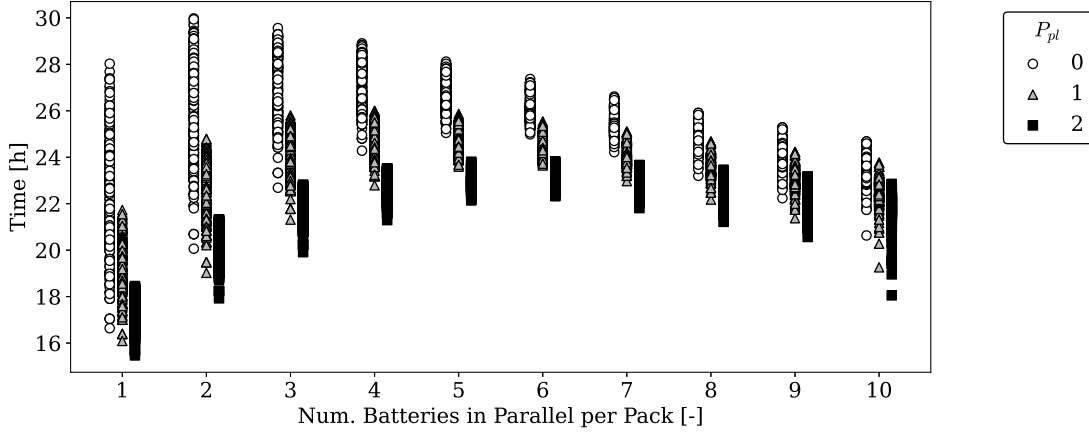


Figure 4.13: Flight time as a function of n_{batt} , identified by payload option at $b_{PV1} = 12$.

Figure 4.14 offers another means of visualizing how higher n_{PV} performs better with higher n_{batt} . Time is shown as a function of n_{batt} , where the colour indicates maximum aerodynamic efficiency, and the markers indicate n_{PV} which are expanded along the x -axis. Another key result is shown in the maximum aerodynamic efficiency. As n_{batt} increases, it is not the highest aerodynamic efficiencies that are achieving the longest flight times. As suspected, there is an inflection where prioritizing energy capacity over aerodynamic efficiency results in longer flight times. This occurs at higher n_{batt} , which is necessary when considering payloads. These trends are consistent at different b_{PV1} and payload cases, shown in Figures F.14 through F.20.

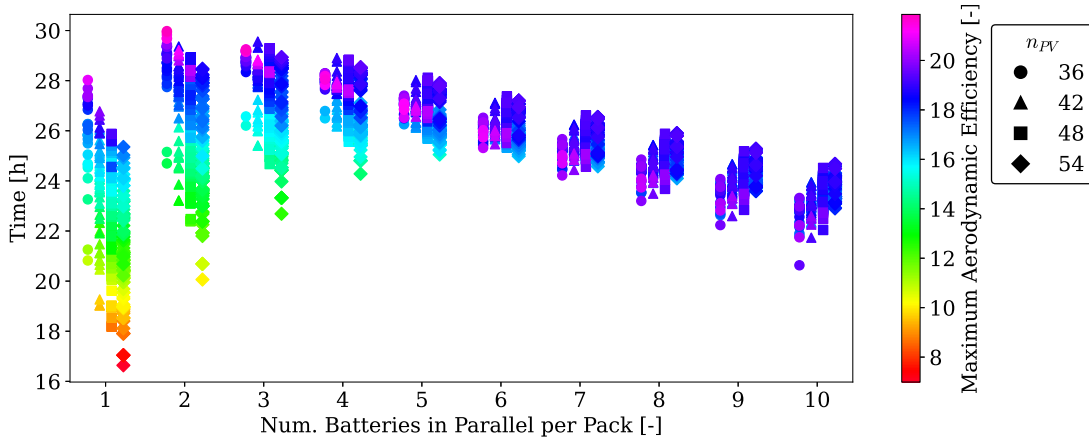


Figure 4.14: Flight time as a function of n_{batt} , identified by n_{PV} for zero payload at $b_{PV1} = 12$.

One of the main objectives of this analysis is to determine which configuration is the best suited for LALE solar-powered flight. Figure 4.15 shows flight time as a function of n_{batt} for $b_{PV1} = 12$ and no payload. The colour bar indicates $\mathcal{R}$ and the markers are expanded on the x -axis to show the configuration. This should not be confused with Figure 4.14, where the markers and colour scale represent different results. An optimum occurred at $b_{PV1} = 12$, $n_{batt} = 2$, with the clean wing configurations out-performing

those with nacelles. The trends in $\mathcal{R}$ are of particular interest in this figure. At the optimum for $b_{PV1} = 12$, the high $\mathcal{R}$ clean wing configurations supports the pursuit of glider-like configurations for attempting flight endurance records at low altitudes, which has been demonstrated by several platforms [1, 10, 12, 34]. However, as n_{batt} increases, the larger $\mathcal{R}$ vehicles do not perform as well as those with lower $\mathcal{R}$. This is due to the smaller chord resulting in a lower Reynold's Number, higher skin friction coefficient, and therefore in higher parasitic drag, which becomes increasingly more sensitive with increasing fuselage and nacelle diameters with increasing n_{batt} . This corresponds to the higher aerodynamic efficiencies not performing as well in the larger n_{batt} values in Figure 4.14. This is demonstrated across all b_{PV1} , supported by Figures F.21 through F.23, as well as with payload option 2 shown in Figures F.24 through F.27. The lower $\mathcal{R}$ vehicles performing well at higher n_{batt} suggests that applications with higher payload requirements and a larger power draw may be better suited to vehicles that deviate from a glider wing, where the higher weight and decrease in aerodynamic efficiency are compensated for by higher energy capacity.

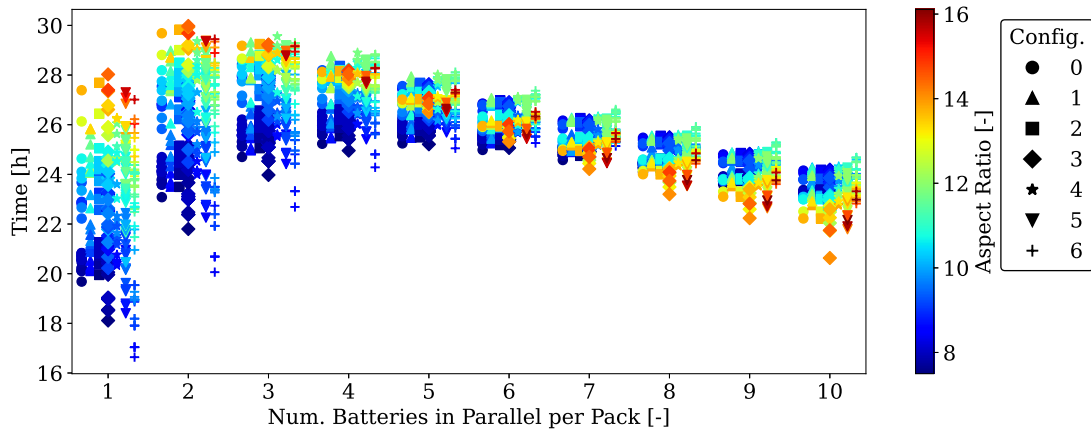


Figure 4.15: Flight time as a function of n_{batt} , coloured by $\mathcal{R}$, identified by Configuration for no payload and $b_{PV1} = 12$.

In Figure 4.15, the configurations perform more similarly to each other with increasing n_{batt} . This does not follow a consistent trend with different wingspans, suggesting a more intricate relationship between n_{batt} and b_{PV1} . Some combinations favour the clean wing, some favour the nacelles, and others perform too similarly to have an obvious distinction. Figure 4.16a shows which wing configuration is better performing as a function of b_{PV1} and n_{batt} with no payload, with 80 different combinations of b_{PV1} and n_{batt} . Of these 80 combinations, 41 are better suited with clean wing configurations, 17 are better suited with the nacelles, and 22 are inconclusive. Figure 4.16b shows the better performing configurations for payload option 2. More of the combinations become inconclusive, indicating that the power draw of the payload has a larger influence on the resulting flight time than the aerodynamic differences between the configurations at the respective wingspan and number of batteries.

With the objective of record setting for a hand-launchable solar UAV, results show that prioritizing aerodynamic efficiency is the best way to achieve this. The clean wing configurations out-perform those with nacelles, where higher $\mathcal{R}$ is more favourable.

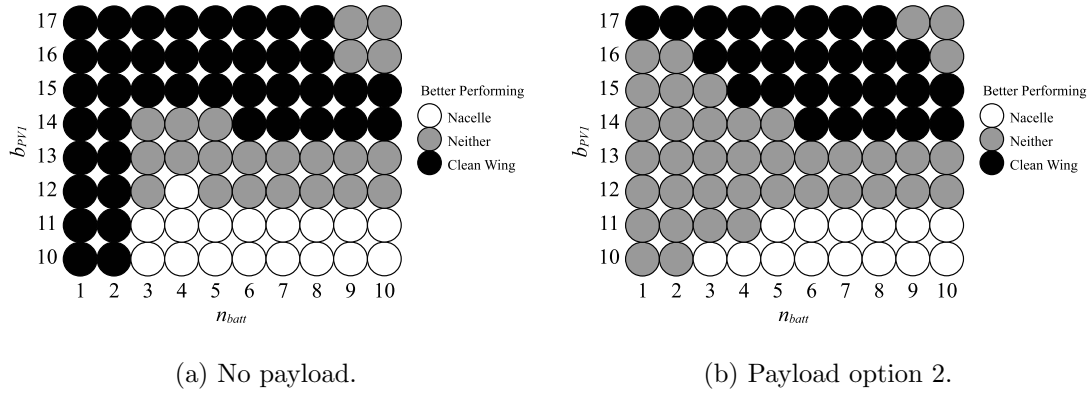


Figure 4.16: Optimal wing configuration based on b_{PV1} and n_{batt} .

With the optimum at $n_{batt} = 2$, the number of solar cells has a stronger influence on the aerodynamics than the charge margin and energy balance because the wing geometry is constructed around the solar array. Therefore, increasing the number of solar cells at the expense of aerodynamic efficiency is not directly beneficial to achieving LALE solar-powered flight. From these results, the T-tail proves to be most effective at achieving long flight times. The horizontal stabilizer encounters cleaner airflow above the wing, resulting in improved aerodynamic properties which gives it the advantage over the conventional tail. Configuration 3 appears suited to achieving a higher flight time than Configuration 2 in Figure 4.15 makes it easy to favour the installation of solar cells on the horizontal tail. However, consider Figure 4.17 which shows the UAVs for Configurations 2, 3, 4, and 5 at $b_{PV1} = 12$, $n_{batt} = 2$, with the flight time as a function of the number of panels on the horizontal tail. Looking at these configurations more closely, there is not a strong indication that either option is better. This is observed across multiple wingspans with and without payload, shown in Figures 4.17 through F.34. Adding solar cells to the horizontal stabilizer decreases the number of cells on the wing, thereby decreasing the mean chord, and increasing the aspect ratio. This explains the improvements in flight time for the shorter flights at each respective number of panels on the horizontal tail, but does not offer enough improvement for the vehicles already achieving high flight times. Manufacturing becomes the next deciding factor in which configuration to pursue. Including solar cells on the horizontal stabilizer increases complexity in the wiring of the solar array, requiring additional connections from the wing to the empennage. Solar panels on the horizontal stabilizer adds more geometrical constraints to the surface, which may increase the difficulty in achieving a solution that also satisfies stability requirements during the optimization. To reduce these challenges and simplify the assembly of the vehicle, Configuration 2 is the most promising configuration.

When considering other applications for solar UAVs where a payload is essential, the best configuration for maximizing flight time is more difficult to discern. From Figure 4.16b, there is no discernible advantage over a clean wing vs nacelle configuration for $b_{PV1} = 12$ to maintain hand-launching. There is no discernible advantage over including solar cells on the horizontal stabilizer similar to the no payload case, which

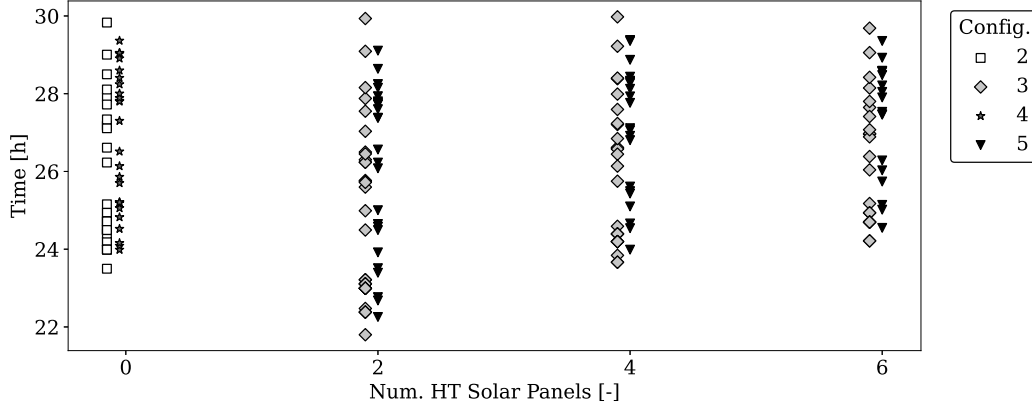


Figure 4.17: Flight time as a function of the number of solar cells on the horizontal stabilizer for Configurations 2, 3, 4, and 5, where $b_{PV1} = 12$ and $n_{batt} = 2$, with no payload.

justifies ruling out those configurations for simplifying manufacturing. Regardless of payload, the T-tail offers an aerodynamic advantage over the conventional tail as a result of the horizontal stabilizer being in cleaner flow. The T-tail with no solar cells on the horizontal stabilizer, Configuration 2, remains the most promising, including at larger wingspans.

4.5.1 Effects of Time and Location

The analysis up to this point has considered only 1 July 2025 in Kingston, ON. Having established that a T-tail empennage is the most promising at LALE solar-powered flight, Configuration 2 and 4 are analyzed at two locations at varying times of year. To reduce computational expense, only configurations with $b_{PV1} = 12$ and $b_{PV1} = 17$ are considered. $b_{PV1} = 12$ corresponds to the limit for a hand-launchable vehicle, while $b_{PV1} = 17$ is the largest wingspan without a significant reduction in data points due to failed geometric or aerodynamic analysis.

Shown previously in Figures 4.10 and 4.12, $b_{PV1} = 17$ is able to achieve longer flight times than $b_{PV1} = 12$. This remains true at different times of the year and latitude, shown in Figures 4.18 and 4.19, where the flight time is a function of the time of year, with the colour indicating b_{PV1} and the markers indicating n_{batt} and offset on the x -axis. Figure 4.18 corresponds to Kingston and Figure 4.19 corresponds to Inuvik. Note that all the flights are simulated to start at the same time of day, regardless of the time of year. The varying sunrise time makes it not possible to compare one month to another, or one location directly to the other. However, the overall trends in flight time can be compared.

Months 6 and 12 show that the optimum n_{batt} is independent of the time of year, which is further supported by the other months shown in Figures 4.18 through 4.21 in Appendix F. When considering the effects of payload, month 12 in Kingston and months 3 and 9 in Inuvik in Figures 4.20 and 4.21, respectively, do not suggest that the optimum n_{batt} is constant. The more drastic decrease in flight time with increasing n_{batt} in

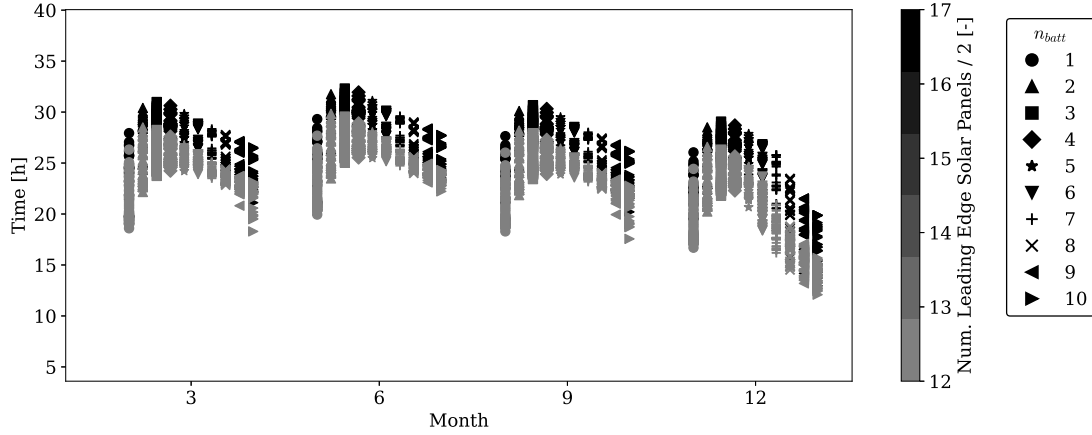


Figure 4.18: Flight time as a function of time of year, identified by n_{batt} for zero payload in Kingston, ON.

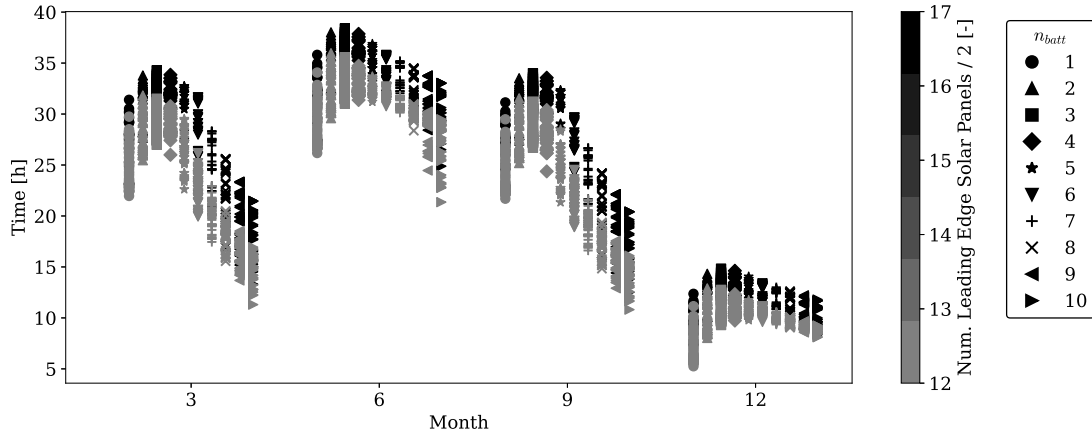


Figure 4.19: Flight time as a function of time of year, identified by n_{batt} for zero payload in Inuvik, NT.

these months is a result of the batteries not achieving a full charge, which is partially a result of the flight beginning before sunrise, demonstrated in Figure 4.22. Figure 4.22 shows the SOC of a sample vehicle with $b_{PV1} = 12$, $n_{batt} = 3$, payload option 2, time 0 corresponding to 1100 UTC, and flown in Inuvik. At higher n_{batt} , the required power increases, which increases the discharge rate and decreases the charging rate of the batteries, resulting in shorter flight times, unable to demonstrate the same optimum n_{batt} as the other months. If trying to widen the window in the year for which continuous flight is possible in higher latitudes, the optimal n_{batt} may shift towards lower values to include the months closer to the equinoxes, as indicated by the maximum flight times in months 3 and 9 in Figure 4.21. Note that the very low flight times relative to the other configurations in months 3 and 9 in Figure 4.21 are caused by those configurations not having enough energy capacity for their respective power draws to maintain flight before sunrise or being able to recharge.

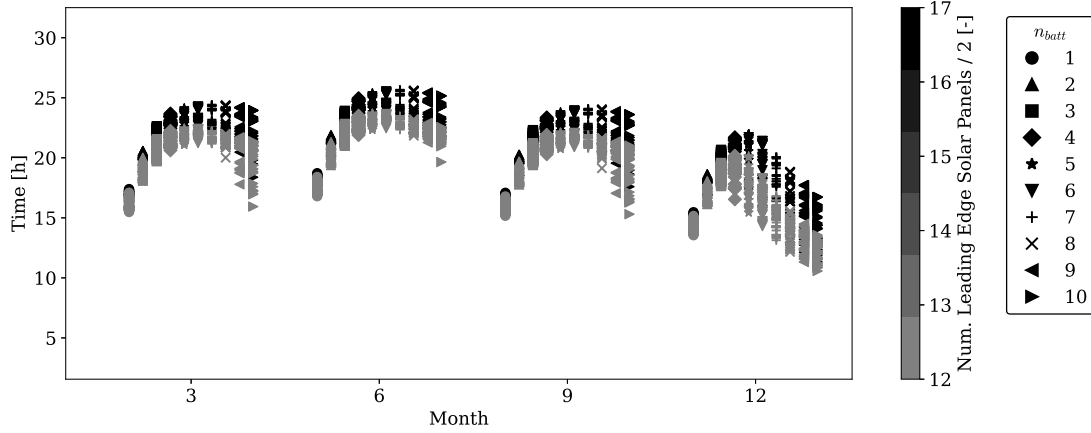


Figure 4.20: Flight time as a function of time of year, identified by n_{batt} for payload option 2 in Kingston, ON.

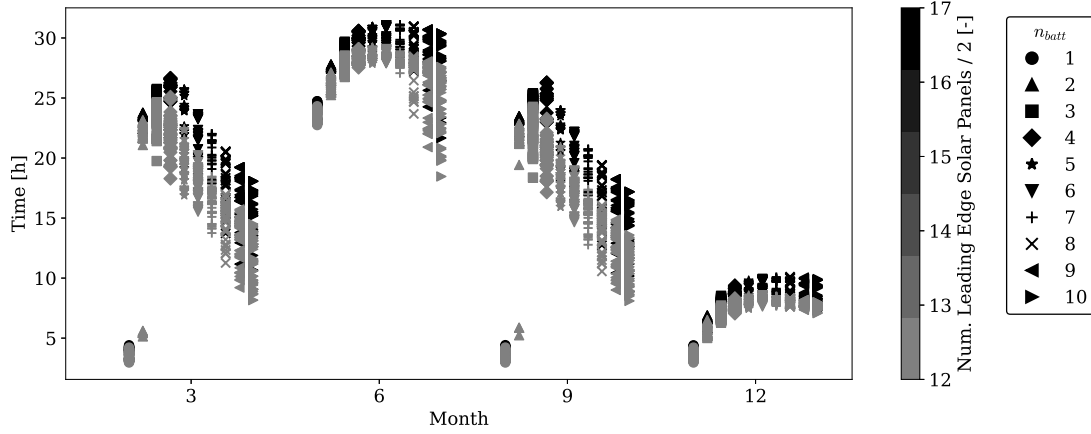


Figure 4.21: Flight time as a function of time of year, identified by n_{batt} for payload option 2 in Inuvik, NT.

The initial analysis did not show any configuration dependence on the charge margin for $b_{PV1} = 12$. Changing the time of year and location shows a different story. At $b_{PV1} = 17$, there is no distinguishable preference to the clean wing or nacelle throughout the year, at either location, with or without payload. The vehicle is less sensitive to the aerodynamic variations in wing configuration at the higher wingspan. At $b_{PV1} = 12$ and higher latitudes, the charge margin is generally longer for the nacelle configurations compared to the clean wing, shown in Figure 4.23 where the charge margin is a function of n_{batt} with markers indicating Configuration for no payload in month 6 in Inuvik. The differences in charge margin between configuration are not significant for lower latitude and payload has minimal further effect, supported by Figures F.39 through F.41.

The takeoff time in relation to the sunrise time may be playing a significant role in this metric, but the differences in battery charging and discharging rates is a far stronger indicator over which configuration performs better. This is dependent on the initial

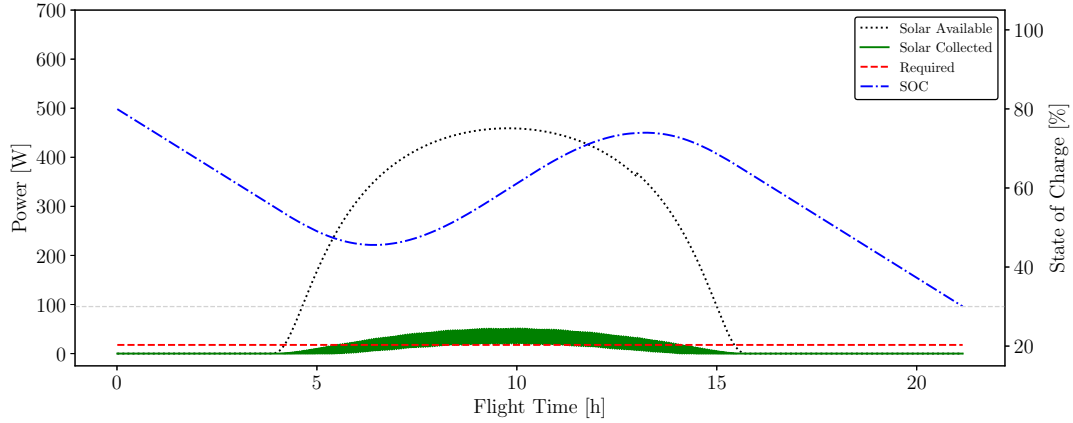


Figure 4.22: Power levels and SOC for a UAV with $b_{PV1} = 12$, $n_{batt} = 3$, beginning at 1100 UTC, flown in Inuvik, with payload option 2.

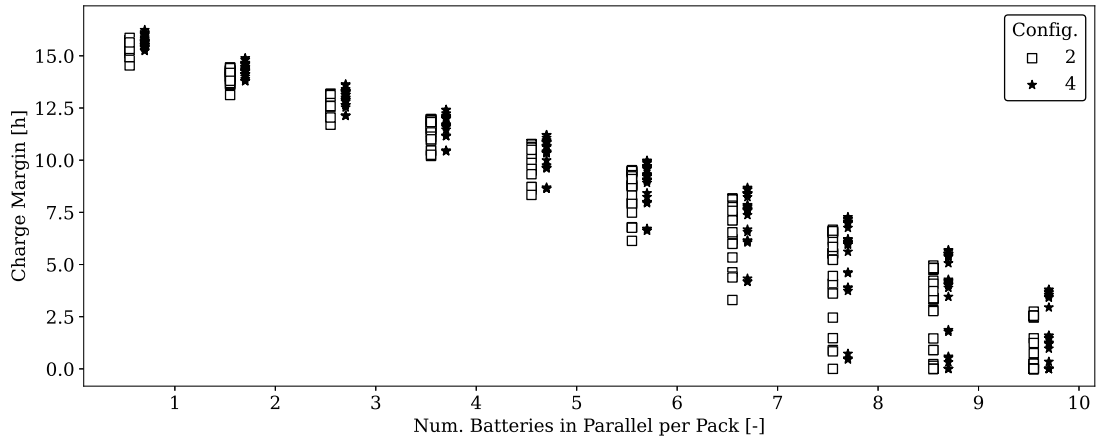


Figure 4.23: Charge margin as a function of n_{batt} , identified by Configuration for zero payload, $b_{PV1} = 12$, month 6, flown in Inuvik.

geometry selected for design, where some solar panel layouts may not result in large differences in wing planform, considering the discontinuities that can occur with the nacelles. Figures 4.24 and 4.25 show the SOC throughout the flight for a vehicle with notable geometric discontinuity in chord length between the clean wing and nacelle. Both figures shows the SOC at all n_{batt} in Inuvik in September (month 9) with no payload. For this particular vehicle, the the clean wing has consistently shorter flight times for all respective n_{batt} . Most notably in $n_{batt} = 1$, the discharge rate is slower for the nacelle wing, indicated by the higher SOC before sunrise. These differences further support the interest in optimizing the nacelle configuration in addition to the clean wing to explore if the discontinuities in the nacelle configuration provide further advantages for LALE solar flight.

The initial analysis showed that the maximum aerodynamic efficiency and highest aspect ratios, indicative of a glider-like wing, were not the most promising configurations

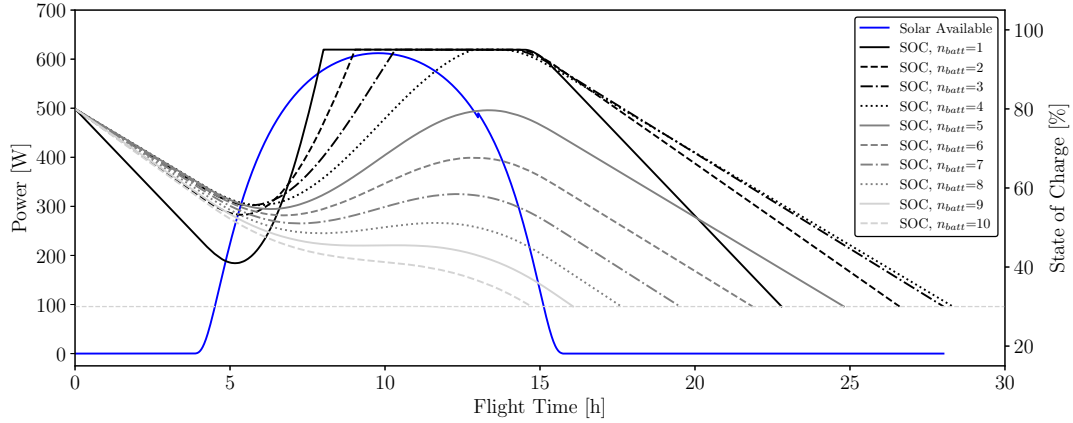


Figure 4.24: SOC for each n_{batt} with a clean wing UAV, in month 9 with no payload, flown in Inuvik.

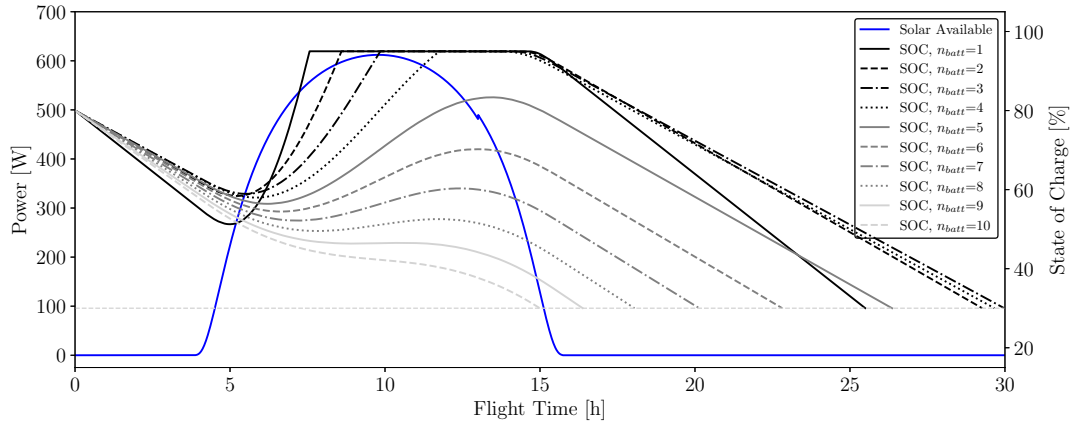


Figure 4.25: SOC for each n_{batt} with nacelles, in month 9 with no payload, flown in Inuvik.

at higher n_{batt} , which are more suitable for larger payloads. This trend is consistent for varying times of year and latitude. Figure 4.26 shows the flight time as a function of n_{batt} , where the colour indicates maximum aerodynamic efficiency, and the markers indicate n_{PV} which are expanded along the x -axis for $b_{PV1} = 12$, no payload, in month 9 in Inuvik. There remains a strong sensitivity to n_{PV} on n_{batt} , where the larger battery packs achieve a higher charge and therefore longer flight time when there are more panels onboard, which is consistently observed throughout the year and at both locations. Figures F.42 through F.48 show the same trends at different times in both analyzed locations.

Ultimately, achieving LALE solar-powered flight at different times of the year in different locations is most influenced by the wingspan of the vehicle. There is little to no variation in the optimal n_{batt} throughout the year or with location, and remains dependent on the power draw of any payload. These results remain supportive that a glider-like wing is

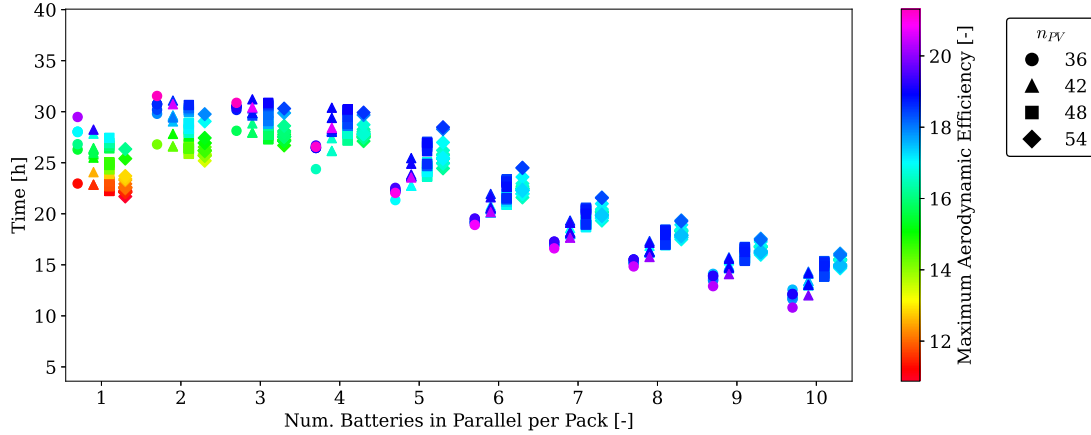


Figure 4.26: Flight time as a function of n_{batt} , identified by n_{PV} for zero payload at $b_{PV1} = 12$, month 9, flown in Inuvik.

suitable for achieving LALE solar-powered flight while also showing that deviating from a glider-like design is beneficial when adding payload. Larger battery packs require more solar panels to fully charge the batteries, especially when there are fewer daylight hours in winter months. If optimizing to achieve the longest window where continuous flight is possible, it is expected that a clean wing design will make the upper wingspan constraint active with as many solar cells as possible, with the $\mathcal{R}$ and n_{batt} being dependent on the payload of the vehicle.

4.6 Parametric Analysis Summary

Results at different latitudes and time of year remain consistent with the initial parametric analysis. The optimal battery capacity is dependent on the payload of the vehicle. Larger payloads are able to achieve longer flight times with higher energy capacity and lowered aerodynamic efficiency, demonstrating the beneficial tradeoff between both parameters. Time of year and latitude has little effect on the most promising configuration. The total flight time is more sensitive to the wingspan of the vehicle. Designing a vehicle within wingspan constraints to be hand launchable, more manageable to transport, with fewer aeroelastic challenges compared to a larger wingspan will reduce the window of the year that continuous flight is possible.

A clean wing T-tail solar UAV is the best configuration to pursue for LALE solar-powered flight, irrespective of application; however, there are a few limitations with this analysis. The battery packs are assumed to be cubes, which is not a realistic layout for the cells. In reality, they would be packed in a fashion to reduce the frontal cross-sectional area of the fuselage and nacelles. The wing weight model does not include a point load for the nacelle configurations, where the weight of the battery packs on the wings at the nacelle locations provides inertial relief and can reduce the weight of the wing structure. Accounting for both of these simplifications have the potential to shift the results to favour the nacelle configuration by resulting in smaller frontal area and reduced structure weight. The nacelles allow the wing to have geometric discontinuity, which could allow the nacelle configuration to be more advantageous when optimized.

The next steps are to move forward with optimizing a T-tail solar UAV for flight specifically in Kingston. The potential advantages of the nacelle configurations are not fully captured in the parametric analysis. This will be addressed and explored by optimization both Configurations 2 and 4 in the next chapter.

5 Solar UAV Optimization

Results from Chapter 4 shows that a T-tail empennage is most advantageous for LALE solar-powered flight. Within the limitations of the parametric analysis modelling, there are potential advantages with a nacelle configuration. This chapter optimizes both the clean and nacelle configurations, with and without payloads to further identify the optimal solar UAV configuration and the effect of payload and power draw.

5.1 Optimization Problem

The primary objective of a optimizing a solar-powered UAV is to prolong flight time. Simulating the vehicle's trajectory over long periods of time can be computationally expensive and when optimizing, does not appropriately account for energetic margins. In this work, having an energetic margin refers to a higher SOC than required for a given day's available solar irradiance or having more time than necessary at the maximum SOC. Energetic margins provide robustness to disruptions in the solar power collected or to the power draw of the vehicle due to changes in weather. Rather than simulate multi-day flights, the optimization objective function will use the definition of a charge margin, t_{cm} , shown in Figure 5.1, which is the time the battery spends at the maximum SOC. The simulated flight begins at the time of day where the solar power collected is equivalent to the required power, *i.e.*, the time that the battery would begin to discharge if it had achieved a full SOC during flight that day. The flight is ended when the SOC reaches 30% or 24 hours have passed, whichever occurs first.

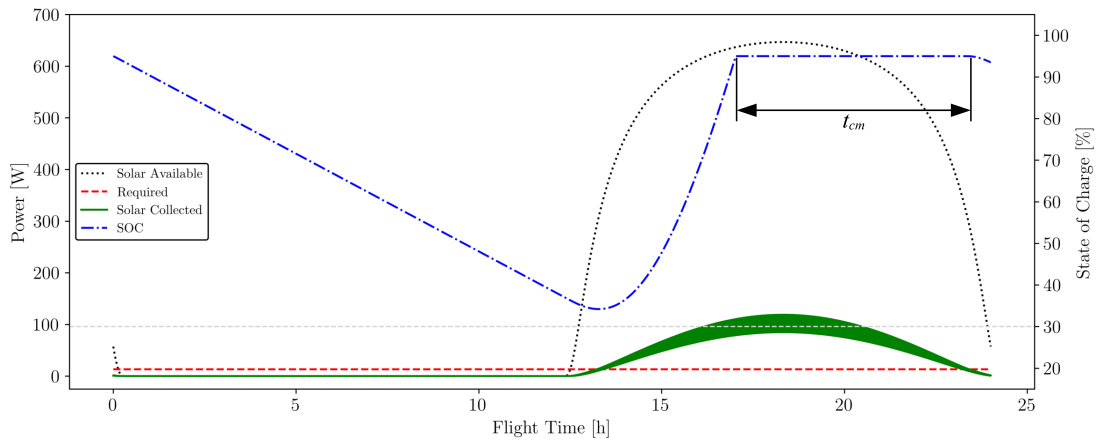


Figure 5.1: Sample simulated flight for optimization objective function definitions.

With the underlying objective of achieving LALE solar-powered flight throughout as much of the year as possible, the objective function is expanded to consider different times of the year. For a given design, the vehicle is simulated for 21 June. If it is able to complete 24 hours of flight successfully and recharge the batteries, then 21 March and 21 September are flown. If successful at recharging the batteries on the equinoxes, then 21 December is flown. If the SOC discharges to 30% at any stage of flight, it is the end of the simulation for that vehicle, regardless of the month. The total fitness is defined as the sum of the flight time t , and charge margin t_{cm} , formulated in Equation 5.1.

$$\begin{aligned}
 \max. \quad & f(X) = \sum_{i \in [6,3,9,12]}^{\text{month}=i} t + t_{cm} \\
 \text{w.r.t} \quad & X = \{x_1, x_2, \dots, x_n\}^T \\
 \text{s.t.} \quad & G(j)|_X \leq 0, \quad j = 1, 2, \dots, m
 \end{aligned} \tag{5.1}$$

The size of X and value of m is dependent on the wing configuration being optimized. Table 5.1 summarizes the design variables and Table 5.2 summarizes the constraints for the clean configuration optimization, where there are 35 design variables and 43 defined constraints. For the nacelle configuration, the variables and constraints are given by Tables 5.3 and 5.4, respectively, with 43 design variables and 41 constraints. The objective function is the same regardless of wing configuration. The constraints in Tables 5.1 and 5.3 are described as inequalities with $a \leq G_j(X) \leq b$, where a and b are the constraint bounds, which are evaluated as $a - G_j(X) = G(j) \leq 0$ or $G_j(X) - b = G(j) \leq 0$. Throughout this chapter, $G(j)|_X$ is referred to as the value of constraint j .

For designs that are capable of recharging in all four months (March, June, September, and December), the contribution of flight time will not contribute to one design being better than another. The flight time is considered to promote designs that are able to fly longer later in the year, even if they are not capable of recharging their batteries. Otherwise the optimization may stagnate at a local maximum that is able to recharge the batteries at the equinoxes but never explore solutions that might also be capable of recharging its batteries at the winter solstice. It is t_{cm} that has the larger influence on the overall design, which is increased by minimizing the required power of the vehicle and by increasing the charging rate after sunrise. These directly influence the aerodynamics, propulsion system and its efficiency, flight speed, trajectory turning radius, and number of solar panels.

Figure 5.2 defines variables and constraints for the clean configuration optimization, showing physical dimensions for span, b , and chord length, c , parameters. Span values indicated as b_{PVx} are integer values for the number of solar panels in the x th row. l_{PV} is the length of the solar panels, i indicates the inner wing segment, o indicates the outer wing segment.

Figure 5.3 defines variables and constraints for the nacelle configuration, showing physical dimensions in the same way as Figure 5.2 for the clean configuration. The inclusion of nacelles allows for discontinuity in the wing planform. These discontinuities, the shape of the nacelle surfaces, and the placement of batteries within the nacelles account

Table 5.1: Variables for clean configuration optimization, where $P(i) \in [a_i, b_i]$.

Label	Design Variable Name	a_i	b_i	$P(i) \in$
P(0)	b_{PV1} [-] (see Figure 5.2)	5	15	$\mathbb{Z}$
P(1)	b_{PV2} [-] (see Figure 5.2)	0	15	$\mathbb{Z}$
P(2)	b_{PV3} [-] (see Figure 5.2)	0	0	$\mathbb{Z}$
P(3)	b_{PV_i} [-] (see Figure 5.2)	0	14	$\mathbb{Z}$
P(4)	b_i [ft] (see Figure 5.2)	0.44	6	$\mathbb{R}$
P(5)	c_i [ft] (see Figure 5.2)	0.44	1.167	$\mathbb{R}$
P(6)	b_o [ft] (see Figure 5.2)	0.44	6	$\mathbb{R}$
P(7)	c_t [ft] (see Figure 5.2)	0.44	1.167	$\mathbb{R}$
P(8)	Wing thickness to chord ratio [-]	0.08	0.15	$\mathbb{R}$
P(9)	Wing airfoil ID [-] (see Table 5.5)	0	4	$\mathbb{Z}$
P(10)	Wing x offset [ft]	-1	1	$\mathbb{R}$
P(11)	Fuselage nose section length [ft]	0.16	0.75	$\mathbb{R}$
P(12)	Fuselage squareness factor [-]	0.4	1.0	$\mathbb{R}$
P(13)	Fuselage tail length [ft]	0.5	5	$\mathbb{R}$
P(14)	Aft fuselage radius [ft]	0.02	0.25	$\mathbb{R}$
P(15)	Fuselage height to width ratio [-]	0.5	2	$\mathbb{R}$
P(16)	Horizontal tail root chord [ft]	0.2	1.167	$\mathbb{R}$
P(17)	Horizontal tail taper ratio [-]	0.4	0.6	$\mathbb{R}$
P(18)	Horizontal tail semi-span [ft]	0.2	2	$\mathbb{R}$
P(19)	Horizontal tail sweep [deg]	0	30	$\mathbb{R}$
P(20)	Horizontal tail dihedral [deg]	-3	3	$\mathbb{R}$
P(21)	Horizontal tail root thickness to chord ratio [-]	0.10	0.18	$\mathbb{R}$
P(22)	Horizontal tail tip thickness to chord ratio [-]	0.08	0.15	$\mathbb{R}$
P(23)	Horizontal tail twist [deg]	-3	3	$\mathbb{R}$
P(24)	Vertical tail root chord [ft]	0.2	1.167	$\mathbb{R}$
P(25)	Vertical tail taper ratio [-]	0.3	0.7	$\mathbb{R}$
P(26)	Vertical tail span [ft]	0.2	1.5	$\mathbb{R}$
P(27)	Vertical tail sweep [deg]	0	30	$\mathbb{R}$
P(28)	Vertical tail root thickness to chord ratio [-]	0.10	0.18	$\mathbb{R}$
P(29)	Vertical tail tip thickness to chord ratio [-]	0.10	0.15	$\mathbb{R}$
P(30)	Vertical tail TE position [% of fuselage length]	93	98	$\mathbb{R}$
P(31)	Turn radius [ft]	50	400	$\mathbb{R}$
P(32)	Cruising airspeed [$\frac{\text{ft}}{\text{s}}$]	5	50	$\mathbb{R}$
P(33)	Number of batteries in parallel [-]	1	10	$\mathbb{Z}$
P(34)	Propeller ID [-] (see Table 5.6)	0	11	$\mathbb{Z}$
P(35)	BLDC motor ID [-] (see Table 5.7)	0	15	$\mathbb{Z}$

Table 5.2: Constraints for clean configuration optimization.

Label	Constraint	Value
G(0)	b_{PV2} [-] (see Figure 5.2)	$\leq b_{PV1}$
G(1)	b_{PV3} [-] (see Figure 5.2)	$\leq b_{PV2}$
G(2)	b_{PV_i} [-] (see Figure 5.2)	$\leq b_{PV1}$
G(3), G(4)	Total number of solar panels [-]	$34 \leq n_{PV,tot} \leq 72$
G(5)	if $b_{PV2} \neq 0$, b_{PV_i} [-] (see Figure 5.2)	$\leq b_{PV2}$
G(6)	if $b_{PV3} \neq 0$, b_{PV_i} [-] (see Figure 5.2)	$\leq b_{PV3}$
G(7)	Wing span [ft]	≤ 10
G(8)	b_i [ft] (see Figure 5.2)	$\geq b_{PV_i} \times l_{PV}$
G(9)	c_i [ft] (see Figure 5.2)	$\left\{ \begin{array}{l} \geq c_3 \text{ if } b_{PV3} > 0 \\ \geq c_2 \text{ if } b_{PV2} > 0 \\ \text{and } b_{PV3} = 0 \\ \geq c_1 \text{ if } b_{PV2} = b_{PV3} = 0 \end{array} \right.$
G(10)	b_o [ft] (see Figure 5.2)	$\geq b_{PV_o} \times l_{PV}$
G(11)	c_3 [ft] (see Figure 5.2)	$\leq c(y_3)$
G(12)	c_2 [ft] (see Figure 5.2)	$\leq c(y_2)$
G(13)	c_1 [ft] (see Figure 5.2)	$\leq c(y_1)$
G(14)	c_t [ft] (see Figure 5.2)	$\leq c_i$
G(15)	Wing taper ratio [-]	≥ 0.4
G(16)	Wing taper ratio $\times$ aspect ratio [-]	≤ 11.8
G(17)	Wing outer mean thickness [ft]	≥ 0.082
G(18)	Wing leading edge position [ft]	$\geq x_{nose_{TE}}$
G(19)	Horizontal tail TE sweep [deg]	≥ 0
G(20), G(21)	Vertical tail tip chord [ft]	$0.5c_{r_{HT}} \leq c_{t_{VT}} \leq 0.8c_{r_{HT}}$
G(22)	Horizontal tail tip thickness [ft]	$\leq t_{HT_r}$
G(23)	Vertical tail tip thickness [ft]	$\leq t_{VT_r}$
G(24)	Horizontal tail thickness [ft]	≥ 0.082
G(25)	Vertical tail thickness [ft]	≥ 0.082
G(26)	Vertical tail TE sweep [deg]	≥ 0
G(27)	Vertical tail leading edge position [ft]	$> x_{wing_{TE}}$
G(28)	Vertical tail chord length [ft]	$\leq l_{tail}$
G(29)	Aft fuselage radius [ft]	$\leq r_{fuselage \text{ body}}$
G(30)	Horizontal tail aspect ratio [-]	≤ 10
G(31)	Vertical tail aspect ratio [-]	≤ 2
G(32), G(33)	Centre of gravity location [% MAC]	$5 \leq x_{CG} \leq 55$
G(34)	Static margin [% MAC]	≥ 10
G(35)	Turning g -limit [-]	≤ 1.5
G(36)	Lift coefficient [-]	$\leq C_{L_{max}}/1.3^2$
G(37)	Horizontal tail lift coefficient [-]	≤ 0.8
G(38), G(39)	Static roll stability	$-0.20 \leq C_{l,\beta} \leq -0.05$
G(40), G(41)	Static yaw stability	$0.03 \leq C_{n,\beta} \leq 0.20$
G(42)	Drag at cruise [lb _f]	$\leq T_{max}$
G(43)	Required irradiance [$\frac{W}{m^2}$]	$< I_{s_{max, June}}$

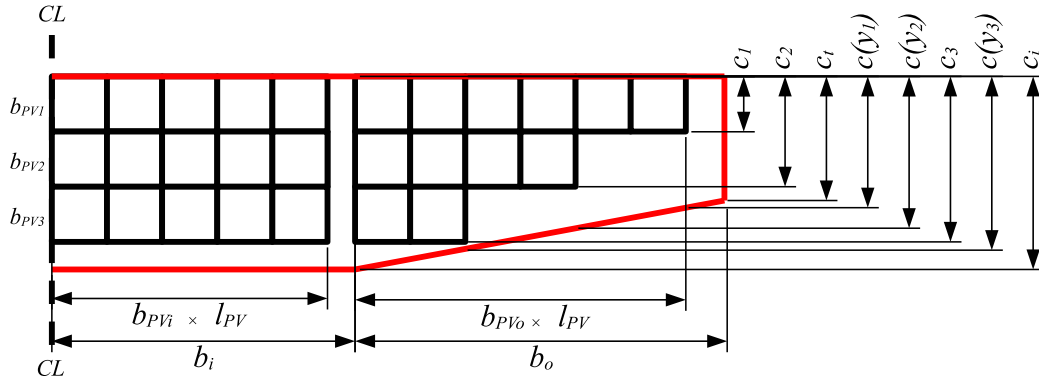


Figure 5.2: Clean configuration span and chord variables.

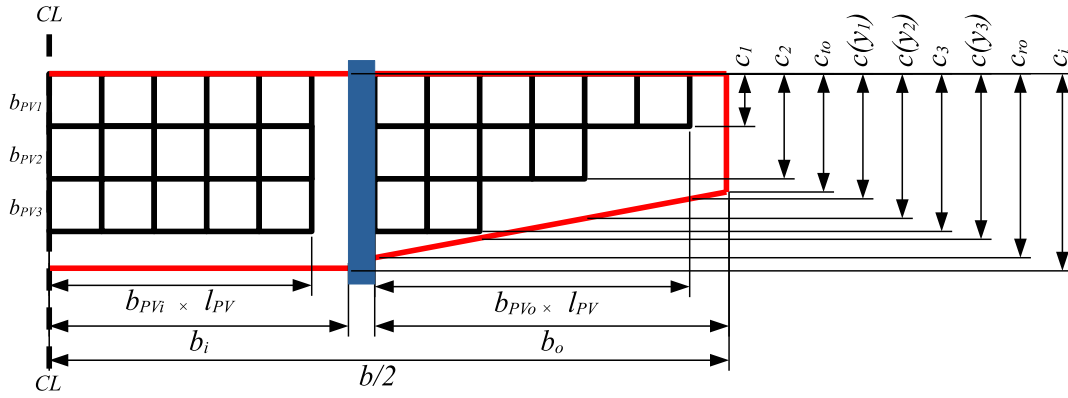


Figure 5.3: Nacelle configuration span and chord variables.

for the additional design variables and constraints that are not included in the clean configuration optimization. The variables and constraints for the nacelle configuration are summarized in Tables 5.3 and 5.4.

Table 5.4: Constraints for nacelle configuration optimization.

Label	Constraint	Value
G(0)	b_{PV2} [-] (see Figure 5.3)	$\leq b_{PV1}$
G(1)	b_{PV3} [-] (see Figure 5.3)	$\leq b_{PV2}$
G(2)	b_{PV_i} [-] (see Figure 5.3)	$\leq b_{PV1}$
G(3), G(4)	Total number of solar panels [-]	$34 \leq n_{PV,tot} \leq 72$
G(5)	if $b_{PV2} \neq 0$, b_{PV_i} [-] (see Figure 5.3)	$\leq b_{PV2}$
G(6)	if $b_{PV3} \neq 0$, b_{PV_i} [-] (see Figure 5.3)	$\leq b_{PV3}$
G(7)	Wing span [ft]	≤ 10
G(8)	$2b_i/b$ [-] (see Figure 5.3)	≤ 0.6
G(9)	b_i [ft] (see Figure 5.3)	$\geq b_{PV_i} \times l_{PV}$

Continued on next page

Table 5.4 – continued from previous page

Label	Constraint	Value
G(10)	c_i [ft] (see Figure 5.3)	$\begin{cases} \geq c_3 & \text{if } b_{PV3} > 0 \\ \geq c_2 & \text{if } b_{PV2} > 0 \\ & \text{and } b_{PV3} = 0 \\ \geq c_1 & \text{if } b_{PV2} = b_{PV3} = 0 \end{cases}$
G(11)	b_o [ft] (see Figure 5.3)	$\geq b_{PVo} \times l_{PV}$
G(12)	c_3 [ft] (see Figure 5.3)	$\leq c(y_3)$
G(13)	c_2 [ft] (see Figure 5.3)	$\leq c(y_2)$
G(14)	c_1 [ft] (see Figure 5.3)	$\leq c(y_1)$
G(15)	c_i [ft] (see Figure 5.3)	$\geq c_{ro}$
G(16)	c_{to} [ft] (see Figure 5.3)	$\leq c_i$
G(17)	Wing taper ratio [-]	≥ 0.4
G(18)	Wing taper ratio $\times$ aspect ratio [-]	≤ 11.8
G(19)	Battery TE x -position	$\leq x_{nacTE}$
G(20)	Wing outer mean thickness [ft]	≥ 0.082
G(21)	Wing leading edge position [ft]	$\geq x_{noseTE}$
G(22)	Nacelle height [ft]	$\geq t_{iwing}$
G(23)	Nacelle height [ft]	$\geq t_{rowing}$
G(24)	Horizontal tail TE sweep [deg]	≥ 0
G(25), G(26)	Vertical tail tip chord [ft]	$0.5c_{rHT} \leq c_{tVT} \leq 0.8c_{rHT}$
G(27)	Horizontal tail tip thickness [ft]	$\leq t_{HTr}$
G(28)	Vertical tail tip thickness [ft]	$\leq t_{VT_r}$
G(29)	Horizontal tail thickness [ft]	≥ 0.082
G(30)	Vertical tail thickness [ft]	≥ 0.082
G(31)	Vertical tail TE sweep [deg]	≥ 0
G(32)	Vertical tail leading edge position [ft]	$> x_{wingTE}$
G(33)	Vertical tail chord length [ft]	$\leq l_{tail}$
G(34)	Aft fuselage radius [ft]	$\leq r_{fuselage\ body}$
G(35)	Horizontal tail aspect ratio [-]	≤ 10
G(36)	Vertical tail aspect ratio [-]	≤ 2
G(37), G(38)	Centre of gravity location [% MAC]	$5 \leq x_{CG} \leq 55$
G(39)	Static margin [% MAC]	≥ 10
G(40)	Turning g -limit [-]	≤ 1.5
G(41)	Lift coefficient [-]	$\leq C_{Lmax}/1.3^2$
G(42)	Horizontal tail lift coefficient [-]	≤ 0.8
G(43), G(44)	Static roll stability	$-0.20 \leq C_{l,\beta} \leq -0.05$
G(45), G(46)	Static yaw stability	$0.03 \leq C_{n,\beta} \leq 0.20$
G(47)	Drag at cruise [lb _f]	$\leq T_{max}$
G(48)	Required irradiance [$\frac{W}{m^2}$]	$< I_{smax_June}$

Table 5.5 summarizes the airfoil associated with each ID number for the wing in variable P(9) in the clean configuration optimization and P(11) and P(12) in the nacelle configuration optimization. The thickness values indicated by XX are given by the variables P(8), and P(9) and P(10), respectively. Various other LALE solar UAVs have custom airfoils designed with a flatter upper surface for the integration of solar cells and for

Table 5.3: Variables for nacelle configuration optimization, where $P(i) \in [a_i, b_i]$.

Label	Design Variable Name	a_i	b_i	$P(i) \in$
P(0)	b_{PV1} [-] (see Figure 5.3)	8	15	$\mathbb{Z}$
P(1)	b_{PV2} [-] (see Figure 5.3)	0	15	$\mathbb{Z}$
P(2)	b_{PV3} [-] (see Figure 5.3)	0	0	$\mathbb{Z}$
P(3)	b_{PV_i} [-] (see Figure 5.3)	1	14	$\mathbb{Z}$
P(4)	b_i [ft] (see Figure 5.3)	0.44	7	$\mathbb{R}$
P(5)	c_i [ft] (see Figure 5.3)	0.44	1.167	$\mathbb{R}$
P(6)	b_o [ft] (see Figure 5.3)	0.44	7	$\mathbb{R}$
P(7)	c_t [ft] (see Figure 5.3)	0.44	1.167	$\mathbb{R}$
P(8)	c_{ro} [ft] (see Figure 5.3)	0.44	1.167	$\mathbb{R}$
P(9)	Wing inner thickness to chord ratio [-]	0.08	0.15	$\mathbb{R}$
P(10)	Wing outer thickness to chord ratio [-]	0.08	0.15	$\mathbb{R}$
P(11)	Wing inner airfoil ID [-] (see Table 5.5)	0	4	$\mathbb{R}$
P(12)	Wing outer airfoil ID [-] (see Table 5.5)	0	4	$\mathbb{Z}$
P(13)	Wing outer incidence angle [deg]	-3	3	$\mathbb{R}$
P(14)	Wing outer dihedral angle [deg]	-3	3	$\mathbb{R}$
P(15)	Wing x offset [ft]	-1	1	$\mathbb{R}$
P(16)	Fuselage nose section length [ft]	0.16	0.75	$\mathbb{R}$
P(17)	Fuselage squareness factor [-]	0.4	1.0	$\mathbb{R}$
P(18)	Fuselage tail length [ft]	0.5	5	$\mathbb{R}$
P(19)	Aft fuselage radius [ft]	0.02	0.25	$\mathbb{R}$
P(20)	Fuselage height to width ratio [-]	0.5	2	$\mathbb{R}$
P(21)	Horizontal tail root chord [ft]	0.2	1.167	$\mathbb{R}$
P(22)	Horizontal tail taper ratio [-]	0.4	0.6	$\mathbb{R}$
P(23)	Horizontal tail semi-span [ft]	0.2	2	$\mathbb{R}$
P(24)	Horizontal tail sweep [deg]	0	30	$\mathbb{R}$
P(25)	Horizontal tail dihedral [deg]	-3	3	$\mathbb{R}$
P(26)	Horizontal tail root thickness to chord ratio [-]	0.10	0.18	$\mathbb{R}$
P(27)	Horizontal tail tip thickness to chord ratio [-]	0.08	0.15	$\mathbb{R}$
P(28)	Horizontal tail twist [deg]	-3	3	$\mathbb{R}$
P(29)	Vertical tail root chord [ft]	0.2	1.167	$\mathbb{R}$
P(30)	Vertical tail taper ratio [-]	0.3	0.7	$\mathbb{R}$
P(31)	Vertical tail span [ft]	0.2	1.5	$\mathbb{R}$
P(32)	Vertical tail sweep [deg]	0	30	$\mathbb{R}$
P(33)	Vertical tail root thickness to chord ratio [-]	0.10	0.18	$\mathbb{R}$
P(34)	Vertical tail tip thickness to chord ratio [-]	0.10	0.15	$\mathbb{R}$
P(35)	Vertical tail TE position [% of fuselage length]	93	98	$\mathbb{R}$
P(36)	Nacelle squareness factor [-]	0.4	1	$\mathbb{R}$
P(37)	Nacelle height to width ratio [-]	0.5	2	$\mathbb{R}$
P(38)	Turn radius [ft]	50	400	$\mathbb{R}$
P(39)	Cruising airspeed [$\frac{\text{ft}}{\text{s}}$]	5	50	$\mathbb{R}$
P(40)	Number of batteries in parallel [-]	1	10	$\mathbb{R}$
P(41)	Battery pack x -offset [ft]	0	1.167	$\mathbb{R}$
P(42)	Propeller ID [-] (see Table 5.6)	0	11	$\mathbb{R}$
P(43)	BLDC motor ID [-] (see Table 5.7)	0	15	$\mathbb{R}$

high aerodynamic efficiency [1, 12, 46]. The design and optimization of an airfoil suitable for a LALE solar UAV is sophisticated enough to be its own project and becomes increasingly more computationally expensive if combined with the vehicle optimization. For the sake of simplicity, only five NACA 4-digit airfoils are considered, to account for some of the effects of maximum camber and its position.

Table 5.5: Wing airfoil ID options.

ID	Airfoil
0	NACA 22XX
1	NACA 24XX
2	NACA 42XX
3	NACA 44XX
4	NACA 00XX

Table 5.6 summarizes the propellers from the APC propeller database considered for the design [49]. The ID corresponds to the values of variables P(34) and P(42) in the clean and nacelle configurations, respectively. The first digit in the propeller name is the diameter of the propeller in inches, the second digit indicates the pitch of the propeller in inches, and the E indicates these are for use with electric motors.

Table 5.6: Propeller ID options.

ID	Propeller
0	6x4E
1	6x6E
2	6x5.5E
3	7x4E
4	7x5E
5	7x6E
6	8x4E
7	8x6E
8	9x6E
9	9x4.5E
10	10x5E
11	10x6E

Table 5.7 summarizes the BLDC motors considered within the optimization. The ID corresponds to variables P(35) and P(43) in the clean and nacelle configurations, respectively. Each motor is characterized by its configuration, where the first two digits represent the stator diameter in mm and the last two digits represent the stator height in mm, and its KV rating.

The intention is to 3D-print the final configuration. To facilitate this, the upper chord limits of the wings and empennage surfaces are dictated by the largest available printing bed, where these components would be printed with the airfoil cross-section on the print bed with the span along the printer's z -coordinate. Each segment of the span is limited by the height of the printer, therefore they would be printed in several segments, glued

Table 5.7: BLDC motor ID options.

ID	Motor Configuration	K_V
0	2208	1100
1	2208	1260
2	2208	1500
3	2212	980
4	2212	1250
5	2212	1400
6	2212	2450
7	2216	880
8	2216	1100
9	2216	1250
10	2216	1400
11	2216	1800
12	2216	2400
13	2814	900
14	2814	1000
15	2814	1250

together, and stiffened with a carbon fibre rod as the primary structure. The thicknesses constraints of the wing and empennage surfaces are to maintain internal volume for the integration of control surface servo motors. The wing geometry constraints ensure that the solar panels fit on the wing surface. Centre of gravity, static margin, and stability derivatives are constrained to follow typical limits and general guidelines for commercial aircraft. There are no cross-wind constraints, since it is assumed that the UAV will be launched and landed in the open where the vehicle may be aligned with the wind direction and cross wind effects can be minimized.

The problem is defined to consider up to three solar cells along the chord of the wing. The chord limitation dictated by the printers is only large enough to accommodate two solar cells. Therefore $P(2)$ in both optimization problems is bounded by $[0,0]$, effectively removing the design variable from the solution space without reformulating the model in its entirety. With a larger print bed, the chord may be able to accommodate the additional panel along the span and redefine the bounds of $P(2)$.

5.2 Flight Assumptions

Although this optimization problem is intended to optimize the vehicle and not the trajectory, it is recognized that there is a coupling between the weight, airspeed, bank angle, and sun elevation angle for the optimal vehicle and trajectory [68]. To consider these effects, the flight speed and turning radius are included as variables. The g -loading constraint and $C_{L_{max}}$ constraint ensure the vehicle is not overloaded or stalled in flight.

Each flight is simulated in a circular trajectory at 2000 ft MSL. The flight begins in the evening when the vehicle would begin to discharge had it been flying previously with a fully charged battery. The flight continues, governed by the equations of motion in

Equation 4.8 (where $V_{D_{min}}$ is the optimization variable V), until the vehicle has flown for 24 hours or the SOC decreases to 30%, whichever occurs first.

5.3 Initial Results

This section presents the results for the optimizations of the clean and nacelle configurations, both with and without payload. The optimization problem is solved using the Augmented Lagrange Particle Swarm Optimizer in *pyOPT*, using 39 particles up to a maximum of 250 iterations [111]. The solution is considered converged after five consecutive iterations within the default absolute tolerance of 10^{-2} for the objective function, which is evaluated in hours. Every constraint is evaluated with a tolerance of 10^{-3} , referred to as the *tolerance threshold*. This section details the results of the optimization problem as defined in Section 5.1, where there is no upper constraint on the static margin.

5.3.1 Clean Configuration without Payload

This configuration converged in 33 iterations, with a fitness of 131.93 h. This fitness level indicates that the converged UAV was able to recharge its batteries in all four months, with $t_{cm} = 5.13$ h in month 12. The fitness for each iteration is plotted in Figure 5.4.

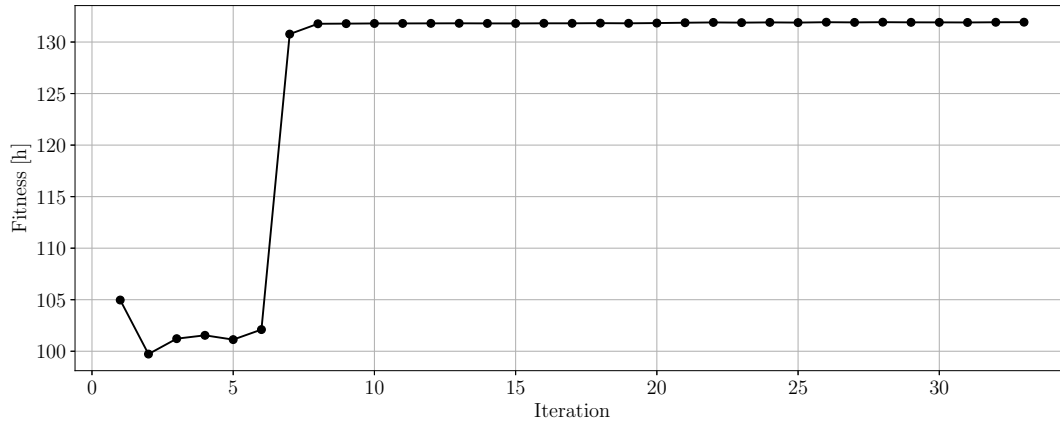


Figure 5.4: Fitness convergence for the clean configuration without payload.

The value of the constraints which do not satisfy $G(j) \leq 0$ within the tolerance threshold are plotted in Figure 5.5 for the member with the highest fitness of each iteration, where the labels in the legend correspond to the constraints in Table 5.2.

In some cases, it is difficult to decipher how many and which constraints are not satisfied as they are presented in Figure 5.5. The constraints are not normalized in any way, so a relatively small positive value for the constraint is not indicative of being close to compliance. It is much more useful to visualize which individual constraints are being violated in a particular iteration. Figure 5.6 presents the constraints as a scatter

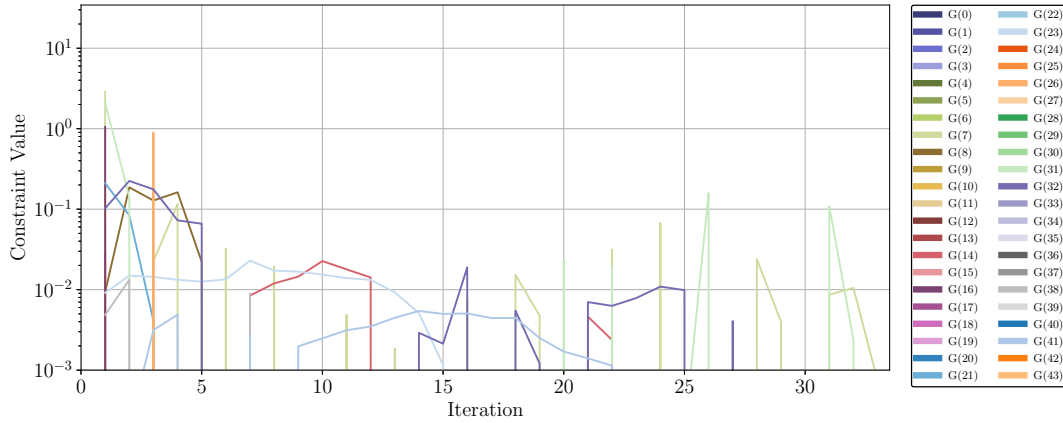


Figure 5.5: Constraints not meeting the tolerance threshold and their values for the clean configuration without payload.

plot, where a point is plotted for a given iteration if that constraint is being violated, regardless of its magnitude if it does not meet the tolerance threshold. The indices for each constraint correspond to Table 5.2. For all other optimizations, the values of any unsatisfied constraints will be included in Appendix G.

The scatter plot shows that early iterations were fighting against the span, vertical tail thickness, and directional stability (constraints 7, 23, and 41). Through later iterations, the span, vertical tail $\mathcal{R}$, and forward CG limit (constraints 7, 31, and 32) were directing the design towards the feasible design space. The vertical tail geometry constraints and directional stability constraint with $C_{n,\beta}$ shows the direct trade off between the two, where the smaller surface does not provide enough directional stability but is limited by the geometry constraints. This is an expected result of the optimization process because of the dependencies on and interconnectedness between the CG location, stabilizer surface areas, and position of their aerodynamic centres. The sweep and taper of the horizontal stabilizer is a result of the leading edge's coincident position with the vertical stabilizer's leading edge, which sweeps the surface to move the aerodynamic centre further aft and increase the moment arm with respect to the CG location.

The evolution of the design variables for the best member of each iteration are presented in a parallel lines plot in Figure 5.7. Each vertical axis represents one design variable, labelled and bounded by the values in Table 5.1. One iteration is represented by a line traversing left to right across the axes, intersecting a given variable axis at the value it holds for that iteration. The colour of each line indicates the iteration, given by the colour bar. This visualization of the design variables gives some insight to how the algorithm searches through the design space and if the optimum is found at the boundary of any given variable.

The inner and outer wing chord lengths (variables 5 and 7) show a strong preference to a rectangular wing, maximizing the total span and chord lengths. Due to the preference to an untapered wing, there is no effect of the inner span length on the final

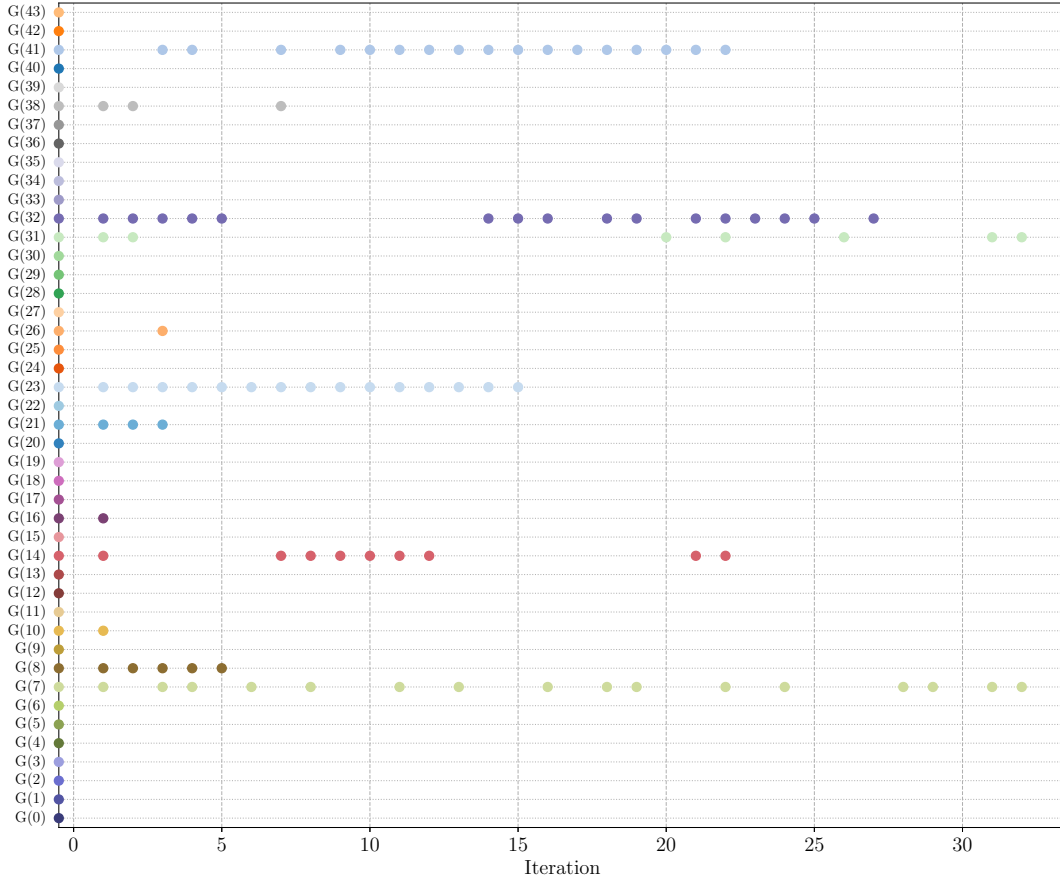


Figure 5.6: Scatter of constraints not meeting the tolerance threshold for the clean configuration without payload.

design. There is a strong preference to the NACA00XX airfoil, suggesting camber is not necessary for the lower airspeed of this vehicle and satisfying the $C_{L_{max}}$ constraint.

Contrary to the results from the parametric analysis in Chapter 4, the optimized UAV uses only one battery cell in parallel in each battery pack. From early iterations, the parallel lines plot shows that the heavier iterations with $n_{batt} = 6$ (P(33)) favour higher airspeeds and a different motor than the lighter, slower iterations. There are very minimal combinations of propeller and motors that are shown, indicating either a large incompatibility between the selected propellers and motors or a very strong preference to propeller 6 and motor 0. Different combinations may be possible or better suited for higher airspeeds, but would be best to explore after the final selection of configuration and geometry to initiate fabrication and testing.

The geometry of the converged solution is shown in Figure 5.8, where the scales are in feet. The vehicle looks similar to other successfully flown LALE solar UAVs, with a high $\mathcal{R}$ rectangular wing and slim fuselage. Some of the UAV properties are summarized later in Table 5.8 after the optimization of the clean configuration with payload.

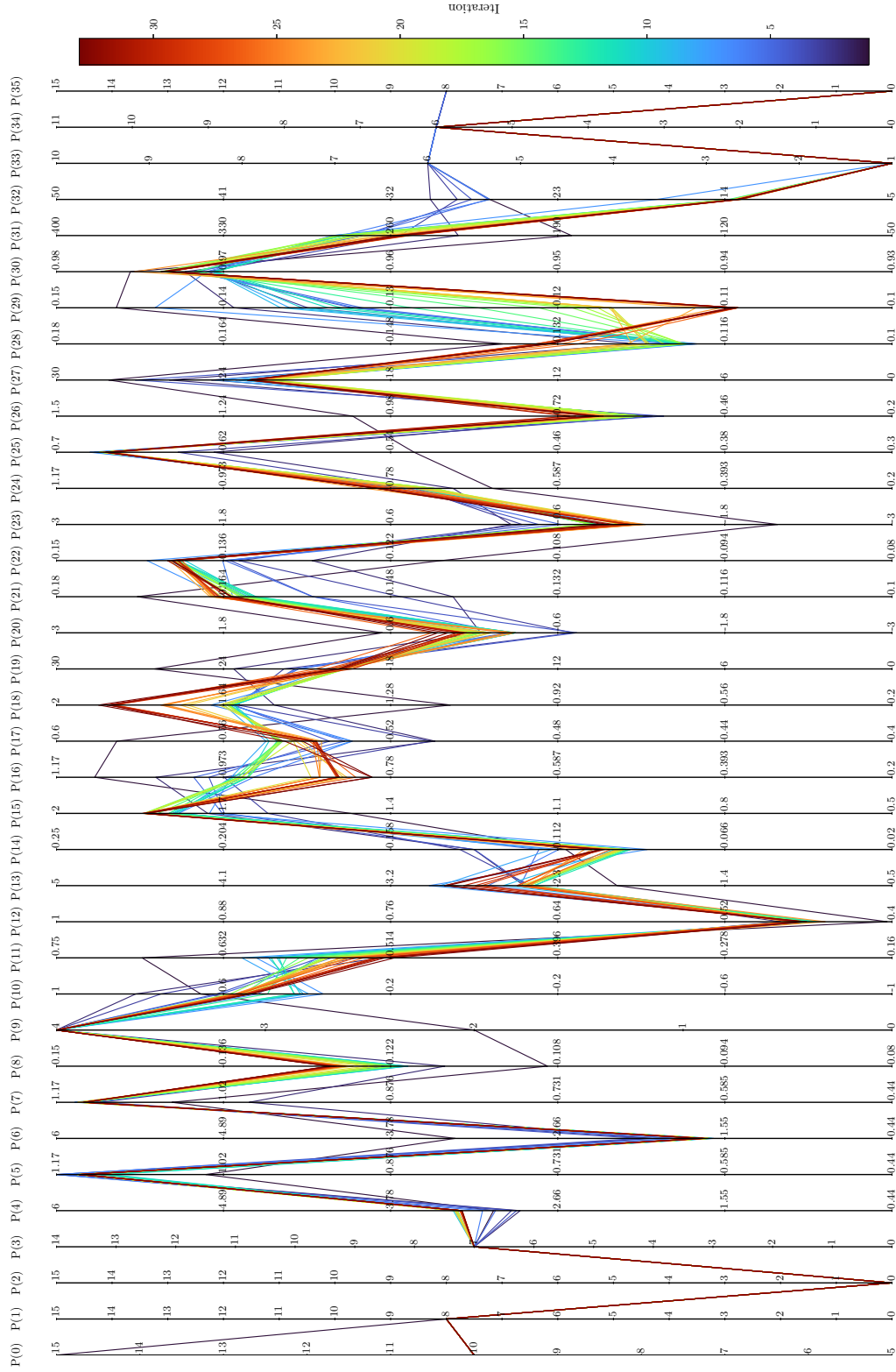


Figure 5.7: Design variables for each iteration for the clean configuration without payload.

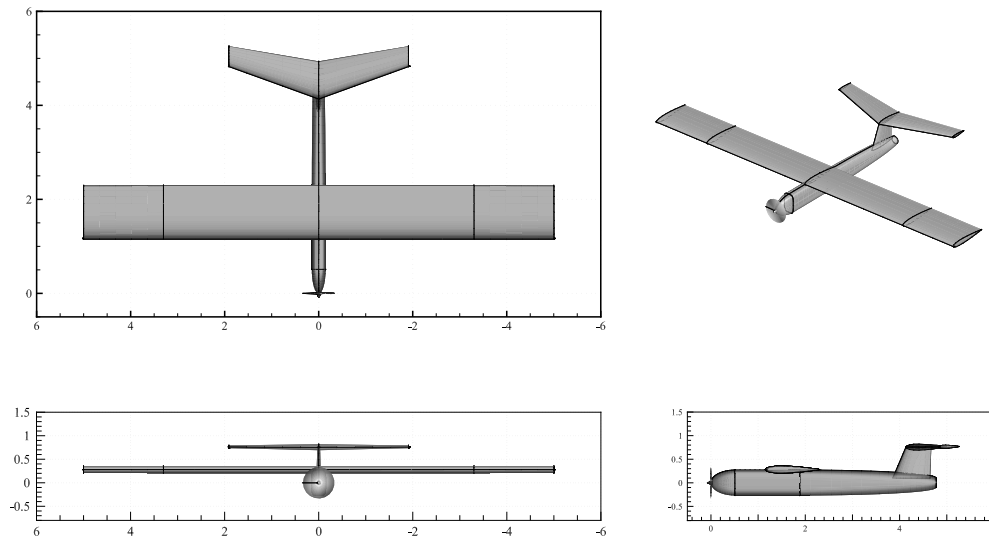


Figure 5.8: Converged solution for the clean configuration without payload. Axes are in feet.

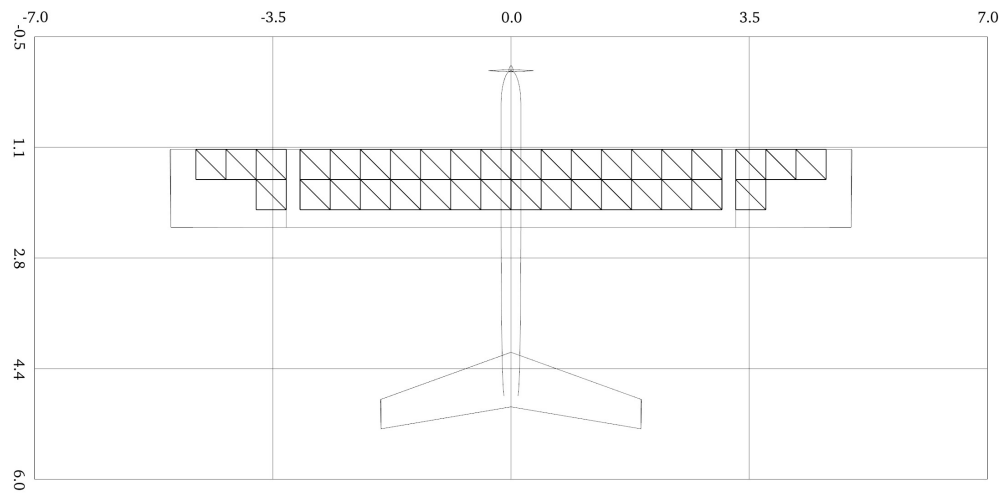


Figure 5.9: Top view with solar panels for the clean configuration without payload. Axes are in feet.

Figure 5.9 shows the top view specifically with the solar cells and wing segments outlined. One of the more interesting results of this final solution is that the wing is not fully covered by solar cells. As the objective function is defined, the design space is more

optimum for vehicles with longer charge margins, or time spent at the maximum SOC. Maximizing the number of solar cells on the wings allows the batteries to charge faster, thereby increasing the overall fitness. The final solution has $P(1)=8$, which gives a fitness of 131.93 h. By changing the number of solar cells in the second row to $P(1)=10$, the fitness increases to 132.25 h. This result shows that the optimizer stagnated at a local maximum. Throughout the various iterations of the optimization problem, the preliminary results were finding similar local maxima with rectangular wings and not fully covering the surface with solar cells. This is likely an artifact of the complexity of the design space where there are multiple discontinuous design variables.

5.3.2 Clean Configuration with Payload

This solution, optimized for payload option 2, converged in 96 iterations, with a fitness of 69.32 h. This fitness level indicates that the converged UAV was able to recharge its batteries in month 6 and 3, then achieving 13.56 h in month 9, with the flight terminating due to $SOC = 30\%$. The fitness for each iteration is plotted in Figure 5.10.

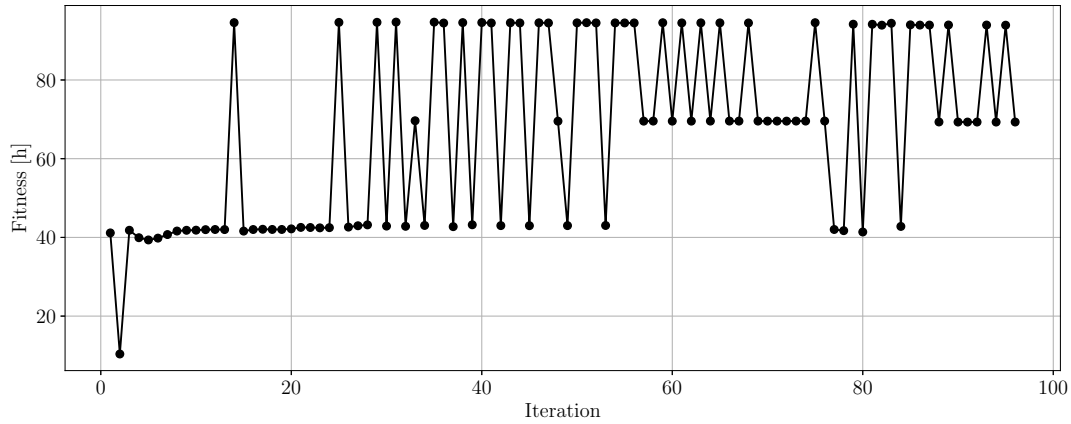


Figure 5.10: Fitness convergence for the clean configuration with payload.

The fitness value appears to oscillate greatly, suggesting that there are major design differences between those iterations. This is not the case, and there are very minor differences between the last 5 iterations, primarily through the wing chord and span. The drastic jumps in fitness are a byproduct of the definition of the fitness function. Iterations with a fitness near 70 are unable to recharge the batteries in month 9 and therefore do not move on to month 12. Iterations with a fitness around 95 are able to recharge in month 9, but not in month 12. The jumps in fitness show a large sensitivity in the design space to minor differences in wing planform properties.

The constraints not meeting the tolerance threshold for each iteration are plotted in Figure 5.11. The values of each constraint are shown in Appendix G in Figure G.1.

In iterations including and after 80, those with an unsatisfied span constraint (constraint 7) are those with the highest fitness values, suggesting that the limited span is

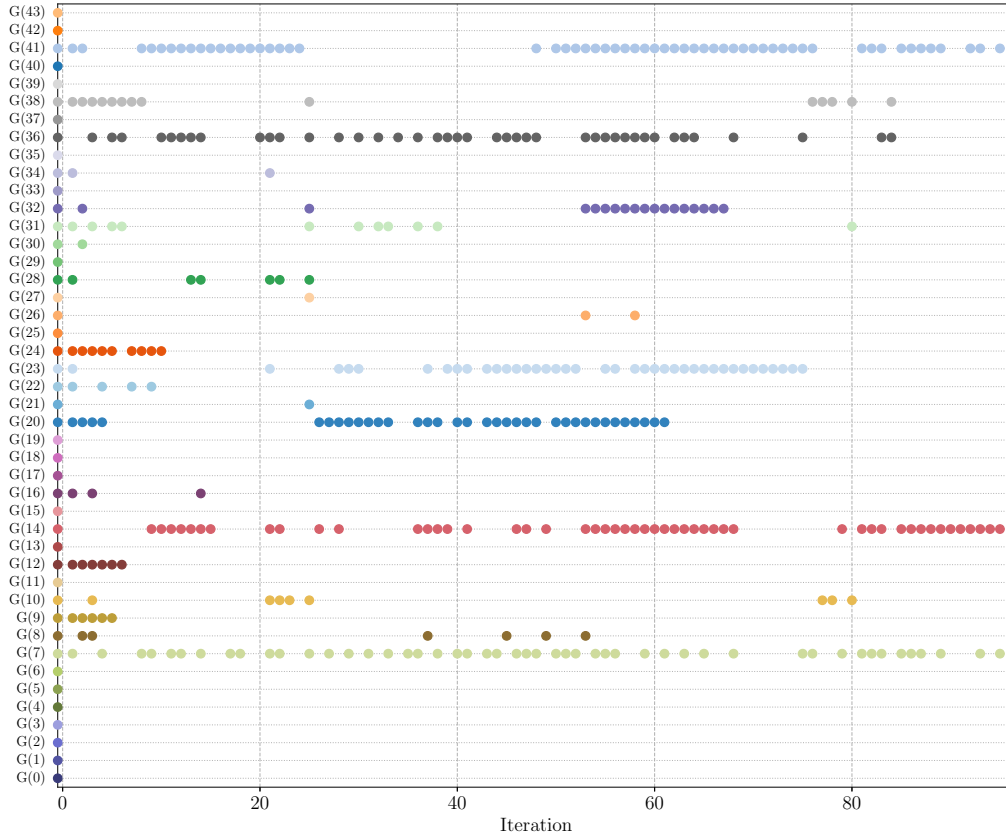


Figure 5.11: Scatter of constraints not meeting the tolerance threshold for the clean configuration with payload.

the greatest barrier to achieving continuous flight at later times of the year. The continuous violation of constraint 14 shows the strong preference to the rectangular wing, rather than a tapered planform.

The evolution of the design variables for the best member of each iteration are shown in a parallel lines plot in Figure 5.12. The variable labels and bounds correspond to Table 5.1. This configuration required more iterations to converge than the previous analysis without payload.

This payload combination has its optimum at $n_{batt} = 6$ and has overall lower fitness than the vehicle without payload. These results are consistent with those of the parametric analysis. The geometry of the converged solution is shown in Figure 5.13.

Similar to the configuration with no payload, this solution also converged without maximizing the number of solar panels on the wing. Figure 5.14 shows the top view with the solar cells on the wing. The final solution has $P(1)=8$, which gives a fitness of 69.32 h. Changing the number of solar cells so that $P(1)=10$ increases the weight and propulsive power more than what is gained through the added solar panels and the fitness decreases to 44.23 h, no longer able to recharge the batteries in month 3. This

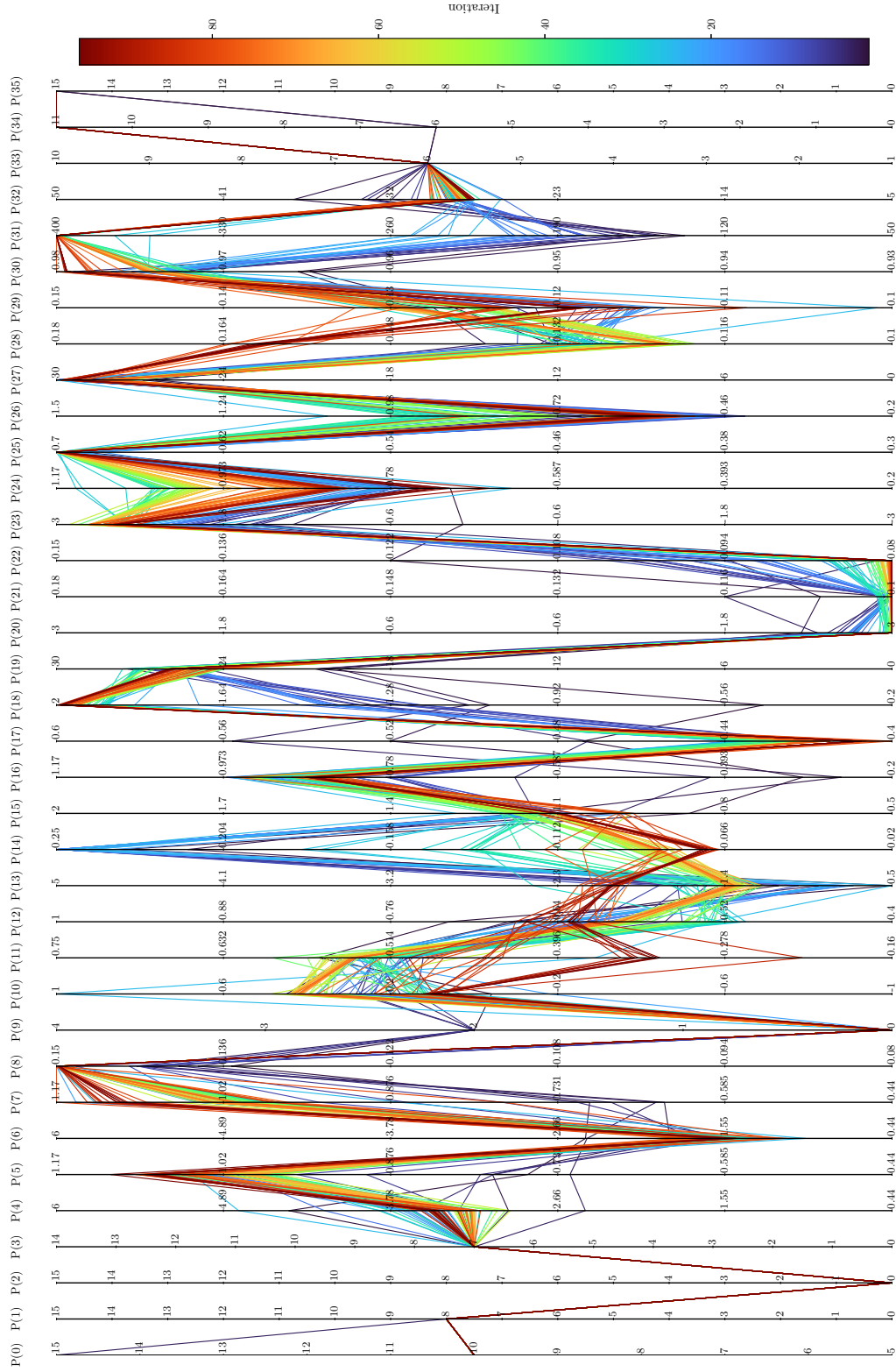


Figure 5.12: Design variables for each iteration for the clean configuration with payload.

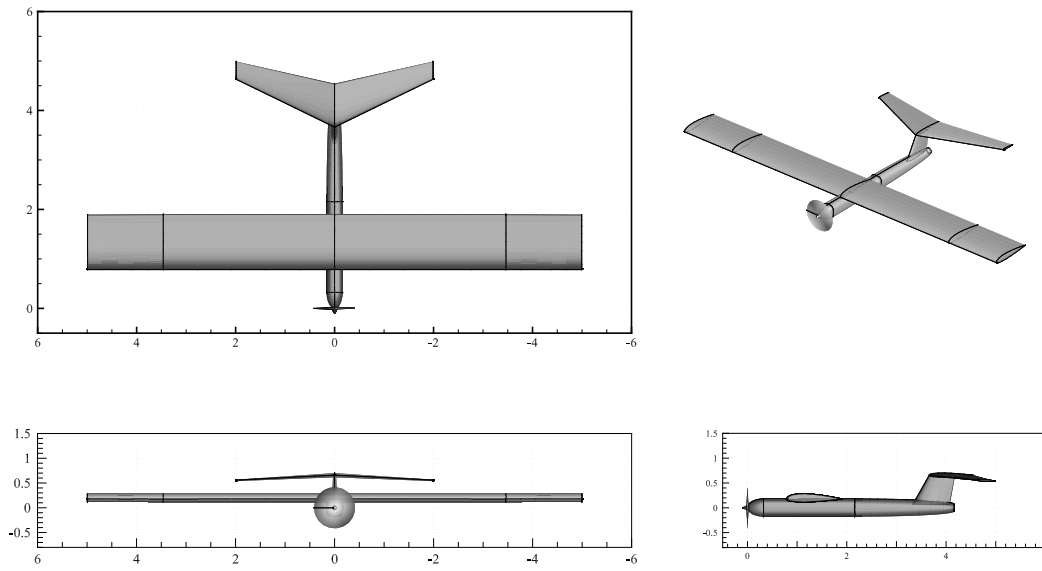


Figure 5.13: Converged solution for the clean configuration with payload. Axes are in feet.

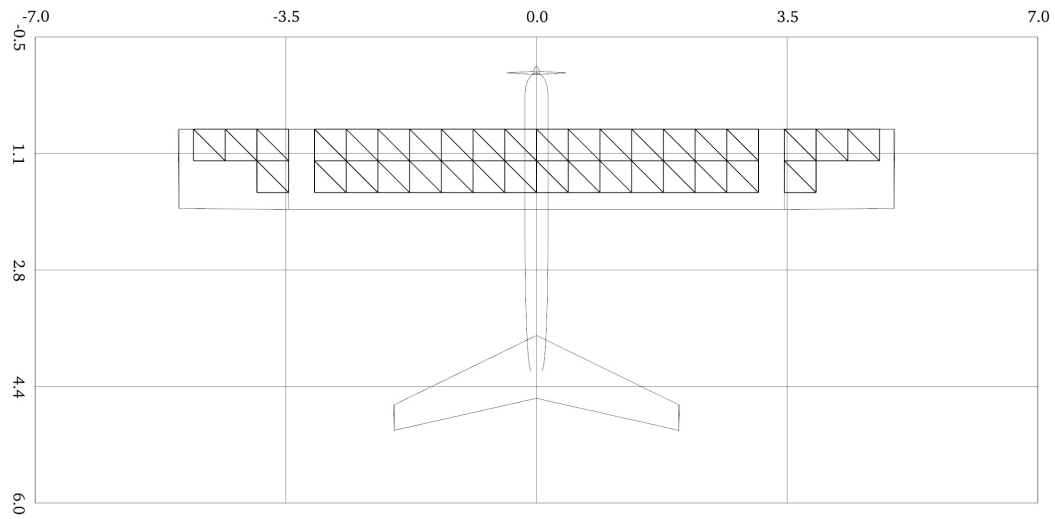


Figure 5.14: Top view with solar panels for the clean configuration with payload. Axes are in feet.

explains why the design is not minimizing the gap between the inner wing panel array and outer wing panel array, as it is already at its maximum fitness.

Comparison of Clean Configuration Solutions

The geometries of the clean wing configurations are shown in Figure 5.15, where the solution without payload is shown in blue and the solution with payload is shown in grey.

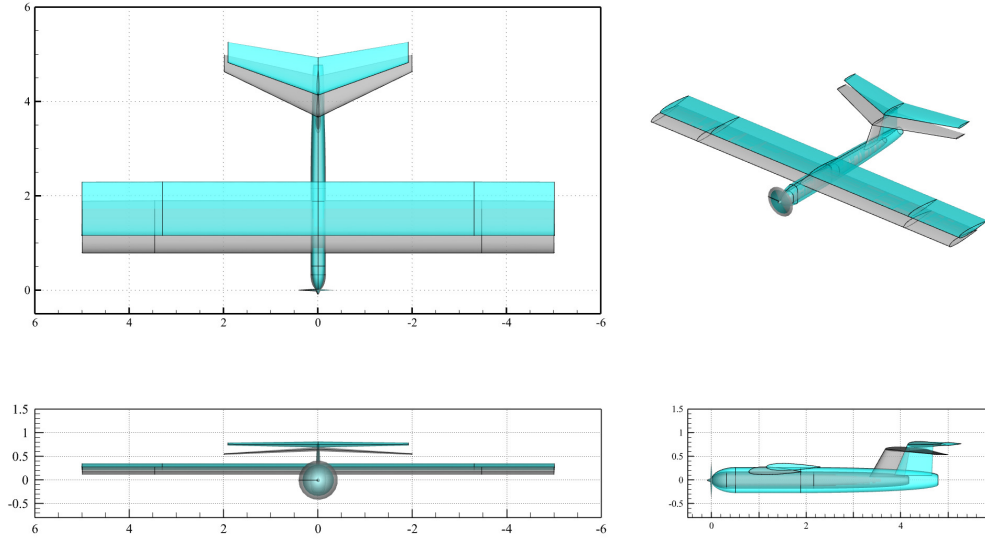


Figure 5.15: Converged solutions for the clean configuration, with payload in grey, without payload in blue. Axes are in feet.

The differences in design variables are more clearly demonstrated through the parallel lines plot, shown in Figure 5.16. The clean configuration with no payload is shown as iteration 33 in blue and the clean configuration with payload is shown as iteration 96 in red. The vehicle without payload is shorter, flies at a slower airspeed and tighter turning radius, uses a smaller propeller, and a smaller total energy capacity compared to the vehicle with payload. A more detailed breakdown of the properties of both vehicles is summarized in Table 5.8. The added payload requires more onboard energy capacity, which increases the weight of the vehicle, thus increasing the total weight and stall speed of the vehicle. The differences in thrust required and cruising airspeed result in different propeller and motor combinations with significantly different power demands.

The fitness is 111.93 h when using the input variables for the vehicle with payload and simulating it without payload, which is able to fully recharge in months 6, 3, and 9, and able to fly 24 consecutive hours in month 12 but not fully recharge. Applying the optimized vehicle without payload to the payload case reduces its flight time to 6.27 h of continuous flight due to the limited onboard energy. The onboard energy capacity, flight speed, and combination of propellers and motors have significant roles in the overall fitness of a vehicle and therefore its ability to achieve LALE solar-powered flight.

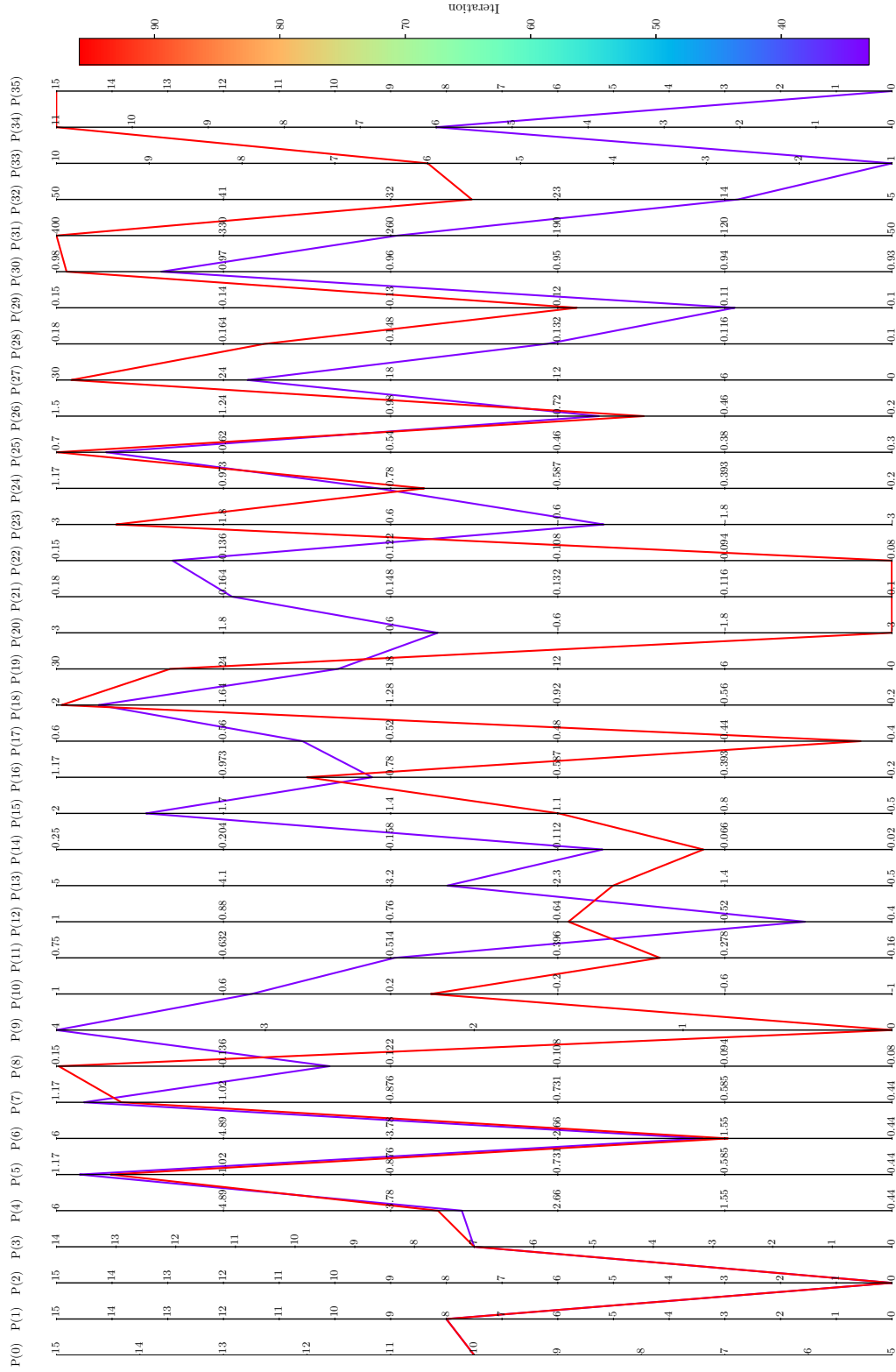


Figure 5.16: Design variables for the clean configuration, with and without payload.

Table 5.8: Clean configuration vehicle properties.

Parameter	Unit	No Payload	With Payload
n_{PV}	[-]	36	36
S	[ft ²]	11.46	11.16
$\bar{c}$	[ft]	1.15	1.12
b	[ft]	10.00	9.98
λ	[-]	0.99	0.99
$\mathcal{R}$	[-]	8.73	8.93
w_{fuse}	[ft]	0.31	0.32
h_{fuse}	[ft]	0.56	0.36
$\mathcal{R}_{HT}$	[-]	6.18	6.45
S_{HT}	[ft ²]	2.36	2.45
$\mathcal{R}_{VT}$	[-]	1.97	1.86
S_{VT}	[ft ²]	0.87	0.74
x_{CG}	[% MAC]	7	34
W_{batt}	[lb _f]	1.2	7.2
W_{struc}	[lb _f]	0.2	0.8
W_{prop}	[lb _f]	0.2	0.5
W_{pl}	[lb _f]	0.5	0.8
W_{oe}	[lb _f]	0.4	1.2
W_0	[lb _f]	2.1	9.3
SM	[% MAC]	51	28
n_z	[-]	1.00	1.00
C_{D0}	[-]	0.0206	0.0189
K	[-]	0.036	0.035
$C_{L_{max}}$	[-]	1.70	1.73
V	[$\frac{ft}{s}$]	13.3	27.6
V_{stall}	[$\frac{ft}{s}$]	9.9	20.7
$V_{P_{min}}$	[$\frac{ft}{s}$]	11.3	24.2
$C_{l,\beta}$	[rad ⁻¹]	-0.063	-0.057
$C_{n,\beta}$	[rad ⁻¹]	0.035	0.029
T_{max}	[lb _f]	1.8	6.2
E	[Wh]	93.2	559.4
$P_p + P_{pl}$	[W]	2.8	27.2
fitness	[h]	131.93	69.32

5.3.3 Nacelles without Payload

This configuration did not converge and the optimizer executed the full 250 iterations. Further inspection of the fitness and constraints is required to determine which iteration is the best result. The fitness is plotted in Figure 5.17. The decrease in fitness in the earlier iterations is due to the algorithm finding solutions with fewer constraints, which is favoured over solutions with higher fitness and more constraints.

The constraints that do not meet the tolerance threshold for each iteration are plotted in Figure 5.18. This plot shows iterations 200 to 250, which contain the most promising

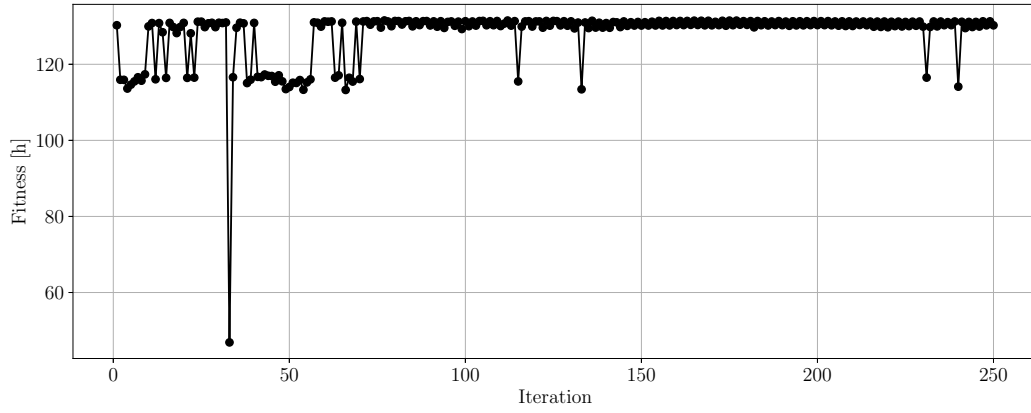


Figure 5.17: Fitness convergence for the nacelle configuration without payload.

results. The values of the constraints not meeting the tolerance threshold and the full scatter plot can be found in Appendix G.

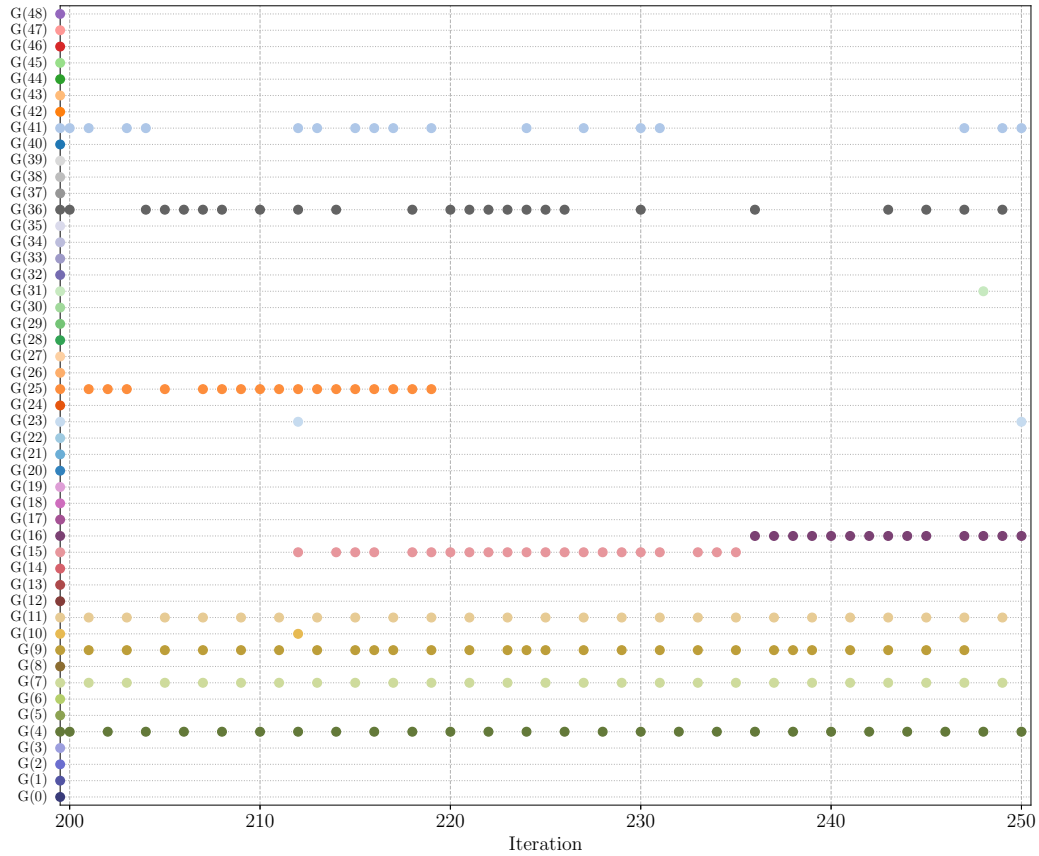


Figure 5.18: Scatter of constraints not meeting the tolerance threshold for the nacelle configuration without payload.

Within the last fifty iterations, iterations 232, 234, 240, 242, 244, 246, and 248 had the least number of violated constraints which may have been appropriately neglected or corrected with minimal modifications. This optimization struggled to maintain the minimum number of solar cells, which is one of the primary constraints violated in the last 15 iterations. As with the clean configuration, the nacelle configuration remains on the boundaries of the chord constraints, maintaining a rectangular planform. Of these iterations, iteration 248 has the best fitness before correcting the solar cell constraint.

A slightly modified iteration 248 is considered as the solution to this optimization. The original geometric properties are maintained, but $P(0)$ and $P(1)$ are modified from $P(0) = P(1) = 8$ to $P(0) = P(1) = 9$. The fitness increases from 130.24 h to 130.72 h. The violated chord constraint ($G(16)$) and vertical tail sweep constraint ($G(31)$) values are small enough to be neglected. The variation in design variables for this optimization was more difficult to decipher due to the large number of iterations. The last 50 iterations are shown in Figure 5.19, whereas the full 250 iterations can be found in Appendix G.

The modified solution is shown in Figure 5.20, with the solar panels shown in Figure 5.21. The fuselage is no longer required to house the battery packs, therefore is able to minimize its drag and be smaller in diameter. The nacelle height and wing geometry appear to be minimizing the discontinuity from the nacelle over the upper surface, which minimizes the loss of effective lifting span from the nacelles. The outer wing thickness is constrained by $G(20)$, requiring large enough thickness to embed a servo motor within the wing, which could not be achieved with the smaller thickness of the inner wing segment. The root of the wing undergoes more bending than the tip, therefore should not be the thinnest part of the wing from a structures perspective. However, there is no requirement to store fuel in the wing and more of the internal volume can be allocated to structure. Despite losing some of its effective lifting surface to the nacelles, this vehicle did not maximize the span fully. This is not an accurate reflection of the design not being more optimal at larger spans, since the final iterations alternated between not satisfying the minimum number of solar panels and the maximum span constraint.

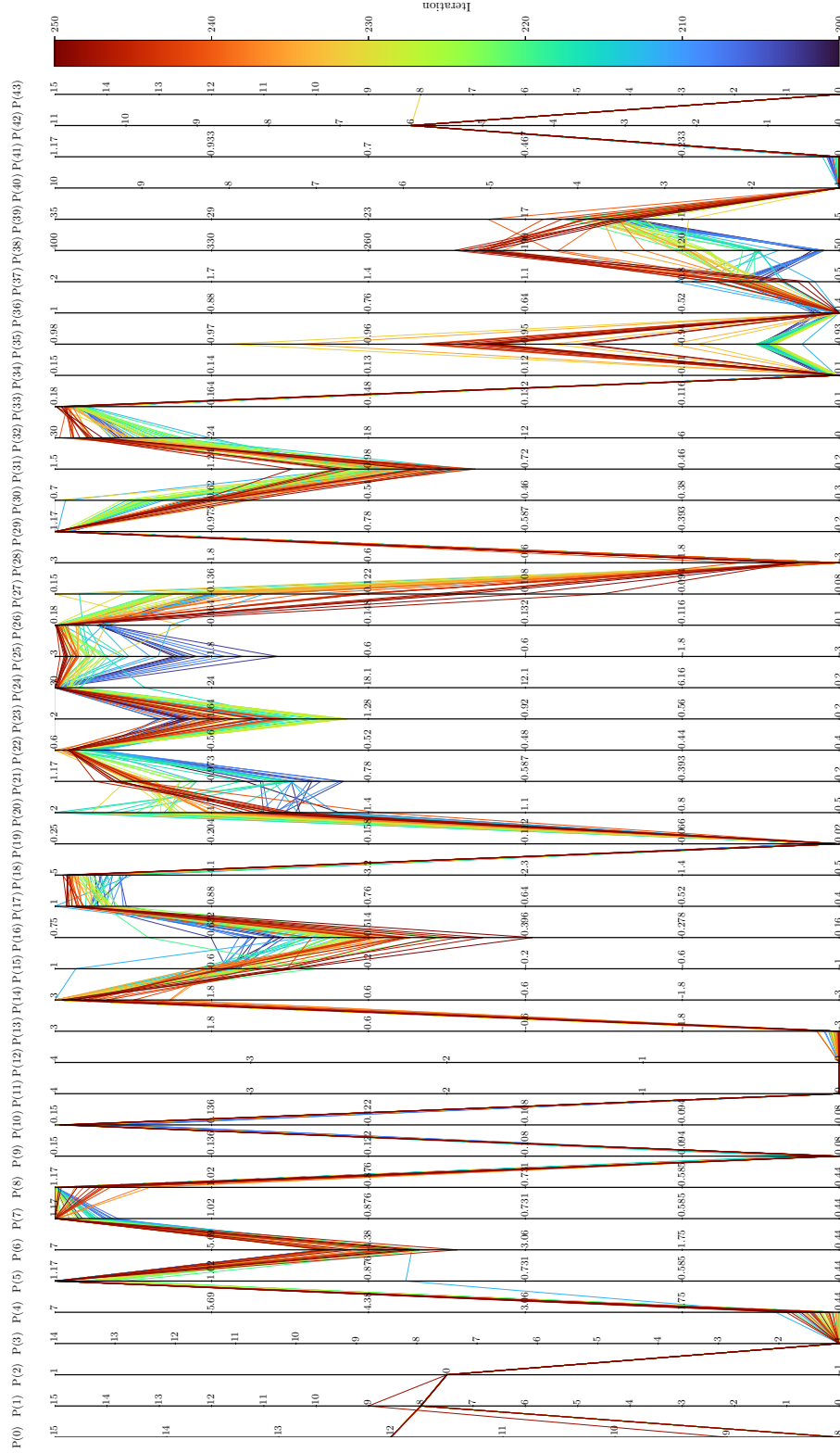


Figure 5.19: Design variables for iterations 200 to 250 for the nacelle configuration, no payload.

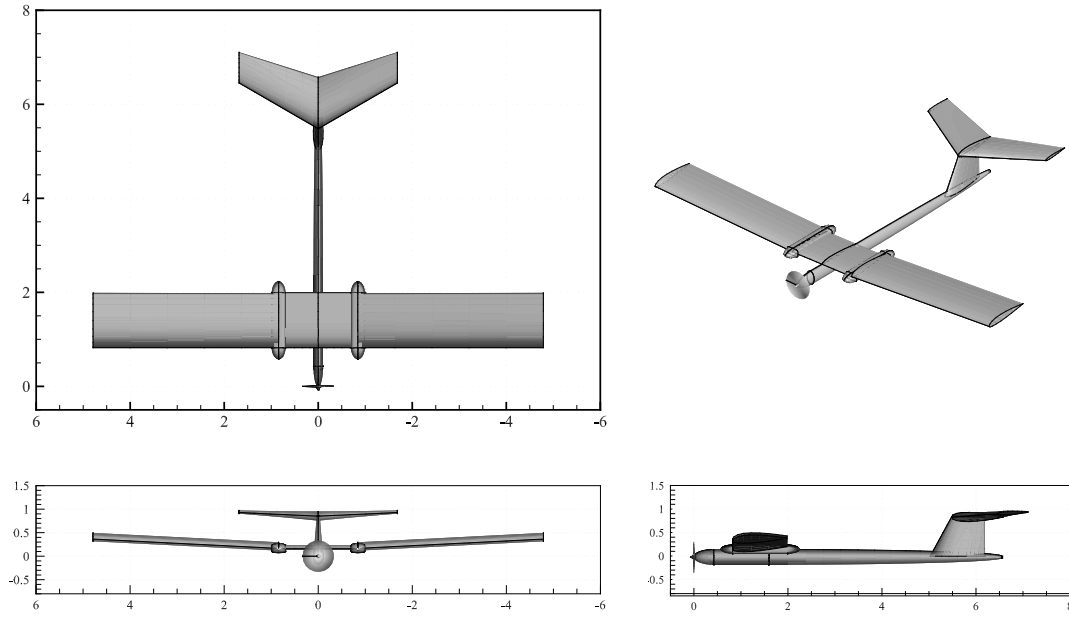


Figure 5.20: Solution for the nacelle configuration without payload. Axes are in feet.

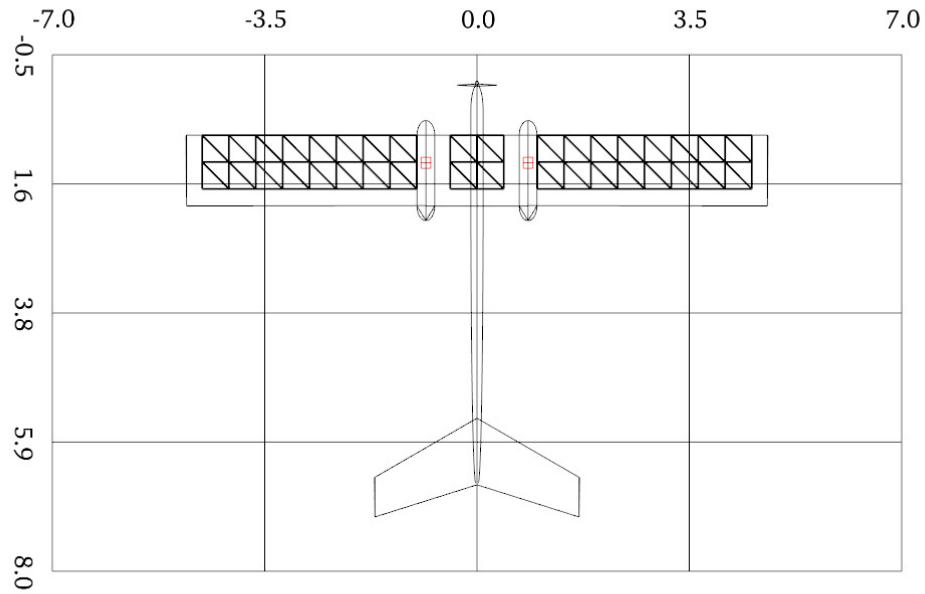


Figure 5.21: Top view with solar panels for the nacelle configuration, without payload. Axes are in feet.

5.3.4 Nacelles with Payload

This configuration did not converge and the optimizer executed the full 250 iterations. The fitness is plotted in Figure 5.22, where the decrease in fitness is due to the algorithm finding solutions with fewer constraint violations. All previous results have shown promising configurations for LALE solar-powered flight. With several iterations of analysis, a promising solution for the nacelle configuration with payload was not found as the problem was defined.

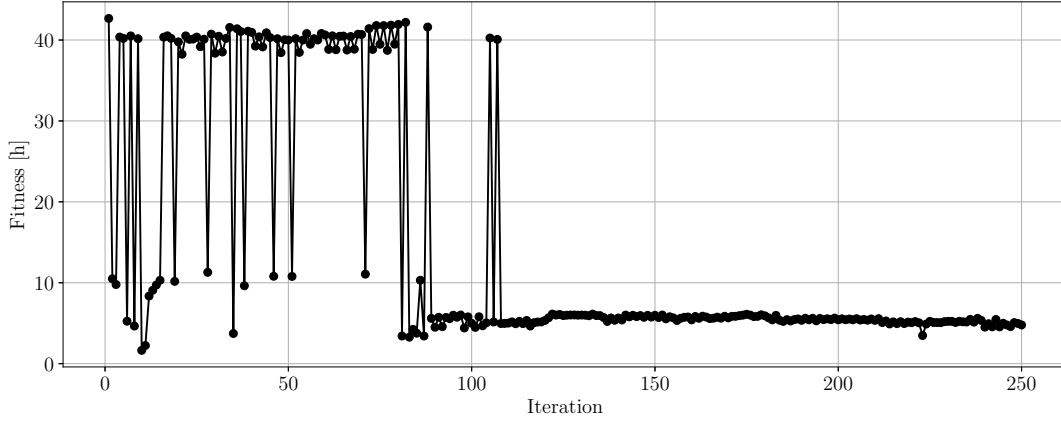


Figure 5.22: Fitness convergence for the nacelle configuration, with payload.

A closer inspection of the constraints is required to determine which iteration is closest to satisfying the defined problem. The constraints not meeting the tolerance threshold are plotted in a scatter plot in Figure 5.23 for iterations 200 to 250. The values of the constraints not meeting the tolerance threshold and the full scatter plot are presented in Appendix G.

Within the last 50 iterations, the best member of the optimization particle swarm is bound by the wing geometry and panel constraints, oscillating between violating span constraints ($G(7, 9, 11)$) and the minimum number of solar panels ($G(4)$). The chord constraints are only violated to a minor degree, since the optimizer tends to a rectangular planform, similar to the previous cases. This configuration specifically struggled with the vertical tail $\mathcal{R}$ constraint ($G(36)$) and the upper roll stability constraint ($G(43)$). Both are small enough values that they could be reasonably neglected. The design variables for iterations 200 to 250 are plotted in Figure 5.24, with all iterations plotted in Appendix G.

The last iteration is taken to be the final result of this optimization, with a minor modification to the number of solar panels in the same way that the nacelle configuration with no payload was modified. By changing $P(0) = P(1) = 8$ to $P(0) = P(1) = 9$, the fitness decreases from 4.79 h to 4.75 h. The vehicle is unable to fly long enough to have any benefits of the increased capacity to collect solar power and instead is penalized in fitness from the added weight of the panels. Because this vehicle does not fly long enough to see daylight and recharge, there is no benefit to the design space in having

solar cells on the vehicle, explaining the tendency to violate the minimum solar panels constraint. The final geometry is shown in Figure 5.25, with the modified solar array shown in Figure 5.26.

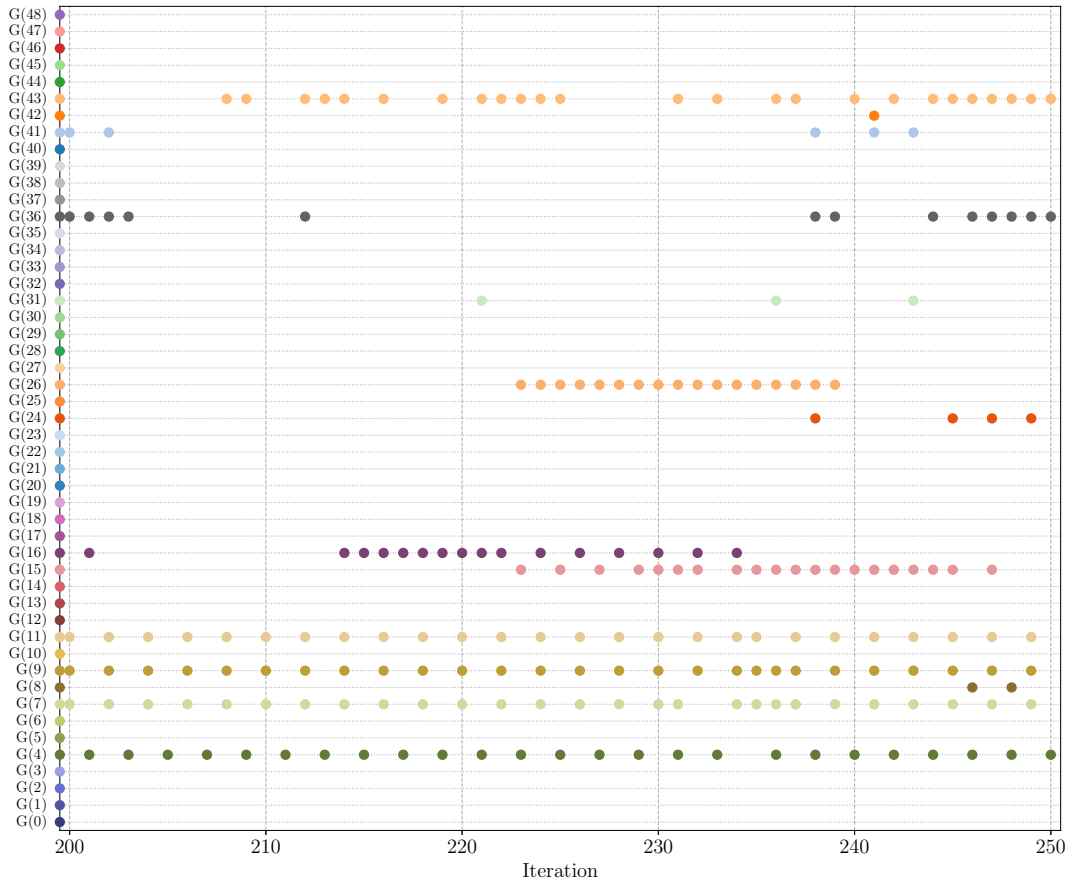


Figure 5.23: Scatter of constraints not meeting the tolerance threshold for the nacelle configuration, with payload.

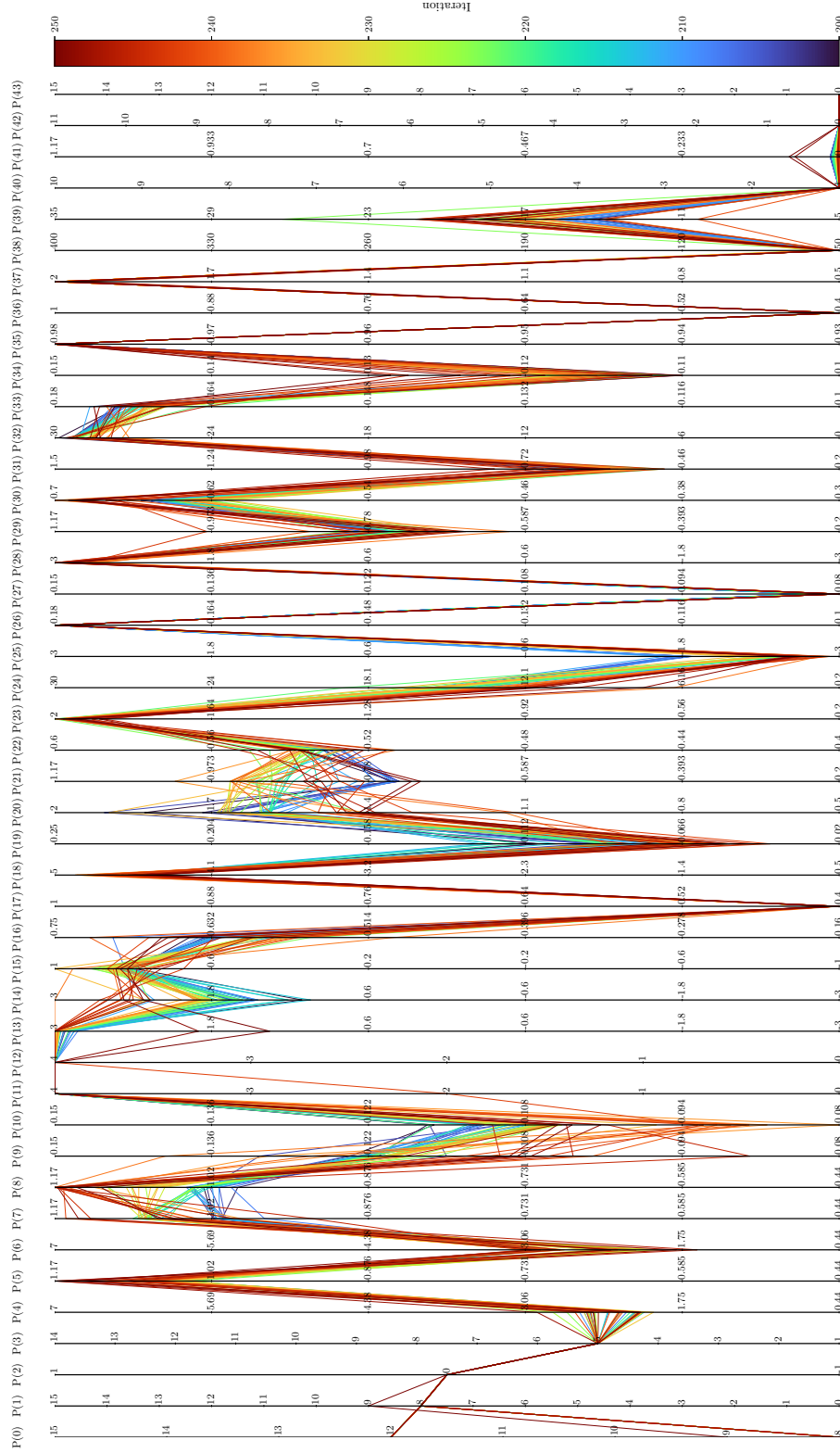


Figure 5.24: Design variables for iterations 200 to 250 for the nacelle configuration, with payload.

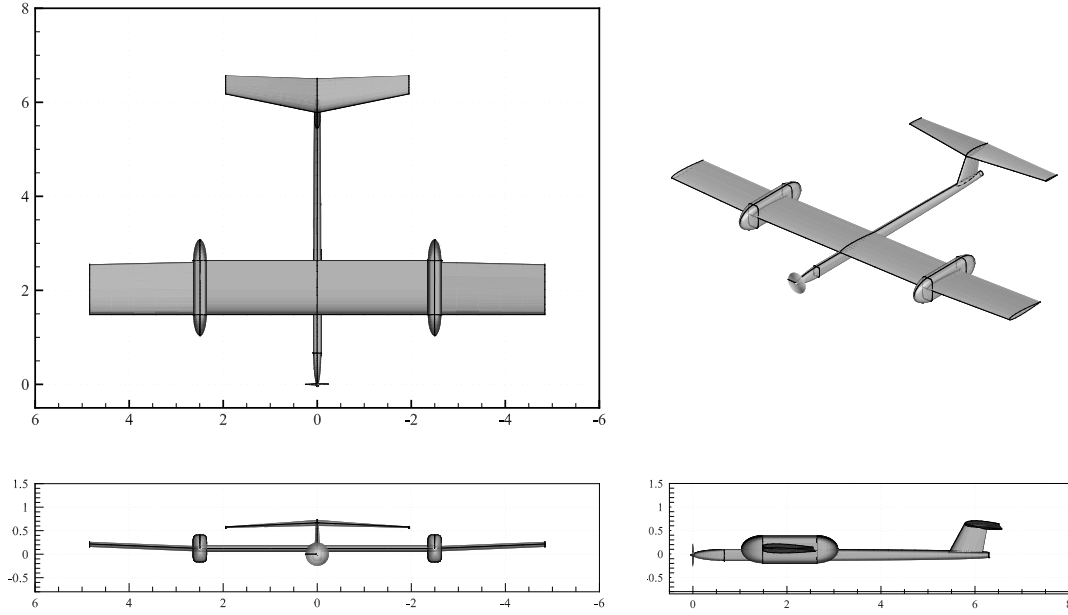


Figure 5.25: Solution for the nacelle configuration, with payload. Axes are in feet.

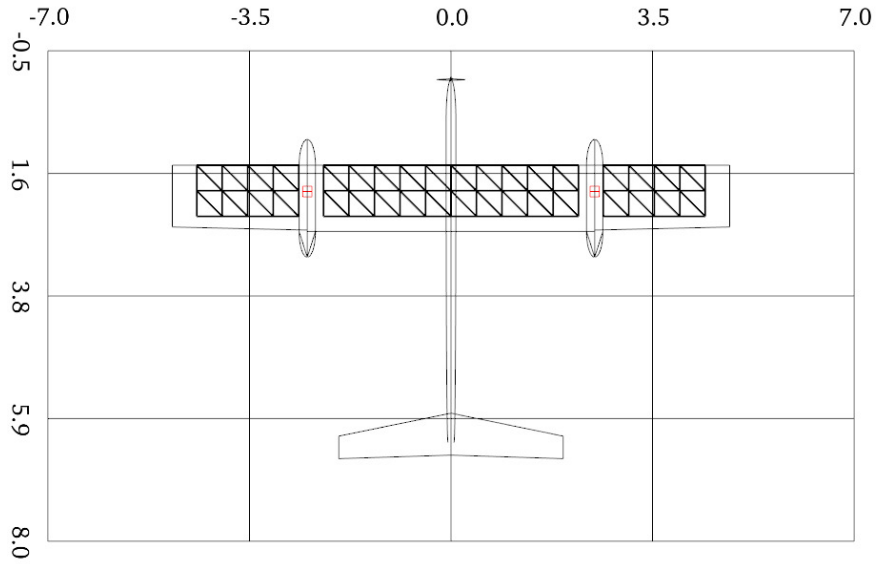


Figure 5.26: Top view with solar panels for the nacelle configuration, with payload. Axes are in feet.

When the geometry for the payload case is flown as if it has no payload, the fitness increases from 4.75 h to 45.06 h, showing the drastic effect of payload power consumption

on the overall fitness. The parallel lines plot shows that the optimizer did not find any solutions with different propeller and motor combinations (P(42, 43)) that improved the fitness. Its final combination of propeller and motor do not appear in the best iterations of any of the other vehicles, indicating the solution may be within a large local maxima. Using the same propulsion combinations as the other solutions resulted in minor increases in fitness, none drastic enough to make the nacelle configuration viable for LALE solar-powered flight with payload.

Results from the parametric analysis in Chapter 4 shows that larger spans generally have greater performance than those with smaller spans. The sensitivity to the fitness and span is demonstrated in the clean configuration analysis with payload, where minor changes in the span have significant impacts on the fitness. These results support the poor fitness of the nacelle configuration with payload, as the span is insufficient to support the required added weight and power draw with the additional surface area and therefore drag of the nacelles.

Comparison of Nacelle Configuration Solutions

This section specifically compares the converged solutions for the nacelle configuration optimizations with and without payload. The geometry of both are shown in Figure 5.15, where the solution without payload is shown in blue and the solution with payload is shown in grey.

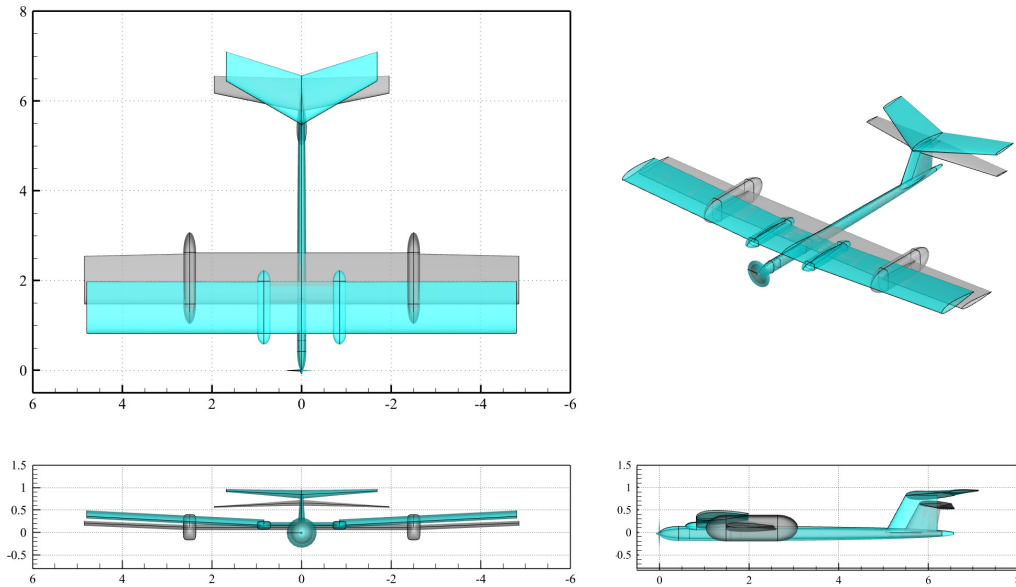


Figure 5.27: Converged solutions for the nacelle configuration, with payload in grey, without payload in blue. Axes are in feet.

The differences in design variables are more clearly demonstrated through the parallel lines plot, shown in Figure 5.28. The configuration with payload is shown as iteration 250 in red and the configuration without payload is shown in blue as itera-

tion 248. A more detailed breakdown of the nacelle configuration vehicle characteristics are summarized in Table 5.9.

Table 5.9: Nacelle configuration vehicle properties.

Parameter	Unit	No Payload	With Payload
n_{PV}	[-]	36	36
S	[ft ²]	11.16	10.88
$\bar{c}$	[ft]	1.16	1.12
b	[ft]	9.58	9.68
λ	[-]	1.00	0.91
$\mathcal{R}$	[-]	8.23	8.62
w_{nac}	[ft]	0.30	0.30
h_{nac}	[ft]	0.18	0.59
w_{fuse}	[ft]	0.21	0.17
h_{fuse}	[ft]	0.33	0.24
$\mathcal{R}_{HT}$	[-]	3.85	6.98
S_{HT}	[ft ²]	2.95	2.18
$\mathcal{R}_{VT}$	[-]	1.78	2.03
S_{VT}	[ft ²]	1.56	0.74
x_{CG}	[% MAC]	0.19	0.17
W_{batt}	[lb _f]	1.2	1.2
W_{struc}	[lb _f]	0.3	0.2
W_{prop}	[lb _f]	0.2	0.2
W_{pl}	[lb _f]	0.5	0.8
W_{oe}	[lb _f]	0.5	0.4
W_0	[lb _f]	2.2	2.4
SM	[% MAC]	73	56
n_z	[-]	1.00	1.03
C_{D0}	[-]	0.0256	0.0255
K	[-]	0.039	0.036
$C_{L_{max}}$	[-]	1.65	1.60
V	[$\frac{ft}{s}$]	13.6	20.3
V_{stall}	[$\frac{ft}{s}$]	10.4	11.1
$V_{P_{min}}$	[$\frac{ft}{s}$]	11.2	11.6
$C_{l,\beta}$	[rad ⁻¹]	-0.103	-0.039
$C_{n,\beta}$	[rad ⁻¹]	0.107	0.037
T_{max}	[lb _f]	1.8	0.7
E	[Wh]	93.2	93.2
$P_p + P_{pl}$	[W]	3.5	12.8
fitness	[h]	130.72	4.75

The vehicle with payload was unable to find a feasible design at $n_{batt} = 6$, which was expected as its optimum from the parametric comparison results and the results of the clean configuration. Both vehicles have very similar drag characteristics, but their differing feasibility for LALE solar-powered flight is largely linked to the propulsion efficiency and power draw, which is also related to the flight airspeed.

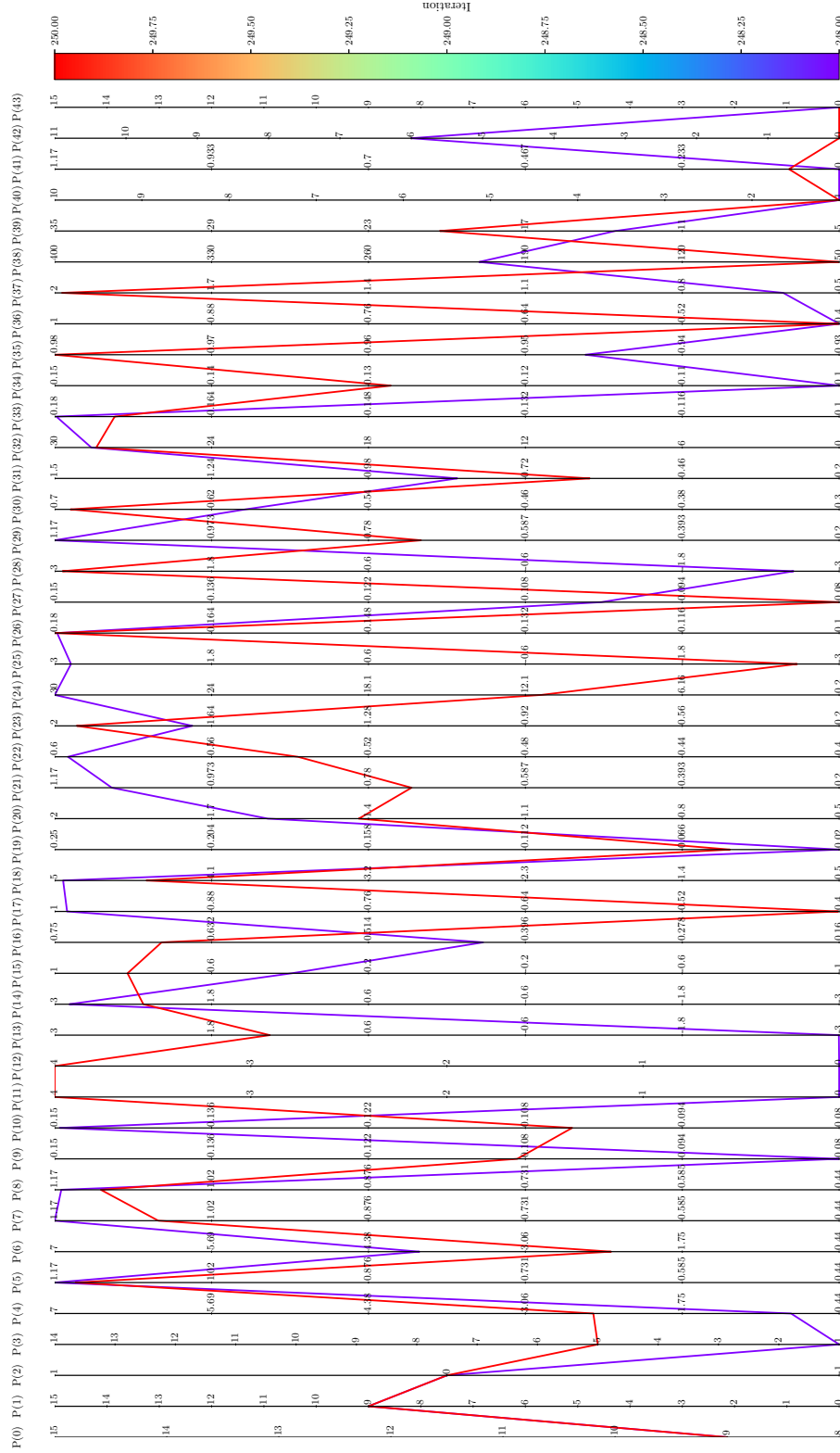


Figure 5.28: Design variables for the nacelle configuration, with and without payload.

5.4 Initial Optimization Summary

With and without payload, the span constraint is a notable limiting factor. Allowing a larger span has the potential to widen the window of opportunity for achieving LALE solar flight through longer portions of the year, particularly when considering more demanding payloads. More solar cells can be integrated to the wing, increasing the rate of charge and allowing for larger energy margins throughout the day by increasing t_{cm} . An additional optimization of each payload option and configuration is performed with more relaxed span constraints in Appendix H.

Both payload cases for the clean configuration and the no payload case for the nacelle configuration have a minimum power airspeed lower than the optimized airspeed. Intuitively, it is anticipated that the optimizer would favour the minimum power airspeed since it would require the least amount of thrust (and therefore power) to maintain steady level flight. The faster airspeed is a direct result of the maximum lift coefficient constraint, which effectively enforces $V \geq 1.3V_{stall}$. All results have a minimum power airspeed that is lower than $1.3V_{stall}$, therefore the optimizer is unable to fly the vehicle at the airspeed for absolute minimum power draw in flight. It is important to match the maximum propulsive efficiency airspeed to the minimum power airspeed of a UAV. The optimizer has found the best combinations for each vehicle out of the propellers and motors identified in Tables 5.6 and 5.7.

All four vehicles have very high static margins. No upper constraint was included in the analysis as the lower value of the static margin has been the limiting factor in other designs. To maintain pitch or to induce manoeuvres, large control surfaces and deflection angles may be required throughout significant portions of the flight. Trim drag, which has not been considered in the optimization, can have great penalizing effects on the fitness or overall endurance of the vehicle. The added drag requires additional thrust and therefore increases the power draw of the propulsion system, reducing the fitness. The analysis is refined in the next section by adding an upper constraint to the static margin.

Furthermore, all four vehicles have high $\frac{W_{batt}}{W_0}$ fractions, with the clean wing with payload UAV as high as $\frac{W_{batt}}{W_0} = 0.77$. For this fraction and the calculated structural weights to be possible, a high strength to weight ratio of the selected materials is required. The intention to 3D print the final vehicle with high rigidity and low density plastics allows the outer structure of the aircraft to be maintained by the printed components while using a carbon fibre rod as the primary structural components. Lab results of concurrent research supports the feasibility of this design approach and these structural weights, where the combined materials offer a desirable high strength to weight ratio.

5.5 Refined Results

This section shows the results for refining the optimization, where the static margin constraints are updated. The changes to the constraints for the clean configuration from Table 5.2 are updated by those in Table 5.10. The changes to the constraints for the nacelle configuration from Table 5.4 are updated by those in Table 5.11.

Table 5.10: Updated constraints for the clean configuration optimization.

Label	Constraint	Value
G(34)	Static margin [% MAC]	≥ 5.0
G(44)	Static margin [% MAC]	≤ 25

Table 5.11: Updated constraints for nacelle configuration optimization.

Label	Constraint	Value
G(39)	Static margin [% MAC]	≥ 5.0
G(49)	Static margin [% MAC]	≤ 25

5.5.1 Clean Configuration without Payload

This configuration converged in 28 iterations with a fitness of 131.45 h. This final solution has only a minor decrease in fitness from the original optimization, with improved stability properties. The fitness for each iteration is plotted in Figure 5.29. Note that the vertical scale is small and the variation in fitness between each iteration is equally small.

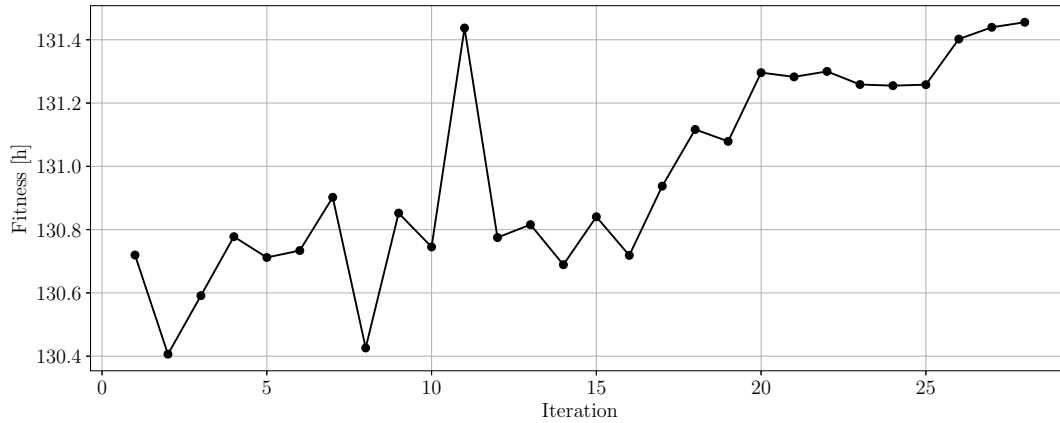


Figure 5.29: Fitness convergence for the refined clean configuration without payload optimization.

The scatter plot of the constraints not meeting the tolerance threshold are shown in Figure 5.30. The values of the constraints are found in Figure G.8 in Appendix G. The final geometry is shown in Figure 5.31 in comparison with the original results. Throughout this section, the original result from Section 5.3 will be shown in blue, with the final geometry with the upper constraint on the static margin shown in grey.

The design variables for the original and the refined clean configuration optimization with no payload are shown in Figure 5.32. The original vehicle is shown in red as iteration 33 and the refined vehicle is shown in blue as iteration 28. The evolution of the design variables for the full optimization of the refined constraints is in Appendix G,

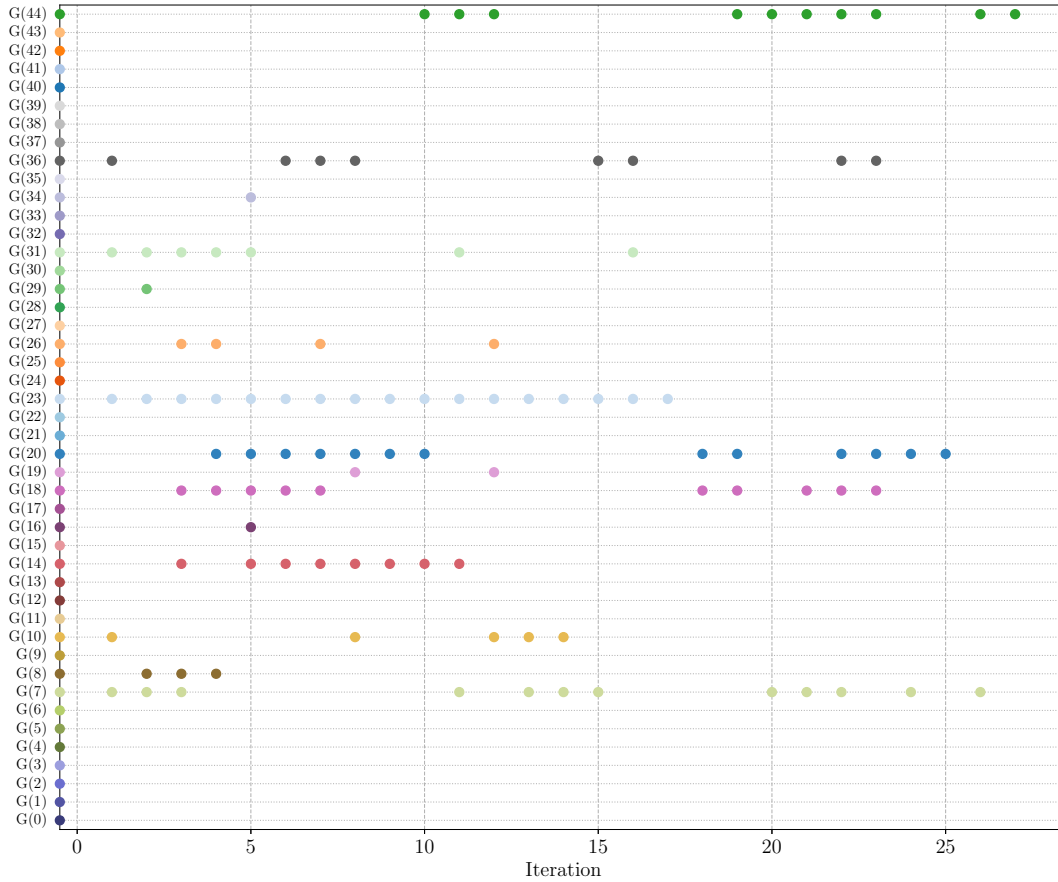


Figure 5.30: Scatter of constraints not meeting the tolerance threshold for the refined clean configuration without payload optimization.

Figure G.9. Further vehicle properties are summarized in Table 5.12, comparing the original vehicle to the refined, final solution.

The final vehicle has a near-identical wing planform, the same solar cell layout, a smaller horizontal stabilizer, a larger vertical stabilizer, and an improved static margin. The final vehicle converged at a different propeller and motor combination, which increases the weight and power draw of the vehicle. If the final vehicle is flown with the propulsion system of the original vehicle, the fitness increases marginally from 131.45 h to 131.89 h.

As with the original vehicle, the refined optimization did not maximize the solar cell coverage on the wing planform. By changing $P(1)=8$ to $P(1)=10$, the fitness increases from 131.45 h to 131.84 h, further supporting that the solution converged in a local maximum. The small increase in performance shows that the performance of the vehicle is not as strongly tied to maximizing the solar area on the vehicle. This conclusion may not hold true when applied to real flight conditions, where the additional four panels may prove highly beneficial when the power draw of control surfaces and electrical losses are no longer negligible.

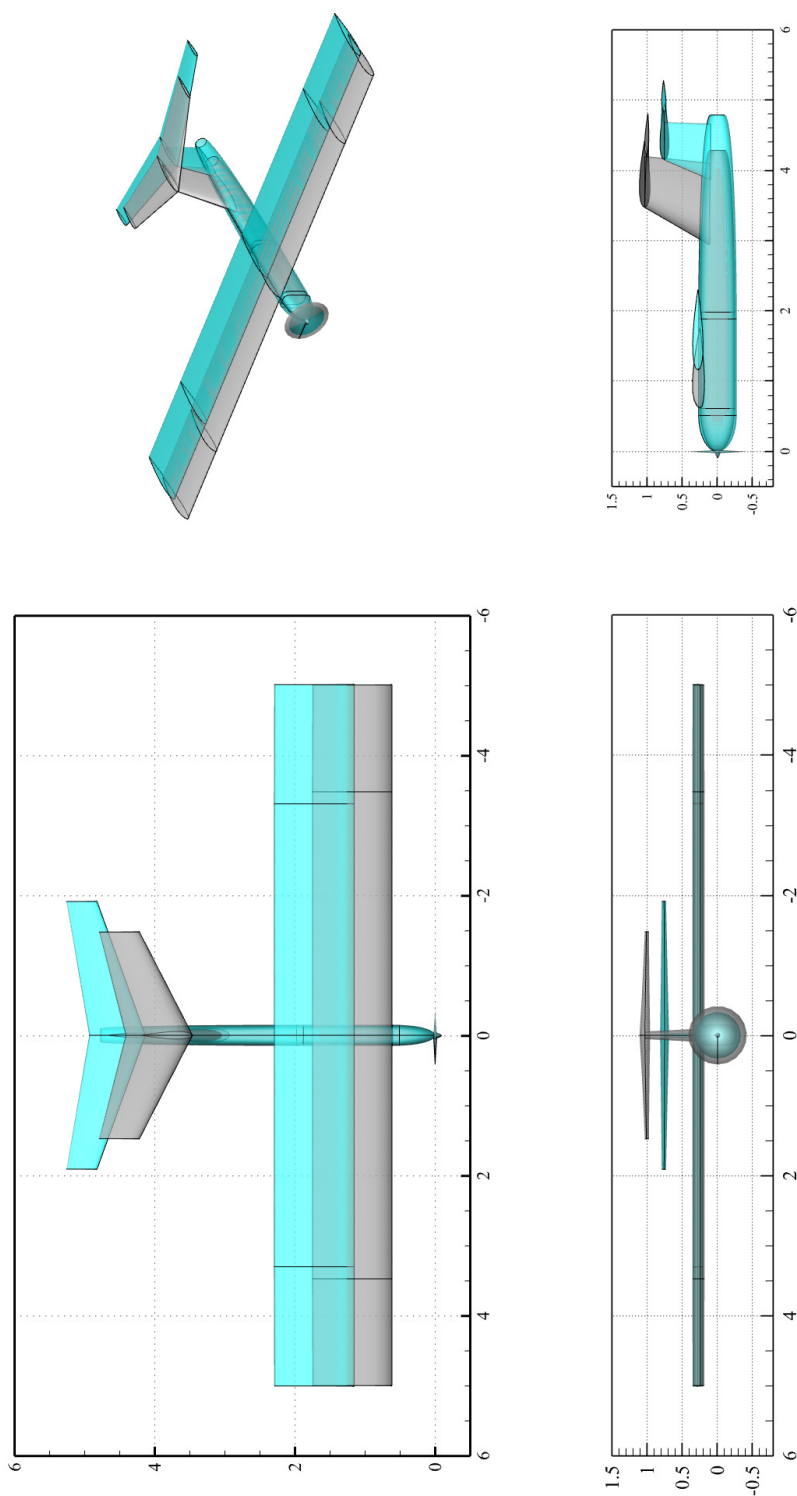


Figure 5.31: Converged solutions for the clean configuration without payload optimization. The initial result is in blue, the final solution is in grey. Axes are in feet.

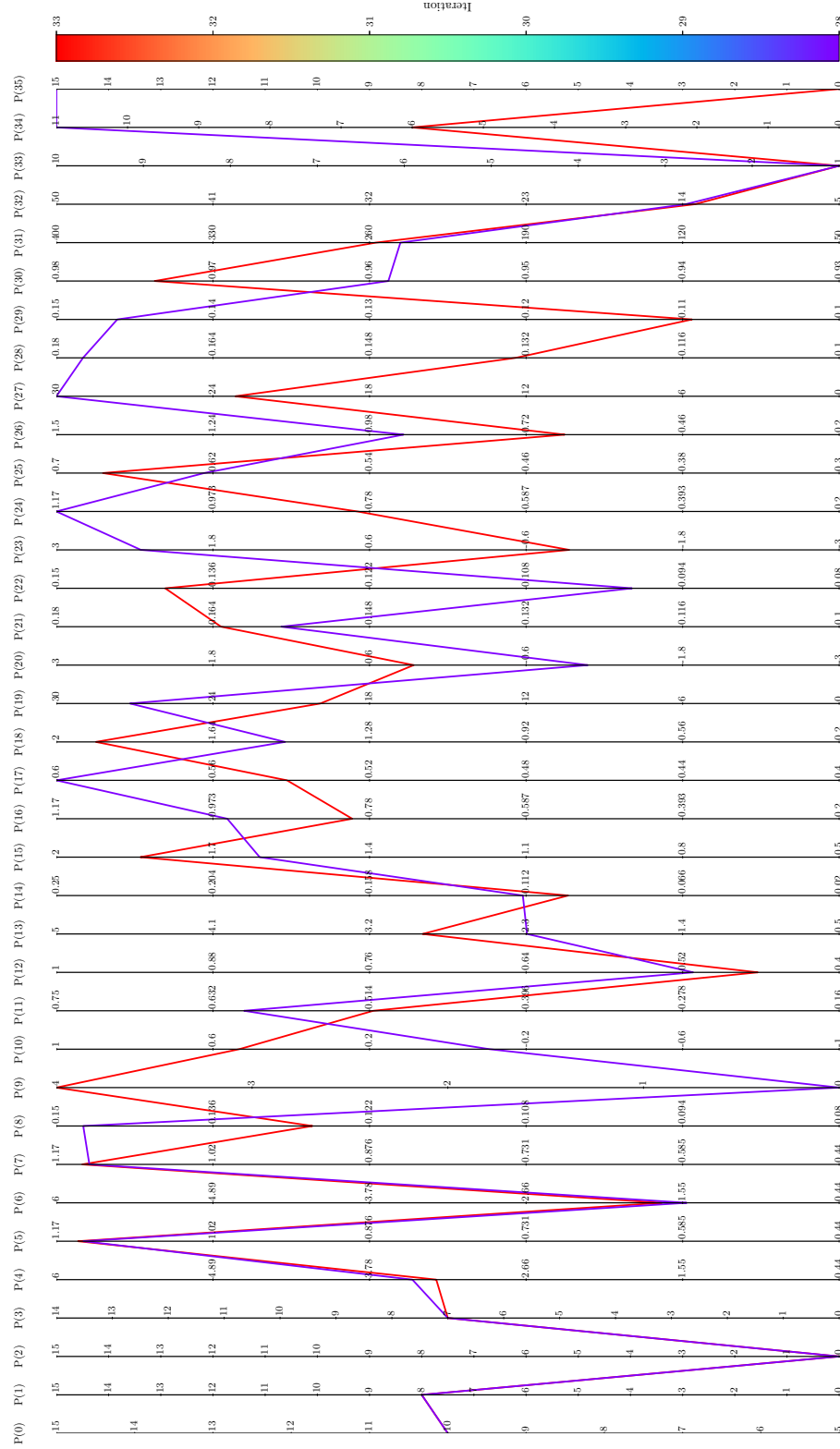


Figure 5.32: Comparison of design variables for the clean configuration without payload optimizations.

Table 5.12: Clean configuration without payload vehicle properties.

Parameter	Unit	Original	Refined
n_{PV}	[-]	36	36
S	[ft ²]	11.46	11.40
$\bar{c}$	[ft]	1.15	1.14
b	[ft]	10.00	9.99
λ	[-]	0.99	0.99
$\mathcal{R}$	[-]	8.73	8.76
w_{fuse}	[ft]	0.31	0.31
h_{fuse}	[ft]	0.56	0.50
$\mathcal{R}_{HT}$	[-]	6.18	3.86
S_{HT}	[ft ²]	2.36	2.25
$\mathcal{R}_{VT}$	[-]	1.97	1.95
S_{VT}	[ft ²]	0.87	1.75
x_{CG}	[% MAC]	7	32
W_{batt}	[lb _f]	1.2	1.2
W_{struc}	[lb _f]	0.2	0.2
W_{prop}	[lb _f]	0.2	0.5
W_{pl}	[lb _f]	0.5	0.5
OEW	[lb _f]	0.4	0.7
W_0	[lb _f]	2.1	2.4
SM	[% MAC]	51	24
n_z	[-]	1.00	1.00
C_{D0}	[-]	0.0206	0.0207
K	[-]	0.036	0.038
$C_{L_{max}}$	[-]	1.70	1.73
V	[$\frac{ft}{s}$]	13.3	13.8
V_{stall}	[$\frac{ft}{s}$]	9.9	10.4
$V_{P_{min}}$	[$\frac{ft}{s}$]	11.3	12.1
$C_{l,\beta}$	[rad ⁻¹]	-0.063	-0.072
$C_{n,\beta}$	[rad ⁻¹]	0.035	0.064
T_{max}	[lb _f]	1.8	6.6
E	[Wh]	93.2	93.2
$P_p + P_{pl}$	[W]	2.8	3.1
fitness	[h]	131.93	131.45

5.5.2 Clean Configuration with Payload

This configuration did not converge, using the allocated 250 iterations to search the design space. The fitness and constraints are analyzed in conjunction to determine the best vehicle. The fitness for each iteration is presented in Figure 5.33. As with the original result, small changes to the span has a significant effect on the fitness value. In this case, it is iterations with spans exceeding the span constraint that have the greatest fitness values but are infeasible due to the extent to which they violate G(7) (the total wingspan constraint).

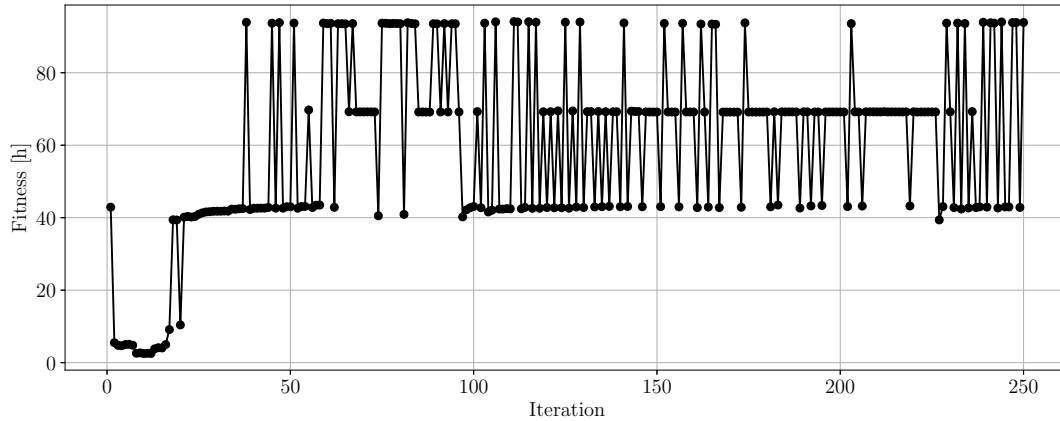


Figure 5.33: Fitness convergence for the refined clean configuration, with payload optimization.

The scatter plot of which constraints do not meet the tolerance threshold is shown in Figure 5.34. The values of the constraints are found in Figure G.10 in Appendix G. Iterations 116, 124, and 195 do not violate any constraints, making them the best results. Iteration 195 has the highest fitness of 43.37 h and is considered the solution for this optimization. The geometry is shown in Figure 5.35, where the original result is shown in blue and the final solution with the refined constraints is shown in grey.

The design variables for the original and the refined clean configuration optimization with payload are shown in Figure 5.36. The design variables of the original vehicle are shown in blue as iteration 96 and those of the refined vehicle are shown in red as iteration 195. The evolution of the design variables for the full optimization of the refined constraints is in Appendix G as Figure G.11. Further vehicle properties are summarized in Table 5.13, comparing the original vehicle to the refined, final solution.

The final vehicle has a slightly smaller wing planform, the same solar cell area, larger empennage surfaces with shorter moment arms, and the same propulsion system. The decrease in fitness is attributed to the slight increase in power draw in comparison to the original result, which renders the final vehicle unable to recharge its batteries in month 3. As with the original vehicle, the final solution is in a local optimum with respect to the solar array. The vehicle does not maximize the area of solar cells fully on the wing. Changing the array from $P(0)=10$ and $P(1)=8$ to $P(0)=P(1)=11$ increases the fitness from 43.37 h to 44.13 h.

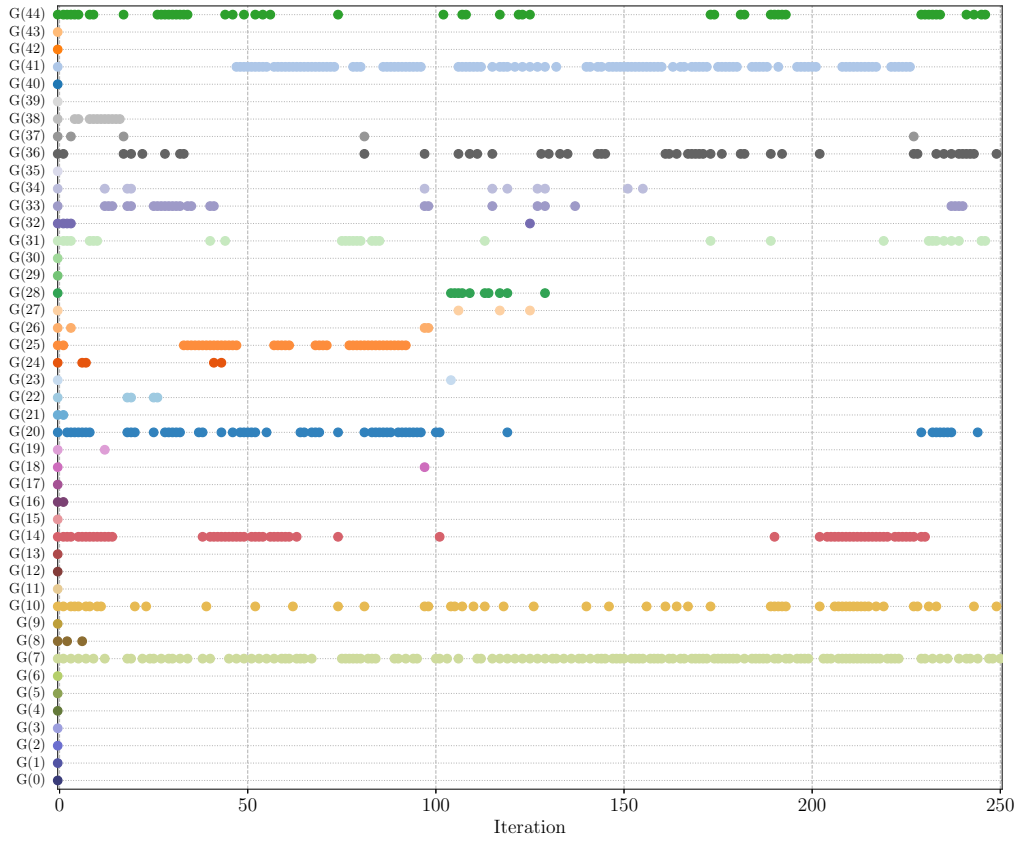


Figure 5.34: Scatter of constraints not meeting the tolerance threshold for the refined clean configuration with payload optimization.

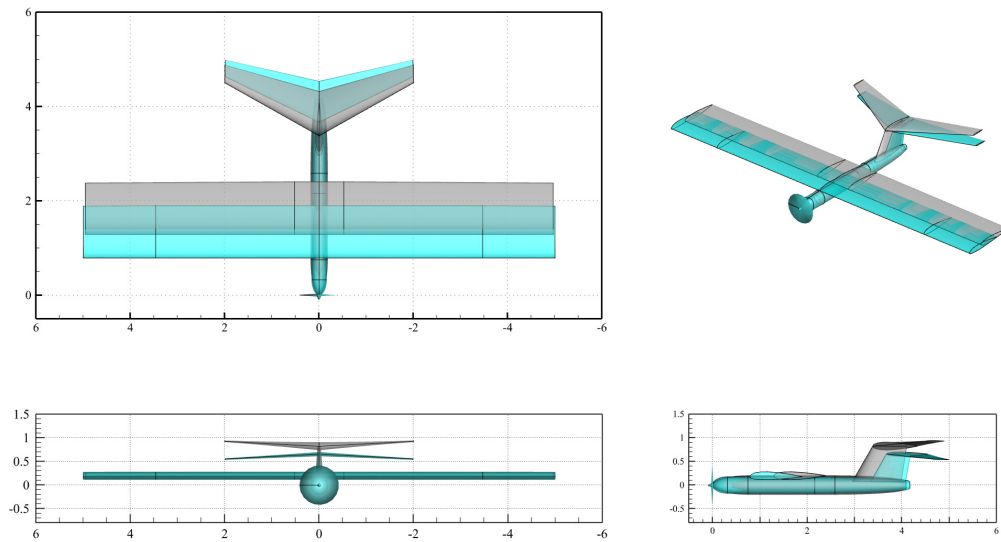


Figure 5.35: Converged solutions for the clean configuration with payload optimization. The initial result is in blue, the final solution is in grey. Axes are in feet.

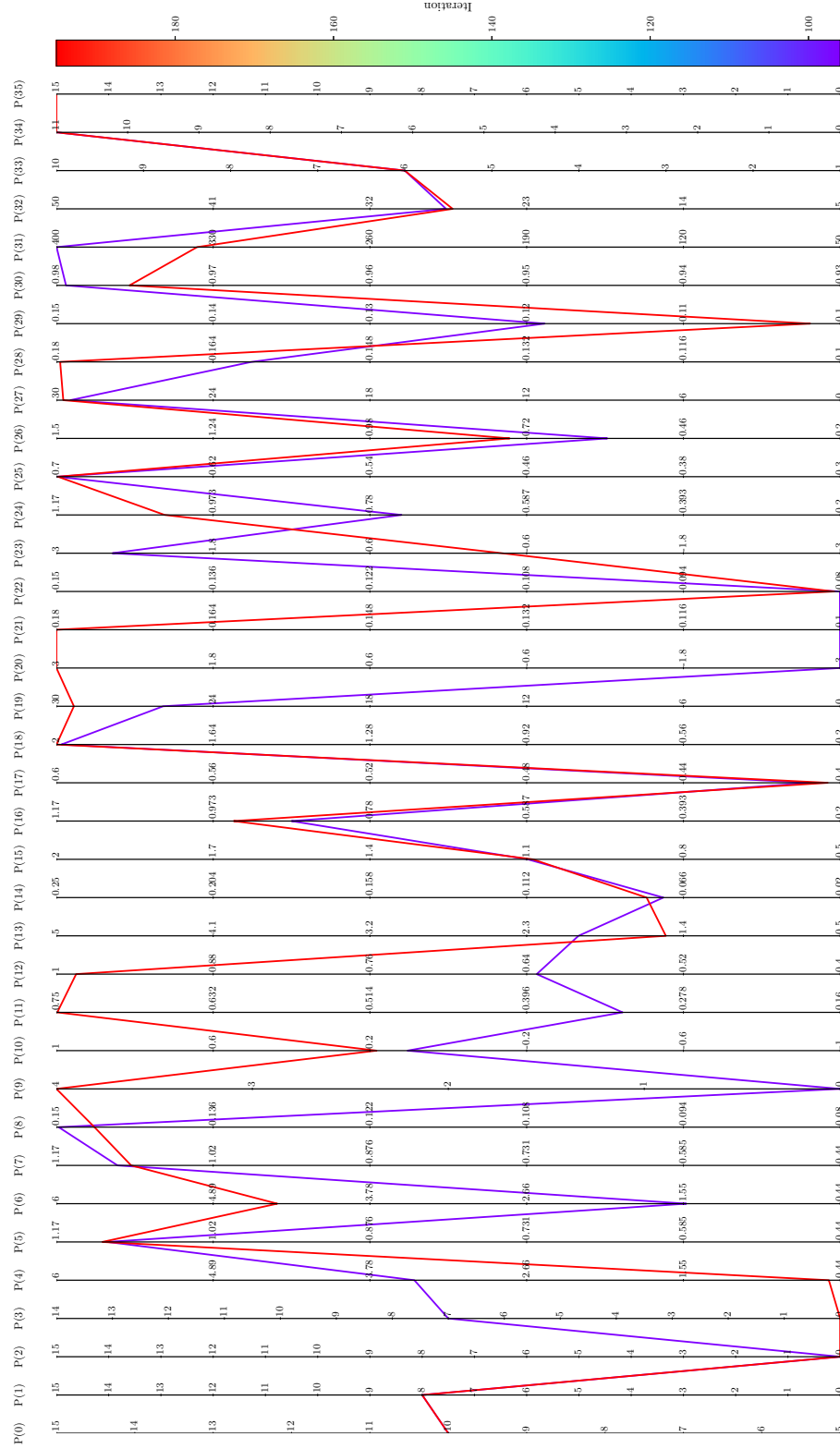


Figure 5.36: Comparison of design variables for the clean configuration with payload optimizations.

Table 5.13: Clean configuration with payload vehicle properties.

Parameter	Unit	Original	Refined
n_{PV}	[-]	36	36
S	[ft ²]	11.16	11.02
$\bar{c}$	[ft]	1.12	1.11
b	[ft]	9.98	9.91
λ	[-]	0.99	0.97
$\mathcal{R}$	[-]	8.93	8.91
w_{fuse}	[ft]	0.32	0.37
h_{fuse}	[ft]	0.36	0.40
$\mathcal{R}_{HT}$	[-]	6.45	6.01
S_{HT}	[ft ²]	2.45	2.66
$\mathcal{R}_{VT}$	[-]	1.86	1.70
S_{VT}	[ft ²]	0.74	1.31
x_{CG}	[% MAC]	34	29
W_{batt}	[lb _f]	7.2	7.2
W_{struc}	[lb _f]	0.8	0.7
W_{prop}	[lb _f]	0.5	0.5
W_{pl}	[lb _f]	0.8	0.8
W_{oe}	[lb _f]	1.2	1.2
W_0	[lb _f]	9.3	9.2
SM	[% MAC]	28	25
n_z	[-]	1.00	1.00
C_{D0}	[-]	0.0189	0.0195
K	[-]	0.035	0.035
$C_{L_{max}}$	[-]	1.73	1.73
V	[$\frac{\text{ft}}{\text{s}}$]	27.6	27.3
V_{stall}	[$\frac{\text{ft}}{\text{s}}$]	20.7	20.8
$V_{P_{min}}$	[$\frac{\text{ft}}{\text{s}}$]	24.2	24.1
$C_{l,\beta}$	[rad ⁻¹]	-0.057	-0.070
$C_{n,\beta}$	[rad ⁻¹]	0.029	0.034
T_{max}	[lb _f]	6.2	6.2
E	[Wh]	559.4	559.4
$P_p + P_{pl}$	[W]	27.2	27.5
fitness	[h]	69.32	43.37

5.5.3 Nacelles without Payload

This configuration did not converge and reached the maximum 250 iterations. The fitness and the constraints must be considered simultaneously to determine the best vehicle. The fitness is shown in Figure 5.37.

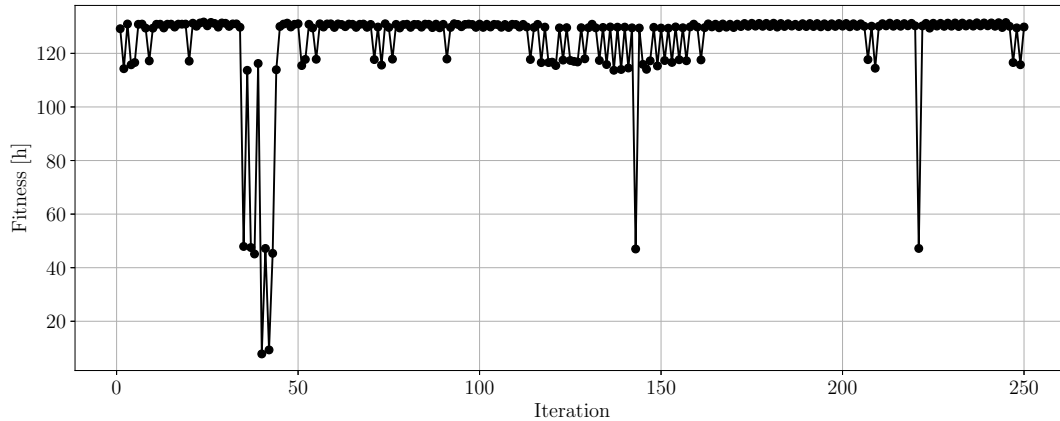


Figure 5.37: Fitness convergence for the refined nacelle configuration without payload optimization.

The constraints that do not meet the tolerance threshold are shown in the scatter plot in Figure 5.38. The values of the constraints are found in Figure G.12 in Appendix G. Despite not having the minimum number of solar panels, iterations 200, 214, and 222 are the most promising configurations. With very minor adjustments, iteration 222 is considered the solution to this optimization. The chord lengths are slightly adjusted to be a rectangular wing to $P(5)=P(7)=P(8)=1.12$ ft, the panels are adjusted so that $P(0)=10$ and $P(1)=8$. These minor modifications result in a fitness of 130.75 h. The final geometry is shown in Figure 5.39 in grey, with the original results shown in blue.

The comparison of design variables of the solutions of the original and refined optimizations are shown in the parallel lines plot in Figure 5.40. The original result is shown in red as iteration 248 and the final, refined result is shown in blue as iteration 222. The full evolution of the design variables for the refined optimization is shown in Figure G.13 in Appendix G. The major differences between both vehicles are the positions of the nacelles on the wing and the length of the fuselage. The upper static margin limit brings the empennage much closer to the wing to reduce the length of the moment arm. Additional vehicle characteristics are summarized in Table 5.14, comparing the original vehicle to the final, refined vehicle. Despite the different motor and propeller combinations, both vehicles have very similar fitness and performance levels. The refined vehicle has a higher weight but lower drag coefficients, resulting in an even trade-off between the two at an airspeed of $14.4 \frac{\text{ft}}{\text{s}}$.

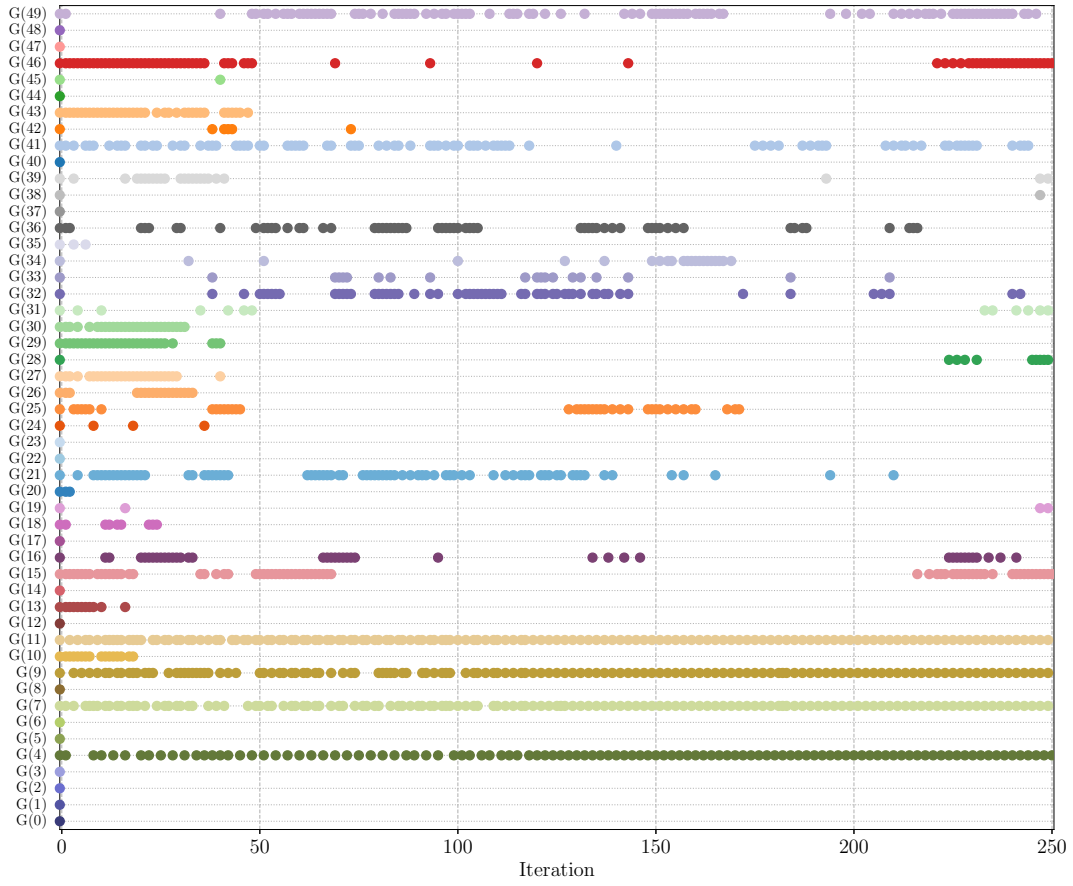


Figure 5.38: Scatter of constraints not meeting the tolerance threshold for the refined nacelle configuration, without payload optimization.

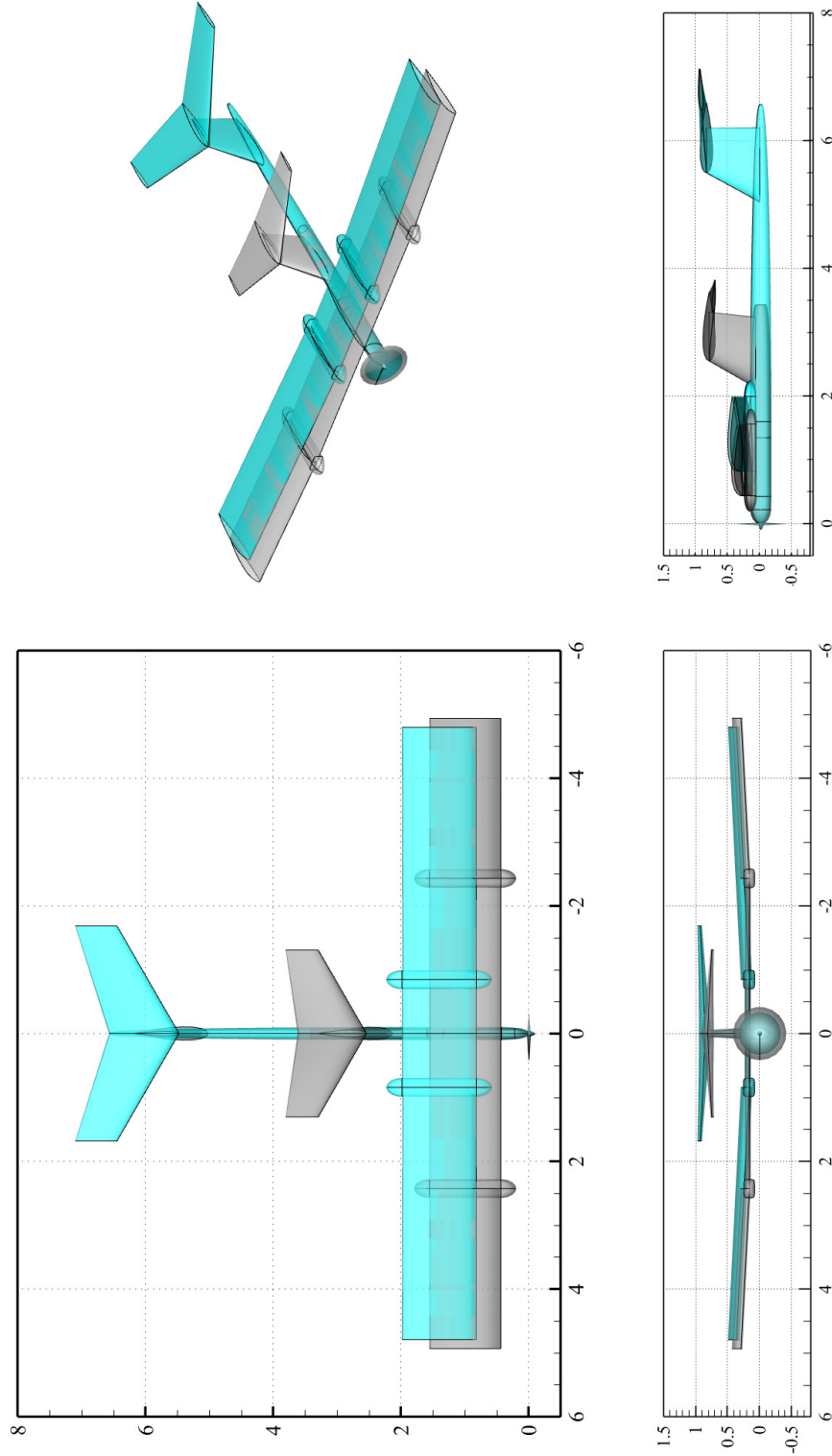


Figure 5.39: Converged solutions for the nacelle configuration without payload optimization. The initial result is in blue, the final solution is in grey. Axes are in feet.

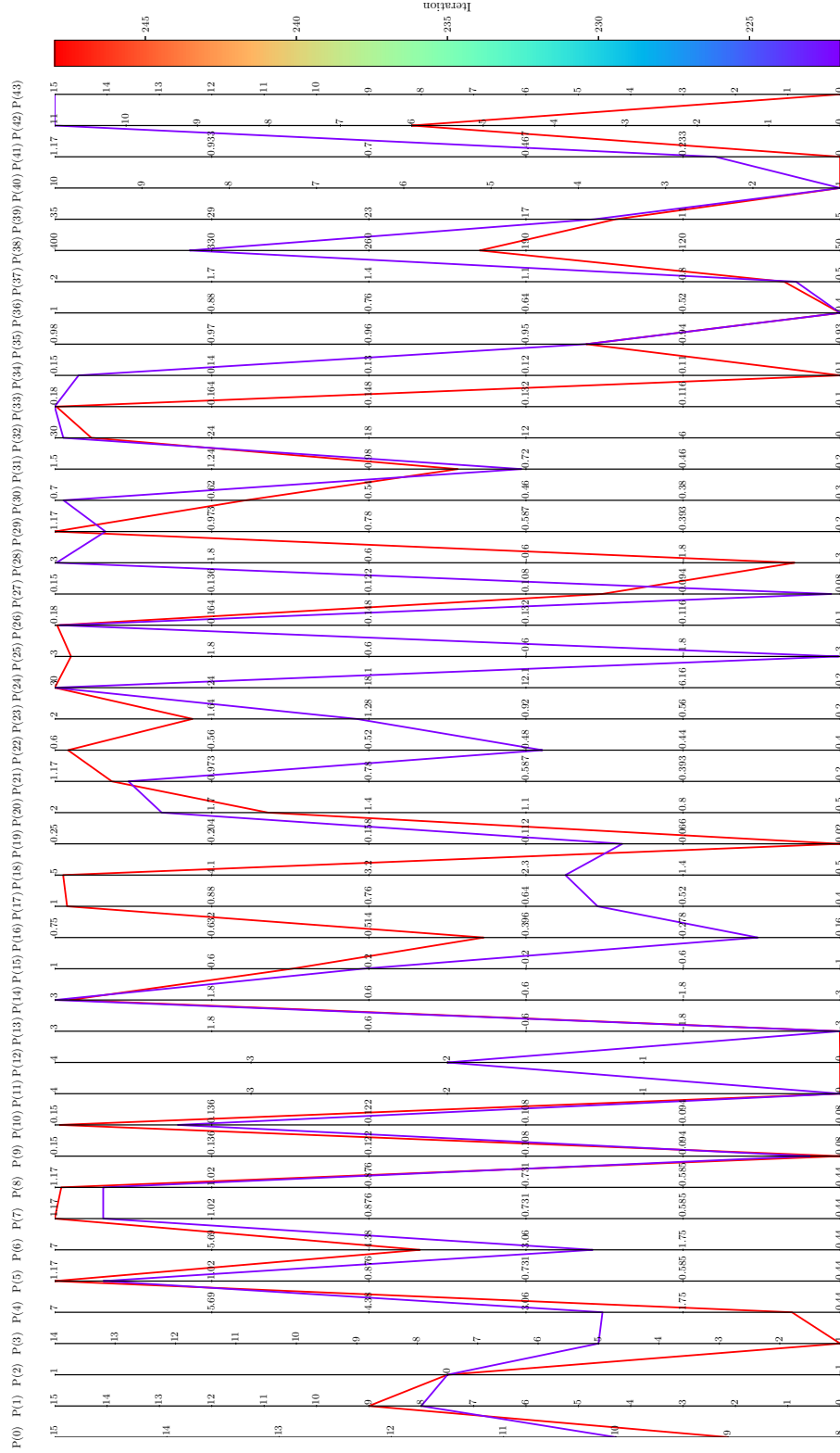


Figure 5.40: Comparison of design variables for the nacelle configuration without payload optimizations.

Table 5.14: Nacelle configuration without payload vehicle properties.

Parameter	Unit	Original	Refined
n_{PV}	[-]	36	36
S	[ft ²]	11.16	11.07
$\bar{c}$	[ft]	1.16	1.12
b	[ft]	9.58	9.86
λ	[-]	1.00	1.00
$\mathcal{R}$	[-]	8.23	8.79
w_{nac}	[ft]	0.30	0.30
h_{nac}	[ft]	0.18	0.17
w_{fuse}	[ft]	0.21	0.18
h_{fuse}	[ft]	0.33	0.33
$\mathcal{R}_{HT}$	[-]	3.85	3.30
S_{HT}	[ft ²]	2.95	2.08
$\mathcal{R}_{VT}$	[-]	1.78	1.55
S_{VT}	[ft ²]	2.95	1.36
x_{CG}	[% MAC]	19	22
W_{batt}	[lb _f]	1.2	1.2
W_{struc}	[lb _f]	0.3	0.3
W_{prop}	[lb _f]	0.2	0.5
W_{pl}	[lb _f]	0.5	0.5
OEW	[lb _f]	0.5	0.7
W_0	[lb _f]	2.2	2.5
SM	[% MAC]	73	25
n_z	[-]	1.00	1.00
C_{D_0}	[-]	0.0256	0.0235
K	[-]	0.039	0.038
$C_{L_{max}}$	[-]	1.65	1.63
V	$[\frac{\text{ft}}{\text{s}}]$	13.6	14.4
V_{stall}	$[\frac{\text{ft}}{\text{s}}]$	10.4	11.0
$V_{P_{min}}$	$[\frac{\text{ft}}{\text{s}}]$	11.2	12.1
$C_{l,\beta}$	[rad ⁻¹]	-0.103	-0.088
$C_{n,\beta}$	[rad ⁻¹]	0.107	0.033
T_{max}	[lb _f]	1.8	6.5
E	[Wh]	93.2	93.2
$P_p + P_{pl}$	[W]	3.5	3.5
fitness	[h]	130.72	130.75

5.5.4 Nacelles with Payload

As with the original definition of the problem, this configuration did not converge. The full 250 iterations were performed and the fitness must be considered in combination with the constraints to determine the best vehicle. The fitness is plotted in Figure 5.41. The limited performance remains attributed to the loss of effective lifting surface through the inclusion of nacelles. There is no geometric combination within these results that serves as a viable option for LALE solar-powered flight.

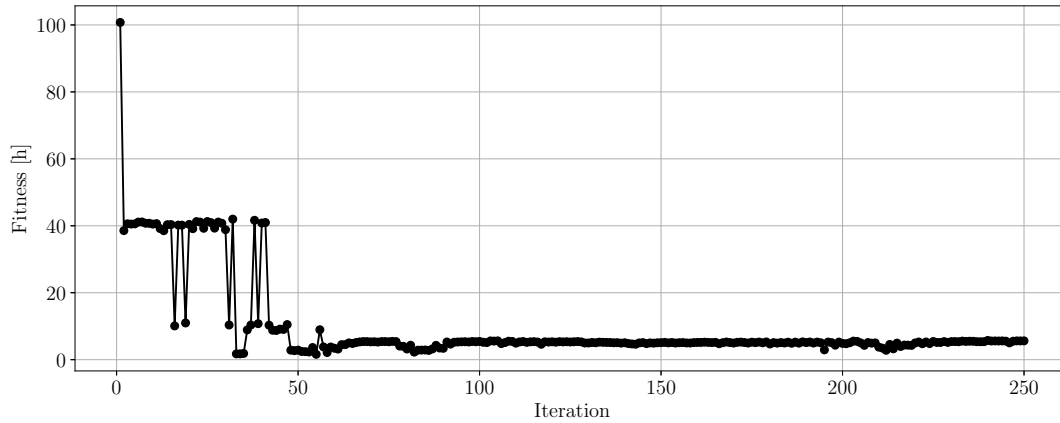


Figure 5.41: Fitness convergence for the refined nacelle configuration, with payload optimization.

The constraints not meeting the tolerance threshold are shown in the scatter plot in Figure 5.42. The values of the constraints are found in Figure G.14 in Appendix G. Iterations 190 and 192 have the fewest and most promising violated constraints. Iteration 190 is taken to be the solution to this optimization. The roll stability violated in G(43) is a small enough violation that it can be neglected. The solar array is changed from $P(0)=P(1)=8$ to $P(0)=P(1)=9$, resulting in a slight decrease in fitness to 5.07 h. The final geometry is shown in Figure 5.43, where the original solution is shown in blue, and the final solution from the refined geometry is shown in grey.

The comparison of design variables of the solutions of the original and refined optimizations are shown in the parallel lines plot in Figure 5.44. The original result is shown in red as iteration 250 and the final, refined result is shown in blue as iteration 190. The full evolution of the design variables for the refined optimization is shown in Figure G.15 in Appendix G. Additional vehicle characteristics are compared in Table 5.15. The most significant difference between the original and the refined vehicles was the fuselage length. To comply with the upper static margin constraint, the refined vehicle is significantly shorter. The nacelle configuration remains infeasible for LALE solar-powered flight with higher power requirements with a maximum span of 10 ft.

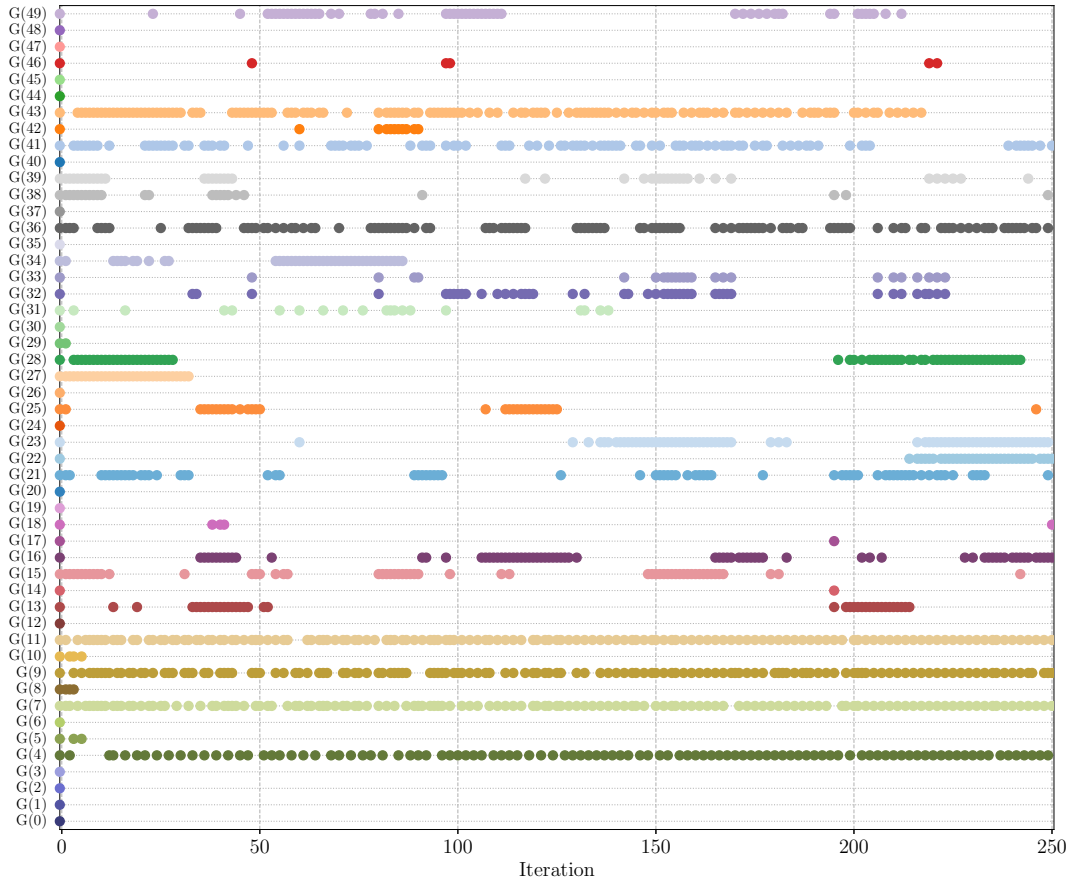


Figure 5.42: Scatter of constraints not meeting the tolerance threshold for the refined nacelle configuration, with payload optimization.

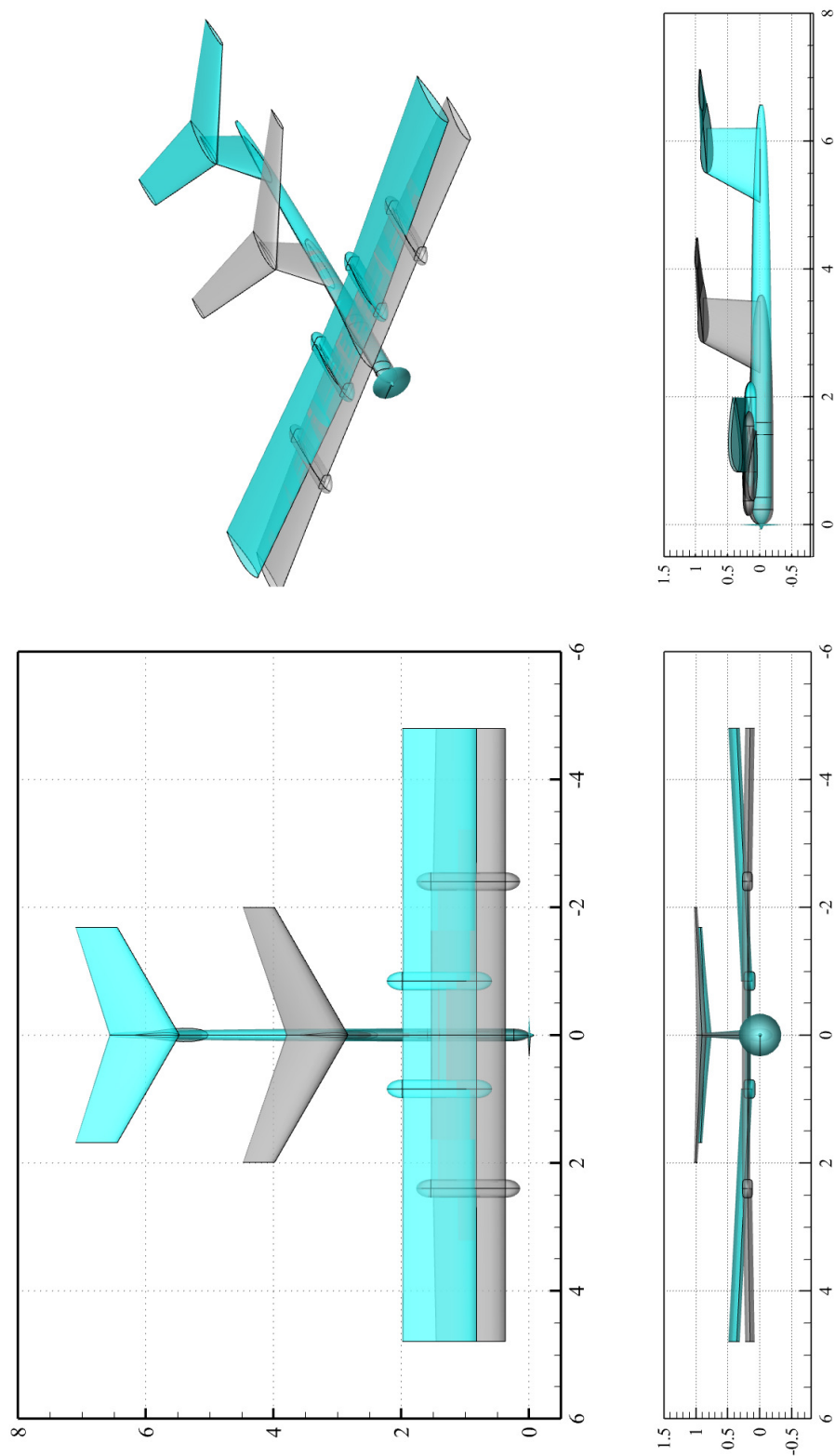


Figure 5.43: Converged solutions for the nacelle configuration with payload optimization. The initial result is in blue, the final solution is in grey. Axes are in feet.

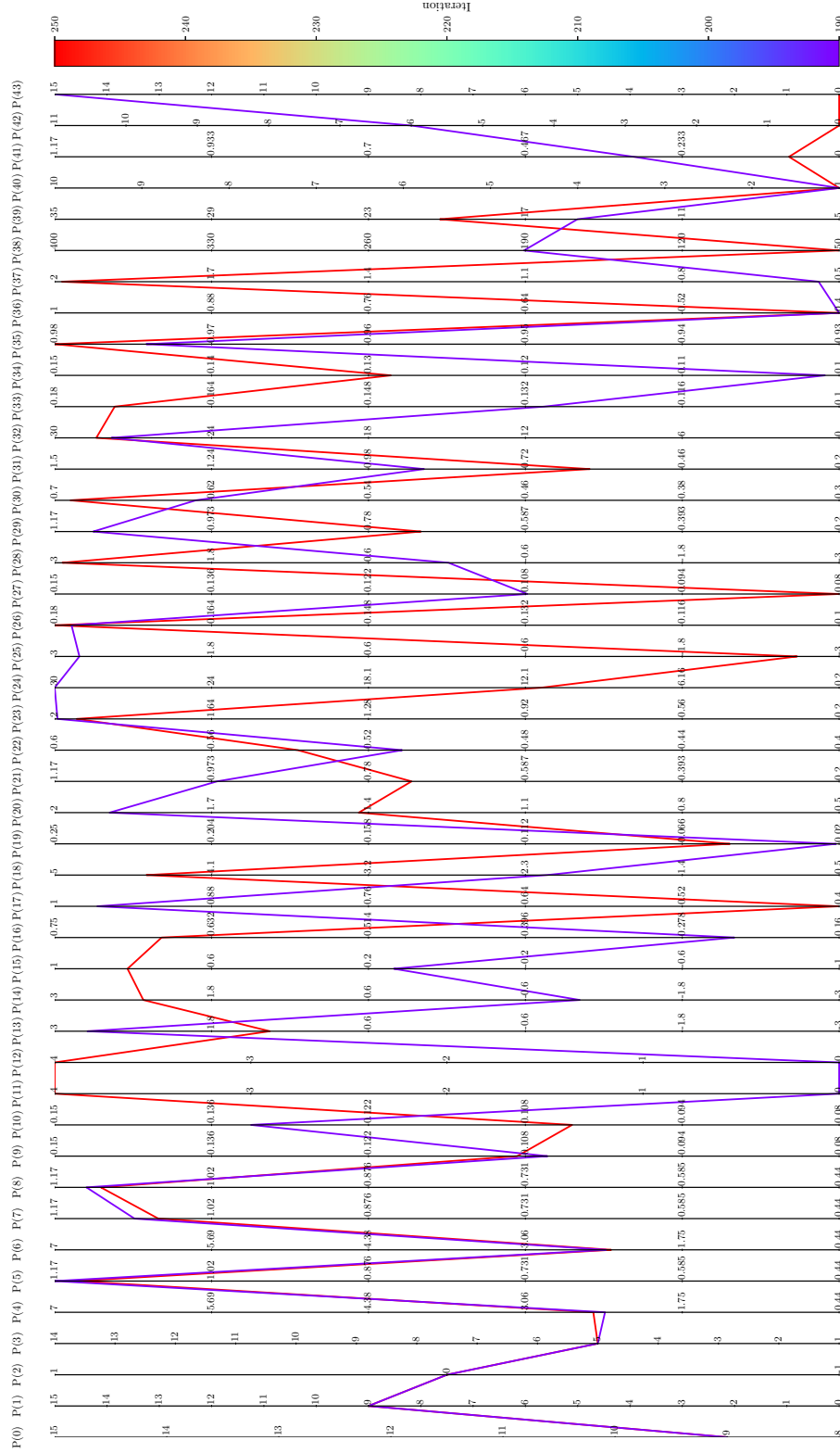


Figure 5.44: Comparison of design variables for the nacelle configuration with payload optimizations.

Table 5.15: Nacelle configuration with payload vehicle properties.

Parameter	Unit	Original	Refined
n_{PV}	[-]	36	36
S	[ft ²]	10.88	10.92
$\bar{c}$	[ft]	1.12	1.14
b	[ft]	9.68	9.57
λ	[-]	0.91	0.92
$\mathcal{R}$	[-]	8.62	8.39
w_{nac}	[ft]	0.30	0.30
h_{nac}	[ft]	0.59	0.16
w_{fuse}	[ft]	0.17	0.21
h_{fuse}	[ft]	0.24	0.39
$\mathcal{R}_{HT}$	[-]	6.98	5.46
S_{HT}	[ft ²]	2.18	2.91
$\mathcal{R}_{VT}$	[-]	2.03	1.95
S_{VT}	[ft ²]	0.74	1.62
x_{CG}	[% MAC]	17	44
W_{batt}	[lb _f]	1.2	1.2
W_{struc}	[lb _f]	0.2	0.2
W_{prop}	[lb _f]	0.2	0.5
W_{pl}	[lb _f]	0.8	0.8
W_{oe}	[lb _f]	0.4	0.7
W_0	[lb _f]	2.4	2.7
SM	[% MAC]	56	18
n_z	[-]	1.03	1.00
C_{D_0}	[-]	0.0255	0.0246
K	[-]	0.036	0.054
$C_{L_{max}}$	[-]	1.60	1.67
V	$[\frac{\text{ft}}{\text{s}}]$	20.3	15.0
V_{stall}	$[\frac{\text{ft}}{\text{s}}]$	11.1	11.4
$V_{P_{min}}$	$[\frac{\text{ft}}{\text{s}}]$	11.6	13.7
$C_{l,\beta}$	[rad ⁻¹]	-0.039	-0.048
$C_{n,\beta}$	[rad ⁻¹]	0.037	0.062
T_{max}	[lb _f]	0.7	2.9
E	[Wh]	93.2	93.2
$P_p + P_{pl}$	[W]	12.8	12.0
fitness	[h]	4.75	5.07

5.6 Final Optimization Summary

The inclusion of an upper constraint on the static margin has a noticeable impact on all four configurations that were optimized. The resulting vehicles have shorter fuselage lengths which decrease the moment arms from the centre of gravity to the aerodynamic centres of the stabilizers. The final results will have less penalizing effects of trim drag and servo-motor power draw by requiring lower inputs to manoeuvre and stabilize the vehicle.

Results of Chapter 4 showed that higher spans were capable of longer flight times due to their increased aerodynamic efficiency and therefore lower power consumption. Each UAV configuration maximized the span of the design up to the span constraint. Further results of Chapter 4 showed that lower aspect ratios are better suited for heavier payloads and higher power demands. In this optimization analysis, the chord length of all surfaces is a manufacturing constraint. The optimizer is unable to produce vehicles with longer chord lengths at the same span to further reduce the aspect ratio for the higher payload power demands, nor does it decrease the chord to increase aspect ratio for the lower payload power demands. As a result, the enforced constraints and limitations in the realistic design space in this optimization do not support the same conclusion that a decrease in aspect ratio benefits the performance of higher payloads in a solar UAV. Further optimization at longer chord limits and a greater span would be required to confirm these findings from Chapter 4, which may result in an unrealistic geometry when considering the production.

As with the initial results, the final results converge at optimum airspeeds higher than their respective minimum power airspeeds. This remains an artifact of the maximum lift coefficient constraint, which is to ensure satisfactory handling and ability to withstand wind gusts without stalling. In calm conditions, the vehicles could be flown closer to their minimum power airspeeds to further extend their flight times and increase charge margins. The cruising airspeed greater than the minimum power airspeed also occurs in the additional optimization at a higher span constraint in Appendix H. Although the minimum power airspeed is a function of the aerodynamic characteristics of the vehicle, the actual power draw from the propulsion system is based on the total efficiency of that system. Therefore, it remains important to match the propeller and motor performance characteristics to the operating speed or flight envelope of the vehicle to properly minimize the power draw of the system. The performance should also be considered for higher airspeeds to allow the vehicle to have a larger flight envelope where it is still possible for the batteries to recharge. This creates robustness to winds, where the vehicle may be required to sustain a higher power draw to overcome headwinds when maintaining a loitering pattern. The higher charge margin a vehicle has, the greater power draw it can sustain during daylight hours. Overnight, the vehicle will be limited by the power draw required to remain above a 30% SOC in order to remain airborne until sunrise.

A main consideration of this project is designing a solar UAV capable of LALE operations for practical applications. The ability to achieve 24+ hours of flight with payload is necessary and must be considered in the design process. Recall that the vehicle is designed with a 10 ft wingspan constraint to facilitate hand launching for takeoff and

minimize ground handling difficulties. With no additional payload, the clean and the nacelle configurations have comparable fitness values, 131.45 h and 130.75 h, respectively. Both are capable of recharging the battery of the vehicle in all four simulated months, allowing for multi-day flights throughout the year with greater energy margins in the months with higher solar irradiance and longer daylight hours. The clean configuration has slightly higher energy margins in each month, indicated by the greater fitness level. With payload and limited to 10 ft in span, the nacelle configuration is not viable for LALE solar-powered flight. It does not allow enough lifting area or aerodynamic efficiency to support the battery size required to accommodate the greater power draw from the payload. The viability for the nacelle configuration for LALE solar-powered flight is highly dependent on the span of the vehicle. The results of the optimization with a longer span constraint in Appendix H has different results, but the same final conclusion: the clean configuration is the most suitable for LALE solar-powered flight. This conclusion is consistent with the successfully designed and tested LALE solar-powered UAVs.

5.7 Proposed Geometry

This section proposes the final geometry and propulsion system for a LALE solar-powered UAV. The results for the clean configuration optimizations have some common properties, which will be propagated into the final design. The vehicle should have the internal volume allocation to accommodate the most required batteries with the greatest expected payload for the most flexibility in its application. The final vehicle is a combination of the solutions from the clean configuration optimization results with and without payload. The individual design variables for the proposed geometry are summarized at the end of Appendix G in Table G.1. The proposed geometry is shown in Figure 5.45.

The proposed solar array does not fully cover the span in both rows of solar cells. It is acknowledged that the energy margins increase by including as many solar cells as possible. As the vehicle is modelled, those increases in fitness are small. The advantage to not filling the entire wing is the simplification of the integration of the solar cells and the ailerons into the wing, where there is empty area on the trailing edge at the tip of the wing, where the ailerons do not need to conform to the dimensions of a given number of solar cells, nor do the electrical connections of the cells need to accommodate the moving surface.

The final horizontal stabilizer raises concerns of whether the sweep and taper is truly necessary. A test optimization was conducted, removing the horizontal tail trailing edge sweep constraint and increasing the taper variable range to $[0.4, 1]$. The results of that test converged with a swept and tapered horizontal tail similar to the proposed geometry and within the original variable bounds. This is likely driven by the centre of gravity and stability constraints. The root leading edge of the horizontal stabilizer is coincident with the tip leading edge of the vertical stabilizer and must adjust the position of its aerodynamic centre by introducing sweep and taper. The tapered and swept geometry for the horizontal stabilizer is preserved in the final geometry. As the fuselage is currently defined, any increase in fuselage length results in an increase in

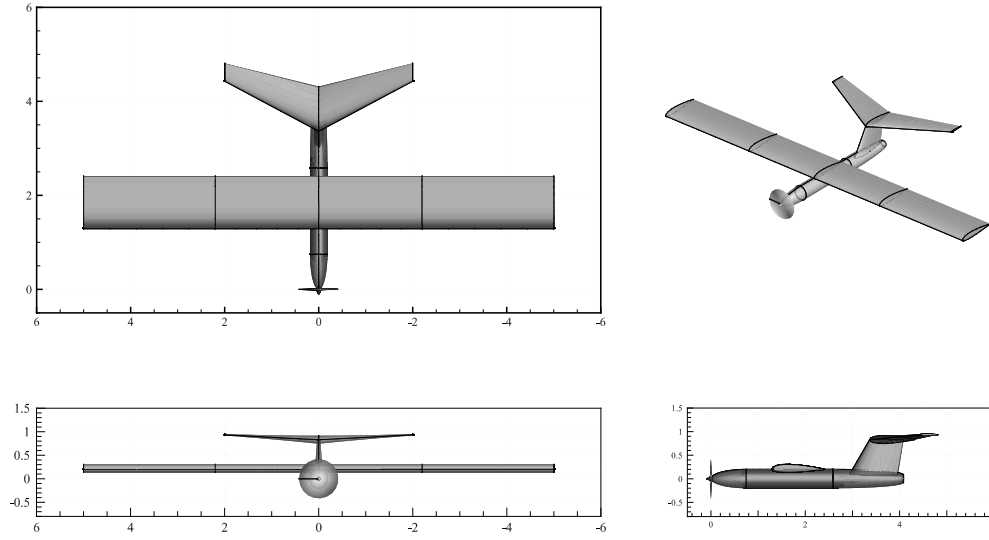


Figure 5.45: Proposed geometry for a LALE solar UAV. Axes are in feet.

wetted surface area and therefore skin friction drag. If an additional segment were added to the fuselage for the purposes of further streamlining the body and decreasing the cross-sectional area of the tail boom, the fuselage may increase in length, moving the empennage further aft from the wing. This increase in moment arm would allow the stabilizing surface areas to decrease, which may result in a lower drag profile than the optimized geometry. With the current fuselage definition, the optimized geometry shows that this increase in drag from a longer fuselage and decrease in drag from smaller stabilizing surfaces is not better than the shorter fuselage and larger stabilizing surfaces.

Table 5.16 summarizes some of the physical and aerodynamic characteristics of the UAV. The weights and performance metrics for the different payload cases are summarized in Table 5.17. The weights are sized to accommodate the greatest payload and battery capacity, so the structural weight is higher than if it was only sized for the zero payload case. As a result, the performance at the zero payload case is reduced as the vehicle is heavier and more thrust is required to maintain steady level flight. It is assumed that the centre of gravity and stability characteristics are preserved with different payloads and sizes of battery packs, as the internal components can be adjusted along the length of the fuselage to maintain balance. The proposed vehicle is not able to recharge its batteries in month 12 with any payload combination. This is a decrease in performance compared to the vehicles optimized for no payload. The ability to accommodate heavier or more power-demanding payloads and greater battery sizes in one design creates larger flexibility for the application of the UAV rather than a slimmer design capable of LALE flight in December but with no margin to execute other tasks.

The lift, drag, and pitching moment coefficients, C_L , C_D , and C_m , respectively, are shown in Figure 5.46. The aerodynamic methods used do not capture the decrease in the lift curve slope as the angle of attack approaches stall, but these analyses do not

Table 5.16: Properties of the proposed solar-powered UAV.

Parameter	Unit	Value
n_{PV}	[-]	40
S	[ft ²]	11.20
$\bar{c}$	[ft]	1.12
b	[ft]	10.00
λ	[-]	1.00
$\mathcal{R}$	[-]	8.93
w_{fuse}	[ft]	0.37
h_{fuse}	[ft]	0.40
$\mathcal{R}_{HT}$	[-]	6.02
S_{HT}	[ft ²]	2.66
$\mathcal{R}_{VT}$	[-]	1.71
S_{VT}	[ft ²]	1.31
x_{CG}	[% MAC]	29
W_{struc}	[lb _f]	0.8
W_{prop}	[lb _f]	0.5
SM	[% MAC]	24
n_z	[-]	1.00
C_{D0}	[-]	0.0198
K	[-]	0.037
C_{Lmax}	[-]	1.73
$C_{l,\beta}$	[rad ⁻¹]	-0.069
$C_{n,\beta}$	[rad ⁻¹]	0.035
T_{max}	[lb _f]	6.4

Table 5.17: Weights and performance of the proposed solar-powered UAV.

Parameter	Unit	Value		
Payload Option	[-]	0	1	2
W_{batt}	[lb _f]	1.2	3.6	7.2
W_{pl}	[lb _f]	0.6	0.8	0.9
OEW	[lb _f]	1.2	1.2	1.2
W_0	[lb _f]	3.0	5.6	9.3
V	[$\frac{ft}{s}$]	15.2	21.0	27.0
V_{stall}	[$\frac{ft}{s}$]	11.7	16.1	20.7
$V_{P_{min}}$	[$\frac{ft}{s}$]	13.7	18.9	24.4
E	[Wh]	93.2	279.7	559.4
$P_p + P_{pl}$	[W]	4.3	13.1	28.7
month 6 t_{cm}	[h]	11.84	9.53	6.31
month 3 t_{cm}	[h]	8.96	6.57	0.00
month 9 t_{cm}	[h]	8.79	6.28	0.00
month 12 t_{cm}	[h]	0.00	0.00	0.00
fitness	[h]	115.78	108.22	42.99

consider flight angles near stall, negating the need for more refined models for stall angles. The lift to drag ratios as a function of airspeed are shown in Figure 5.47 for the proposed geometry configured for payload option 1. The maximum aerodynamic efficiency, C_L/C_D , occurs at the airspeed for maximum range of a propeller propulsion system, which is the minimum drag airspeed. The minimum power airspeed corresponds to the maximum $C_L^{3/2}/C_D$, which is the speed for maximum endurance of a propeller propulsion system.

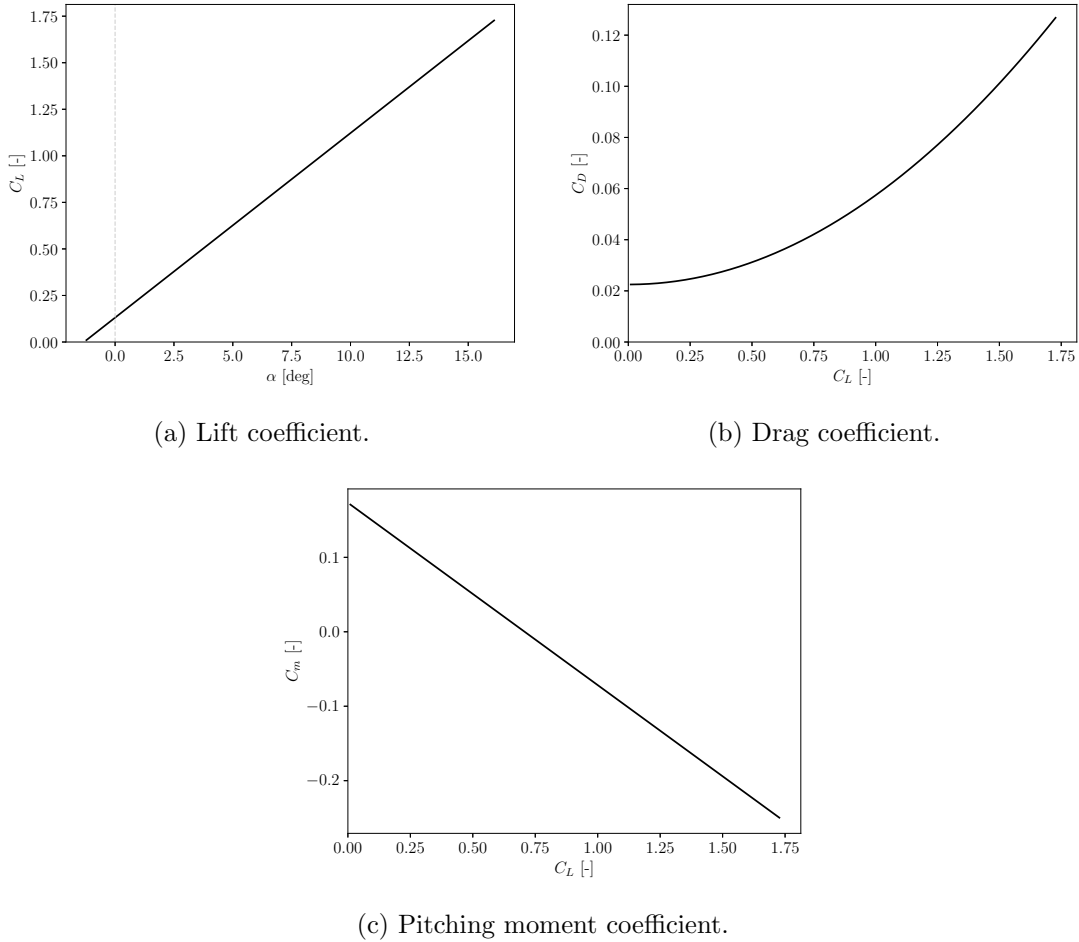


Figure 5.46: Aerodynamic coefficients of the proposed geometry, $Re_c = 142\,000$.

The reported fitness value is based on several efficiency and modelling assumptions including assumed efficiencies and linear rates of charging and discharging of the batteries. There are further internal resistances and losses that cannot be included in the theoretical modelling of the vehicle. Consequently, the reported fitness values and charge margins are upper limits of how the real vehicle might perform. The true performance and energy margins of the real vehicle can only be confirmed through flight-testing.

Fabrication and testing are the next steps for this geometry. The fabrication itself will be an exercise in integrating various components. The 3D-printed exterior of the

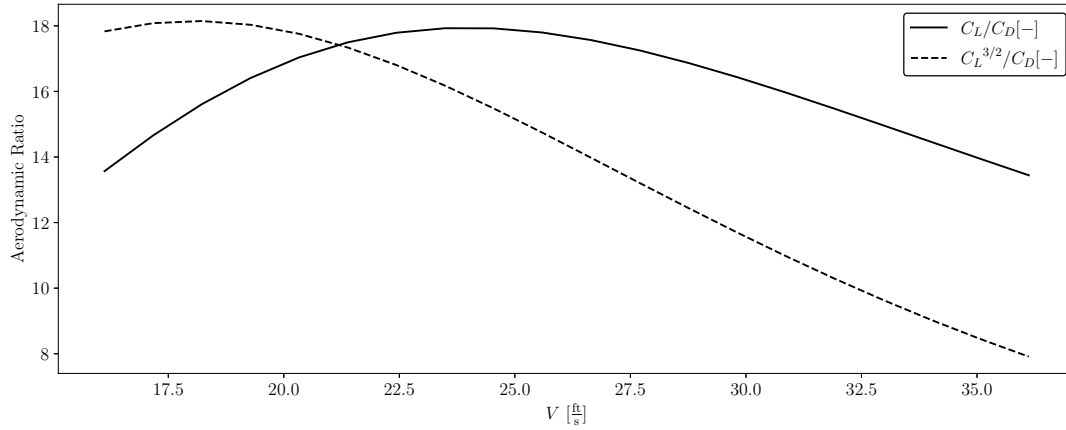


Figure 5.47: Aerodynamic ratios with airspeed for the proposed geometry configured for payload option 1.

solar-powered UAV will require carbon fibre rods through the wing, fuselage length, and possibly stabilizing surfaces for structural stiffness. Furthermore, the integration of the solar cells onto the suction surface of the wing will require cautious techniques to prevent cracking of the cells or the reduction of efficiency with any coverings over the cells. The design itself may require minor modifications to fit the aircraft with external sensors or the placement and housing of an external camera, which falls under more detailed design beyond the scope of the optimization of the solar UAV.

6 NEAT Trajectory Planning

This chapter addresses the trajectory optimization and path planning aspect of optimizing a solar UAV. Proposed strategies for optimal paths to either maximize collected solar power or minimize expended power consist of climbing to high altitudes during the day and gliding throughout the night, changing the bank angle of a loitering trajectory depending on the sun's elevation angle, *chasing* the sun, or pursuing other snaking patterns, as discussed in Chapter 2. The overarching objective of each trajectory optimization has significant effects on the resulting flight strategies for prolonging solar-powered flight. This chapter uses the NEAT algorithm as a means for trajectory optimization to determine and or confirm the best flight strategies for LALE solar-powered flight. A verification for the application of NEAT for trajectory optimization and optimal controls is presented in Appendix I.

6.1 Fitness Function

The fitness of a given neural network is assessed by its performance of a simulated flight. As with the optimization, the UAV is modelled as a point mass, governed by the dynamic equations of motion in Equation 3.36 and the kinematic equations of motion in Equation 3.41. The fitness is defined in a similar manner to the optimization objective function, with credit given to the flight time and the time spent at the maximum SOC. The fitness, J , of each network is evaluated as Equation 6.1, where t is the flight time and t_{cm} is the charge margin.

$$J = t + t_{cm} \quad (6.1)$$

The networks are evaluated for their ability to fly the point mass solar UAV, attempting to achieve 48 consecutive hours of flight, with credit given to the networks capable of maximizing the charge margin. A successful network would be expected to have a flight path that would result in similar power characteristics as the sample flight in Figure 6.1, which shows the definition of t_{cm} in both days. The flights are simulated to begin at sundown, then flown for 48 hours to ensure the network is capable of flying the vehicle overnight and recharging afterwards. From Chapter 5, the simulated vehicle is expected to achieve a 9.53 h charge margin on 21 June. When applied to the definition of the fitness function in Equation 6.1, the expected fitness for 48 h of flight is approximately 67 hours ($2 \times [24 + 9.53]$ h). A higher fitness would indicate that the NEAT algorithm has found a network and flight path with better energetic margins than the 337 ft turning radius at 2000 ft MSL from the vehicle optimization.

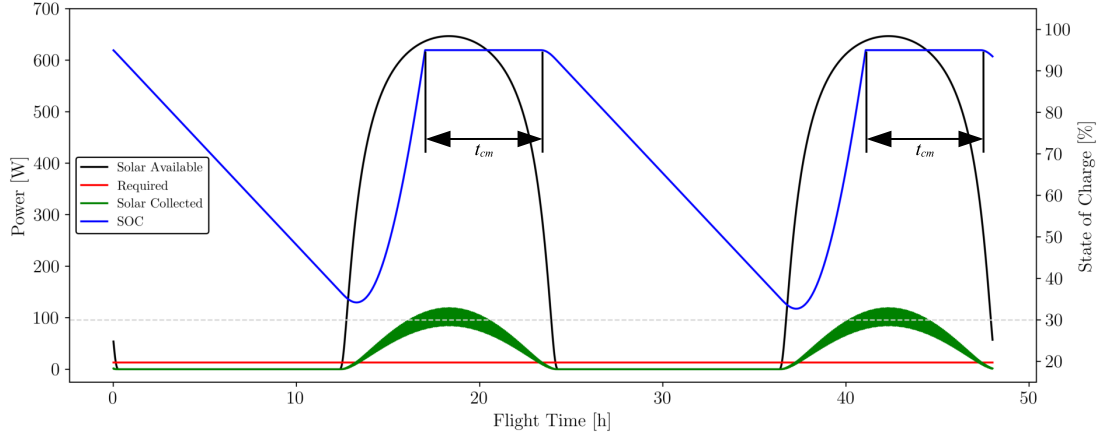


Figure 6.1: Sample flight with expected energy trends for the neural-controlled flight simulation.

6.1.1 Simulation

The network is evolved to fly the solar UAV in Table 5.16 with payload option 1. The NEAT algorithm evolves the network with the airspeed V , pitch angle γ , azimuth angle ψ , altitude h , time-of-day t , and SOC as the input states. The angle of attack α , roll angle ϕ , and thrust T are the controls. Table 6.1 presents the initial states for the start of each simulation, with bounds to the controls and other properties found in Table 6.2. Each flight simulation is flown until certain identified limits in Table 6.2 denoted by * are violated or the full 48 hours are simulated, beginning on 21 June 2025 at sundown.

Table 6.1: Initial conditions for each simulated flight.

Variable	Unit	Value
V	$[\frac{\text{ft}}{\text{s}}]$	36.2
γ	[deg]	0
ψ	[deg]	0
h	[ft]	2000
SOC	[-]	0.95

The initial SOC assumes that the UAV would have recharged its batteries during the day, maintaining the maximum SOC until sundown, which marks the start of the flight simulation, which is the same initial condition used in Chapter 5. The initial altitude remains consistent with the assumed constant altitude in Chapters 4 and 5. This chapter assumes the simplified and constant propulsion efficiency throughout the analysis, therefore it is not a function of altitude or airspeed. It is expected that the maximum fitness of a neural network would be achieved if the resulting trajectory flies at the minimum power speed, resulting in the lowest power draw from the batteries throughout the flight. The initial velocity is set higher than this speed in an attempt to encourage the algorithm to connect the velocity to the controls, but it is not directly enforced in the fitness function.

Table 6.2: State, control, and vehicle simulation limits.

Variable		Unit	Min	Max
Airspeed	V	$[\frac{\text{ft}}{\text{s}}]$	18.8*	46.9*
Pitch	γ	[deg]	-90	90
Altitude	h	[ft]	200*	2500*
SOC	SOC	[-]	0.3*	0.95
Lift Coefficient	C_L	[-]	-0.5	1.73
g -loading	n	[-]	0.5	2*
Roll Angle	ϕ	[deg]	-30	30
Thrust	T	[lbf]	0	6.4

* Violation of limit will terminate flight simulation.

There are no penalties or additional gains included in the fitness to encourage or dissuade any particular trajectory or flight behaviour. The vehicle is either successfully flown within the state's bounds in Table 6.2 or it is terminated when those bounds are violated and the fitness value is the flight time leading up to the failure. The different flight strategies are presented in the following section.

6.2 NEAT Trajectories

There are two primary flight strategies that emerged from 10 different neural networks capable of simulating flight for 48 hours based on the initial conditions in Table 6.1. The first strategy is to adopt a circular trajectory, where the turning radius is dependent on the specific network weights and connections. The second strategy is to fly with a fixed heading, effectively attempting to chase the sun.

6.2.1 Constant Radius Circular Trajectory

Constant radius circular trajectories are the most common paths output from the NEAT algorithm. In several cases, the vehicle descends from the initial 2000 ft altitude then holds a constant gentle bank angle. The network that results in a circular trajectory with the highest fitness is presented in this section. The trajectory itself is shown in Figure 6.2. Note that north is in the positive x direction and east is in the positive y direction and the x and y scales are in nmi while the altitude is denoted in ft. The starting location is denoted as (0, 0, 2000) ft in green, with the location at the end of the simulation shown in red. Note that the trajectory appears to be elliptical in Figure 6.2, which is merely an artifact of different x and y scales.

The network that results in the large circular trajectory is shown in Figure 6.3. The weighting and biases of the connections and nodes are summarized in Table 6.3.

The values at the output nodes are values normalized by the respective variable's bounds to have a value between 0 and 1. They are calculated using the weights and sigmoid function as defined in Equation Set 6.2.

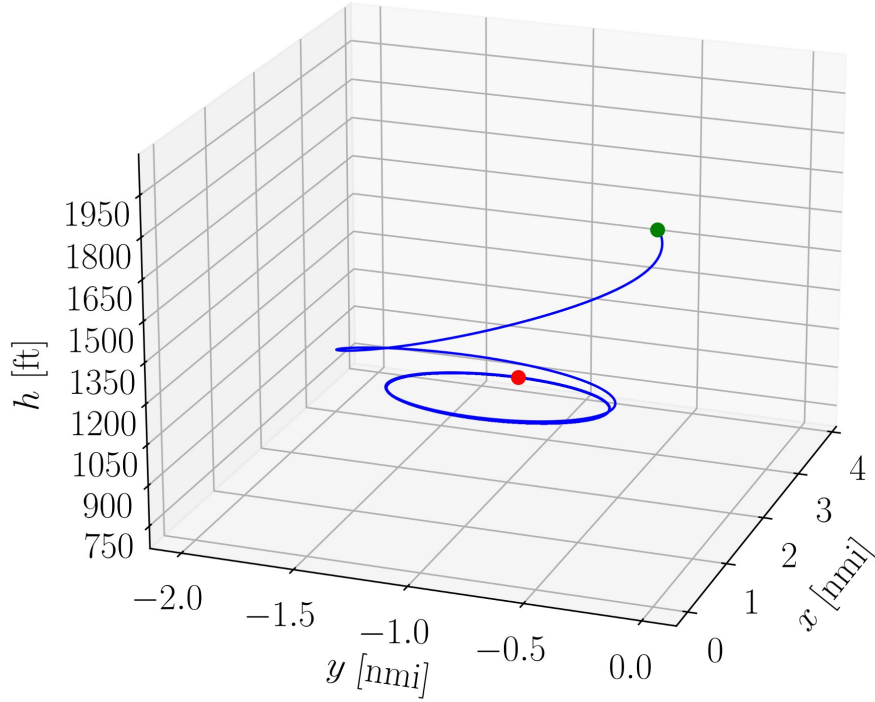


Figure 6.2: NEAT constant radius circular trajectory.

Table 6.3: Constant radius circular trajectory network weights and biases.

Node Bias		Connection Weight		
Node	Value	From	To	Value
k	0.435	γ	α'	-1.405
α'	0.125	γ	k	0.545
ϕ'	-1.987	k	ϕ'	2.222
T'	-0.071	h	T'	-1.470

$$\begin{aligned}
 k &= \sigma(\gamma w_{k,\gamma} + b_k) \\
 \alpha &= \sigma(\gamma w_{\alpha',\gamma} + b_{\alpha'}) \\
 \phi &= \sigma(k w_{\phi',k} + k + b_{\phi'}) \\
 T &= \sigma(h w_{T',h} + b_{T'}) \\
 \sigma(x) &= \frac{1}{1 + e^{-x}}
 \end{aligned} \tag{6.2}$$

The resulting normalized controls are denormalized with Equation 6.3 to restore their values to within the original variable bounds, that are used for the next time step of the simulation to update the state of the vehicle.

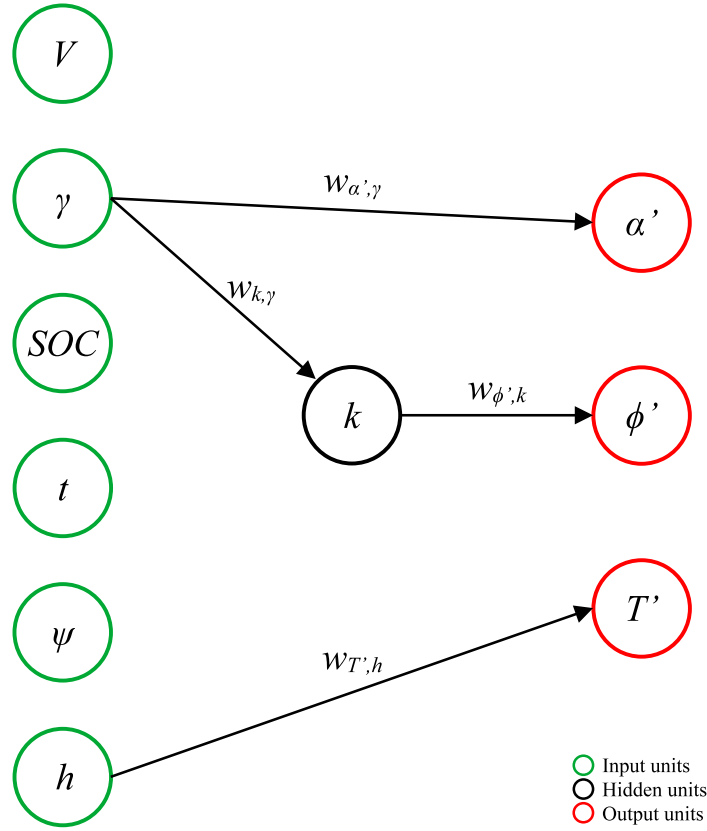


Figure 6.3: Network for a constant radius circular trajectory.

$$\begin{bmatrix} \alpha \\ \phi \\ T \end{bmatrix} = \begin{bmatrix} \alpha' & 0 & 0 \\ 0 & \phi' & 0 \\ 0 & 0 & T' \end{bmatrix} \begin{bmatrix} \alpha_{max} - \alpha_{min} \\ \phi_{max} - \phi_{min} \\ T_{max} - T_{min} \end{bmatrix} + \begin{bmatrix} \alpha_{min} \\ \phi_{min} \\ T_{min} \end{bmatrix} \quad (6.3)$$

The states and controls throughout the first hour of the flight are shown in Figure 6.4, with the entire flight shown in Figure 6.5, with the upper and lower bounds plotted in red where applicable. The green horizontal line in the airspeed indicates the minimum drag speed, with the actual airspeed plotted in black. The available, collected, and required power and SOC throughout the flight are shown in Figure 6.6.

In this network, the vehicle descends from 2000 ft to 760 ft, settling into a circular trajectory with a 3140 ft turning radius at $33.7 \frac{\text{ft}}{\text{s}}$. The resulting fitness across 48 h of flight time is 63.16 h, which is lower than the projected fitness from the vehicle optimization. The roll is not linked to the SOC and the thrust is not linked to the velocity, explaining the network's inability to settle on the minimum power airspeed or adjust the turning radius to better maximize the collected solar power, both of which would increase the network's fitness. The high turning radius results in minimal fluctuations in the solar power collection capacity, making it more realistic to increase the flight speed or have higher energy-demanding manoeuvres during the day without significantly sac-

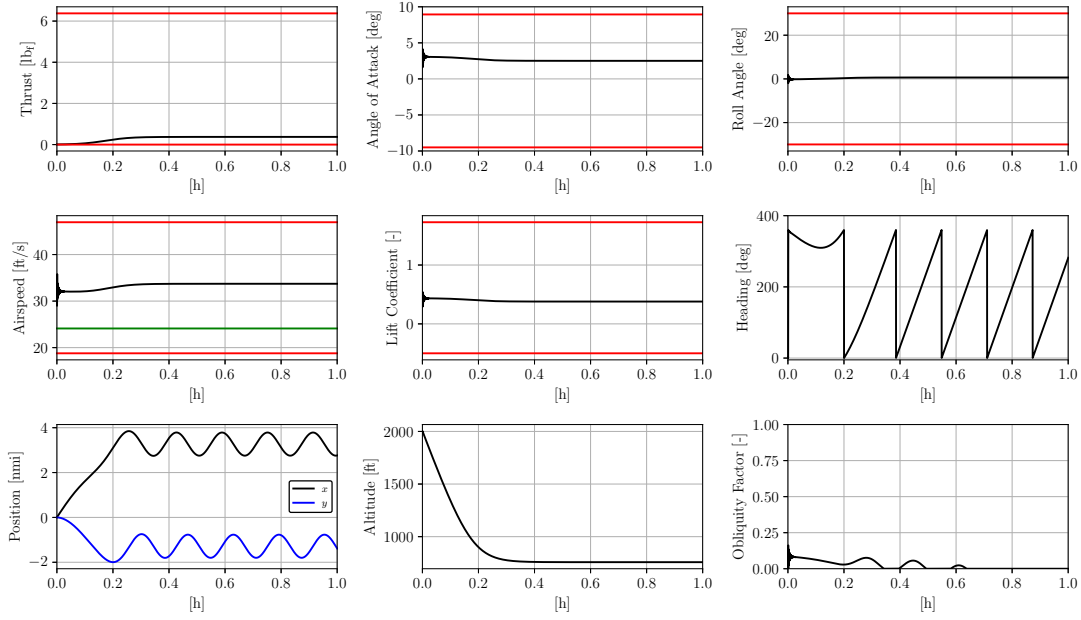


Figure 6.4: States and controls for the first hour of a constant radius circular trajectory.

rificing the charging performance of the vehicle. The minimum SOC in the first 24 hours (53.5%) is higher than the minimum SOC from the last 24 hours (50.3%). When simulated for a third day, the minimum SOC of the last two 24 hour cycles are the same, showing that the higher minimum SOC in the first 24 hours is gained by coasting from the initial altitude of 2000 ft to 760 ft. Although not adopted by any of the optimized networks, this difference in minimum SOC shows that the excess solar energy collected can be used to climb at no cost to the SOC or charge margin, allowing for a gliding segment overnight to reduce the power draw. Gliding segments can be used to reduce the required power overnight, making year-round operations more attainable.

The controls are not linked to the time-of-day or SOC, therefore the resulting trajectory is independent of the time of year. This network is trained specifically for solar conditions on 21 June. The same trajectory across different months does not mean that this turning radius is optimal for all times of year, merely that a circular trajectory is a suitable approach for LALE operations.

Other constant radius circular trajectories found by the algorithm have significantly smaller turning radii but cannot be directly compared to the effectiveness of this network due to the significant differences in cruise airspeed and therefore power draw. There is no distinct difference in performance associated with a smaller or larger turning radius through the NEAT results.

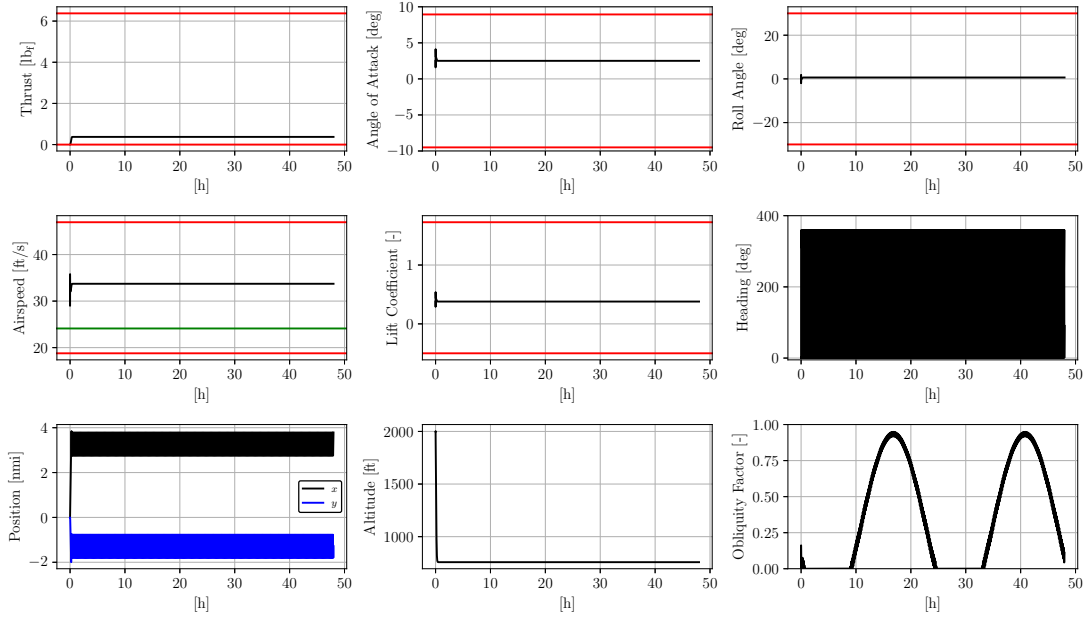


Figure 6.5: States and controls for a constant radius circular trajectory.

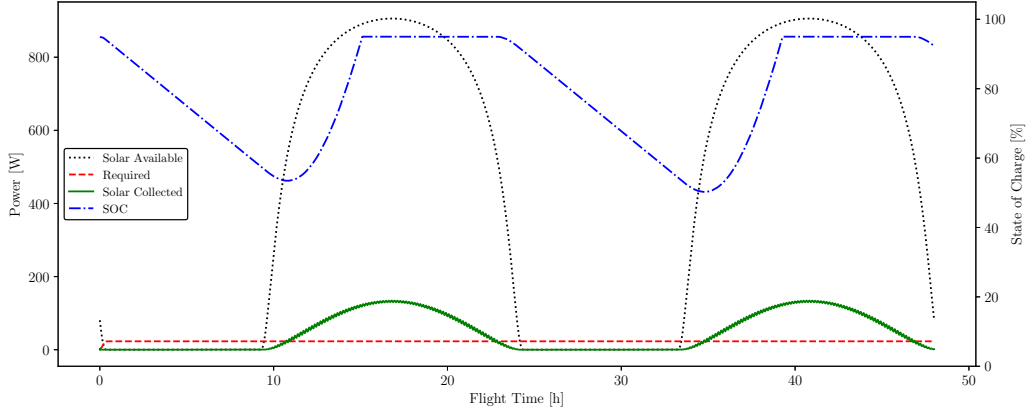


Figure 6.6: Power and SOC throughout the flight with the constant radius circular trajectory.

6.2.2 Variable Radius Circular Trajectory

Another approach within a circular trajectory is to have a variable turning radius, dependent on the time-of-day and sun elevation angle. This result follows similar results found in [68], where the optimal bank angle is dependent on the vehicle mass and sun elevation angle with larger radii at high elevation angles and smaller radii at lower elevation angles. The network that produces this effect has a resulting path shown in Figure 6.7 where the green point is the starting point, the red is the final point, north is in the positive x direction and east is in the positive y direction. The network that results in this trajectory is shown in Figure 6.8, with the weights and biases of the connections and nodes summarized in Table 6.4.

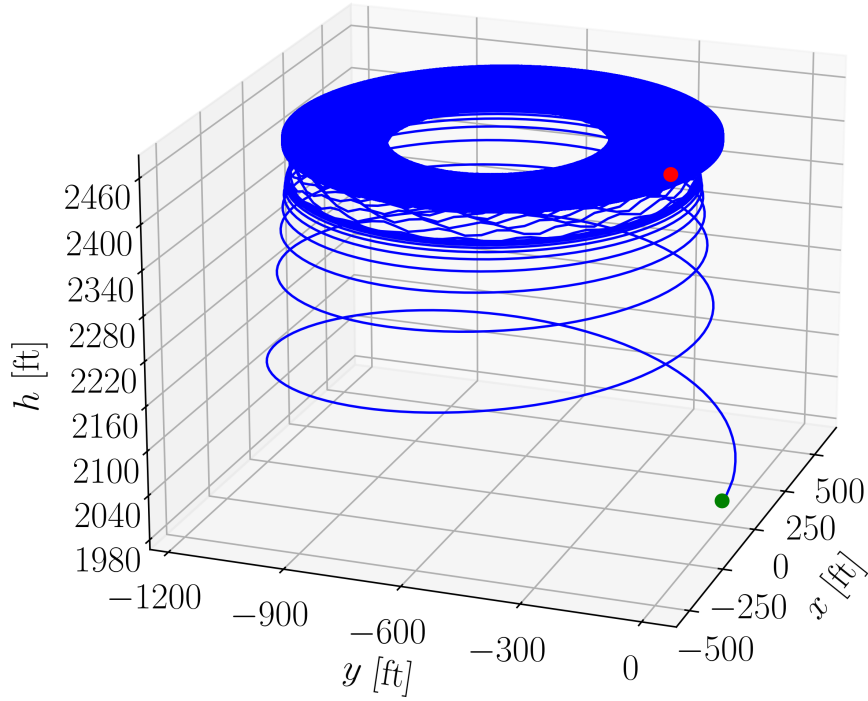


Figure 6.7: Variable radius circular trajectory path.

Table 6.4: Variable radius circular trajectory network weights and biases.

Node Bias		Connection Weight		
Node	Value	From	To	Value
k_0	-1.428	γ	α'	-1.053
k_1	-0.454	SOC	ϕ'	0.078
k_2	0.185	SOC	k_0	2.481
α'	0.127	t	k_1	1.658
ϕ'	-0.160	k_0	k_2	-0.781
T'	0.731	k_1	k_2	-0.336
		k_2	ϕ'	-0.813
		h	T'	-1.233

The states and controls in the first hour of the flight are shown in Figure 6.9, with the remainder of the flight shown in Figure 6.10, with the upper and lower bounds plotted in red where applicable. The green horizontal line in the airspeed plot indicates the minimum drag airspeed, with the actual airspeed shown in black. The available, collected, and required power and SOC throughout the flight are shown in Figure 6.11.

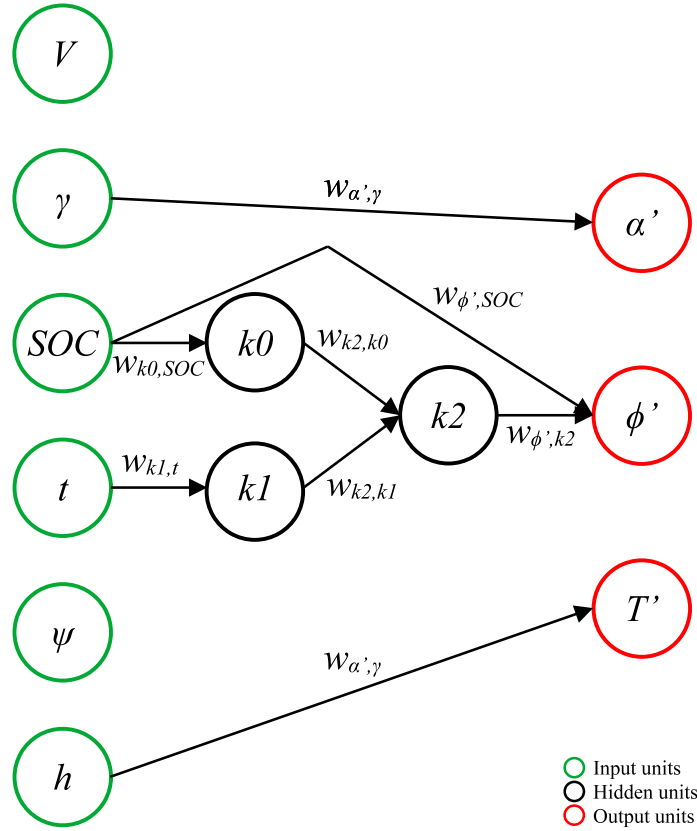


Figure 6.8: Network for the variable radius circular trajectory.

This network results in a response where the aircraft is porpoising. For this network to be viable for real-time control, penalties and smoothing of the controls would be necessary in the evolution of the network. Considering that the objective of this chapter is to determine suitable flight strategies for LALE solar-powered flight, the jittery effect of the network is neglected and the general path and flight strategy is the realistic result.

The connections from the SOC and time-of-day to the roll angle causes the vehicle to have a higher bank angle in the morning which results in greater alignment to the solar vector at the cost of greater variations in the obliquity. The connections between pitch, angle of attack, altitude, and thrust are similar to the constant radius trajectory turn, but the weights of the connections are different. In the variable radius trajectory, the vehicle climbs to an altitude that oscillates around 2450 ft. The average airspeed is approximately $35 \frac{\text{ft}}{\text{s}}$ with a maximum turning radius of 525 ft. The start time of the simulation is slightly after the time that the battery would begin to discharge. The initial climb segment at a higher thrust and power setting combined with the slightly delayed-from-initial-discharge start time causes the minimum SOC in the first 24 hours to have a smaller difference than the constant radius trajectory with the gliding segment. In the first 24 hours, the minimum SOC is 55.9% and is slightly lower at 54.2% in the last 24 hours. The slight offset in the start time highlights the importance of

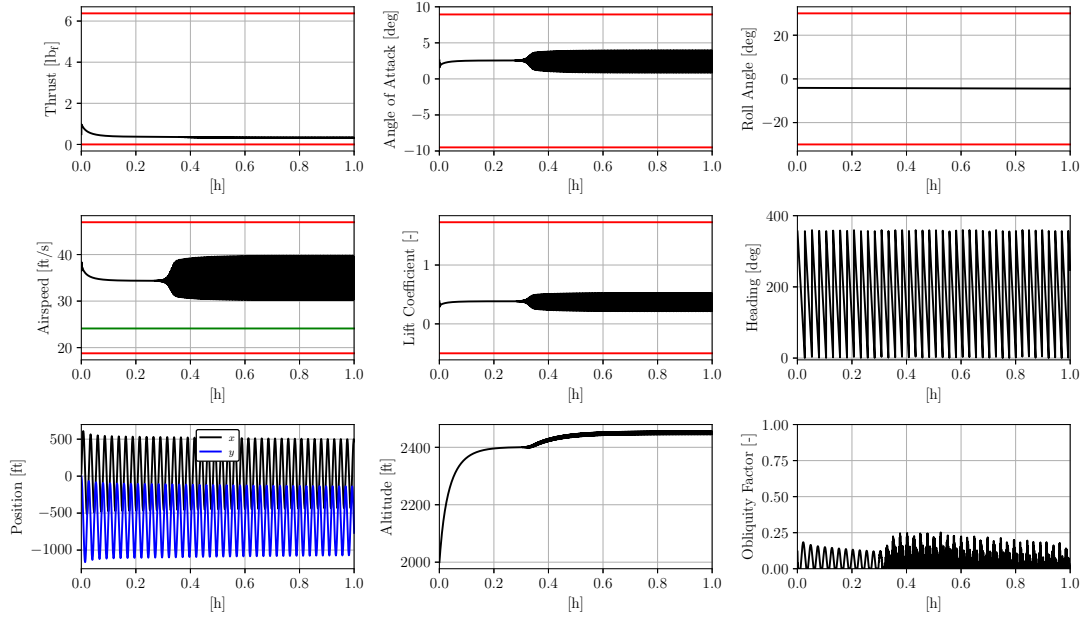


Figure 6.9: States and controls for the first hour of a variable radius circular trajectory.

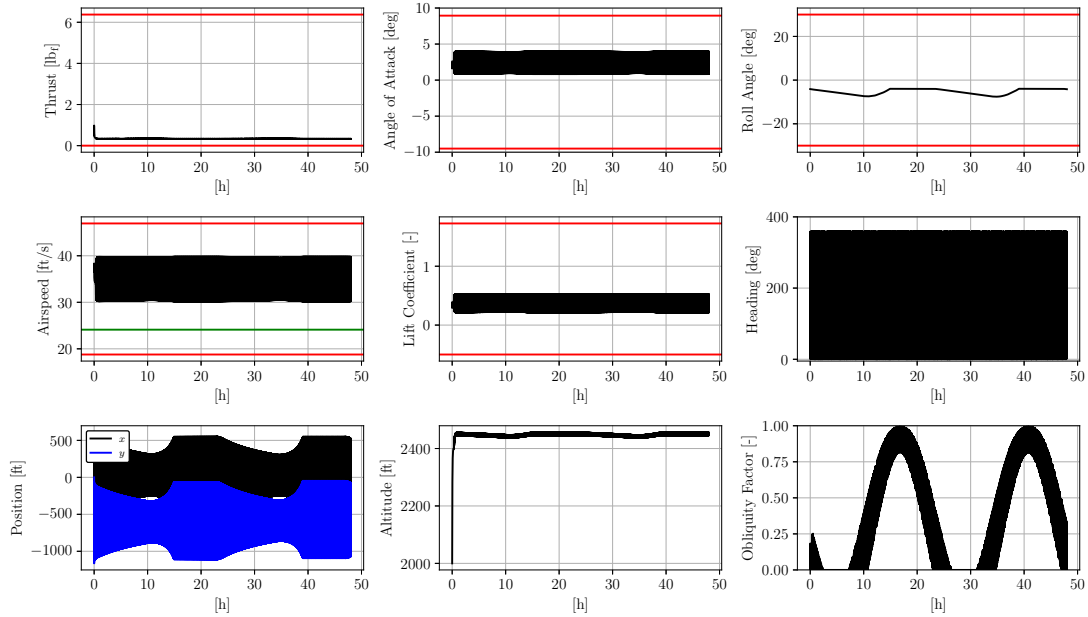


Figure 6.10: States and controls for a variable radius circular trajectory.

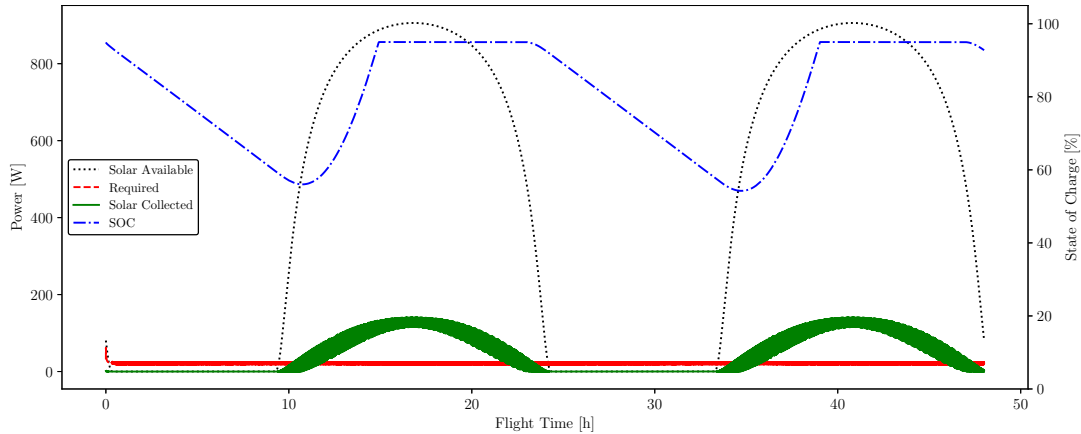


Figure 6.11: Power and SOC throughout the flight with the variable radius circular trajectory.

simulating the second day, to ensure the vehicle can recharge after a realistic discharge. This network has a fitness of 63.35 h, which is marginally better than the constant radius trajectory.

Without the jittery effect of the controller and flying at the minimum power airspeed, the variable radius circular trajectory offers strong battery charging performance to help increase the charge margin. Further optimization studies such as the one Edwards *et al.* in [68] would be beneficial to determine the best angles for different solar elevation angles. This flight strategy would result in different turning radii depending on the time of year, which may not be practical for all applications involving monitoring over a greater area outside of the optimal turning radius.

Although this network is trained specifically for the solar conditions on 21 June, the roll dependency on the SOC significantly improves the performance in other times of the year. Of all the networks produced, this was the only one capable of exceeding 24 hours of continuous flight time on the winter solstice. The trajectory, states, and power curves for the network in Figure 6.8 simulating flight for 21 December are shown in Figures 6.12 to 6.14.

The vehicle was unable to fully recharge, but was able to adequately align the solar vector to the solar cells to maintain flight throughout the night until sunrise, due to the connection between the SOC and roll control. The SOC dropped as low as 31% before recharging. In any real flight, the flight would likely be terminated before this point to preserve the safety of the electrical system. Any in flight disturbances that would increase the power draw would likely render the vehicle unable to exceed 24 hours on the solstice. Even though the vehicle is able to achieve this in simulation, LALE flight would not be possible since the SOC did not follow a full recharge cycle and was based on the assumption of a maximum SOC at sunset. Nonetheless, the strategy of a variable radius circular trajectory is a good strategy for increasing energy margins and charging the batteries quickly.

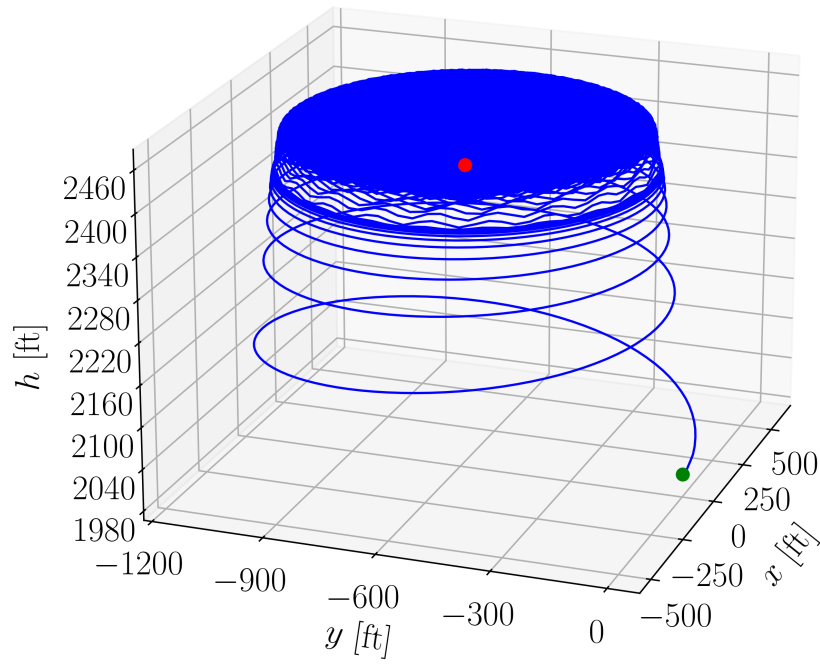


Figure 6.12: Variable radius circular trajectory path, 21 December.

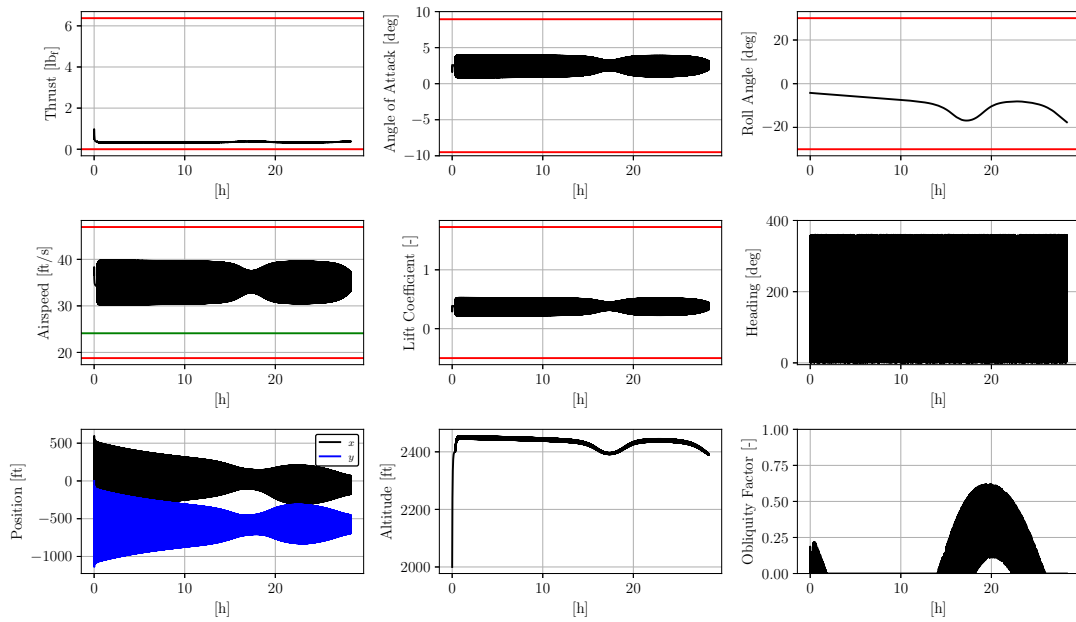


Figure 6.13: States and controls for a variable radius circular trajectory, 21 December.

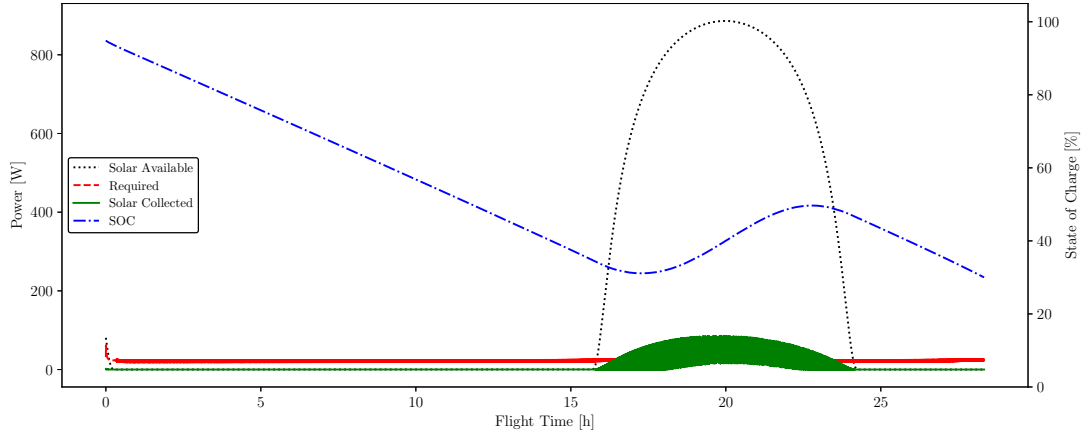


Figure 6.14: Power and SOC throughout the flight with the variable radius circular trajectory, 21 December.

6.2.3 Constant Heading Trajectory

The other common result from the NEAT algorithm for the LALE solar-powered UAV model is a constant heading trajectory. In various results, the vehicle descends to an altitude then holds a constant altitude, constant airspeed, constant heading trajectory. One of the resulting trajectories descends to 570 ft then holds a 244° heading at $33.7 \frac{\text{ft}}{\text{s}}$, shown in Figure 6.15 where the starting position is in green, the final position is in red, north is in the positive x , east to the positive y .

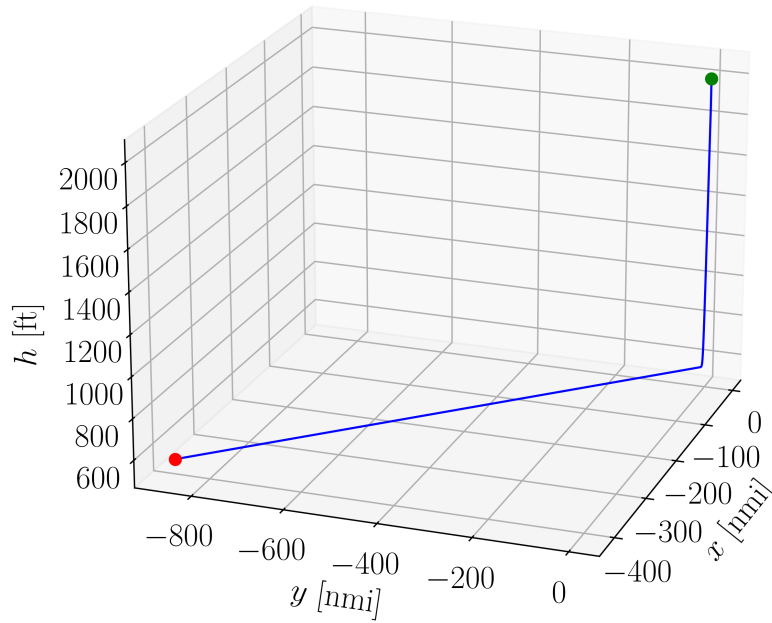


Figure 6.15: Constant heading trajectory path.

The network that resulted in this trajectory is shown in Figure 6.16, with the weights and biases of the connections and nodes summarized in Table 6.5.

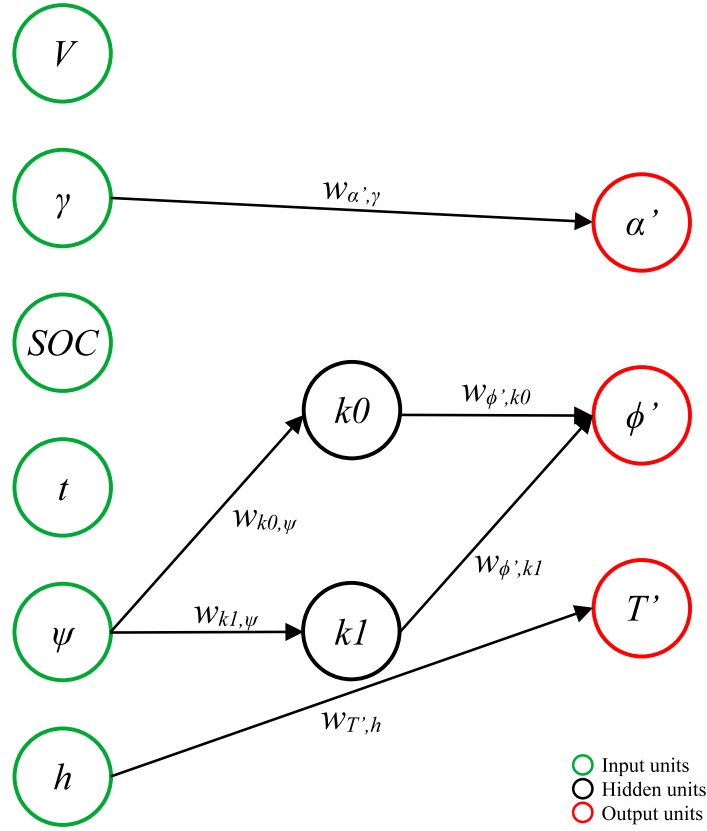


Figure 6.16: Network for the constant heading trajectory.

Table 6.5: Constant heading trajectory weights and biases.

Node Bias		Connection Weight		
Node	Value	From	To	Value
k_0	-0.139	γ	α'	-1.509
k_1	-1.089	ψ	k_0	-0.129
α'	0.124	k_0	ϕ'	3.577
ϕ'	-0.823	ψ	k_1	0.302
T'	-0.102	k_1	ϕ'	1.010
		h	T'	-1.826

The states and controls throughout the flight are shown in Figure 6.17 with the upper and lower bounds plotted in red where applicable. The green horizontal line in the airspeed plot indicates the minimum drag airspeed, with the actual airspeed shown in black. From the trajectory and the altitude, the vehicle appears to be diving from the

initial altitude to the cruising altitude. However, the airspeed and heading angle show that the vehicle is gliding in a spiral pattern to the cruising altitude before it continues its straight path. The available, collected, and required power and SOC throughout the flight are shown in Figure 6.18.

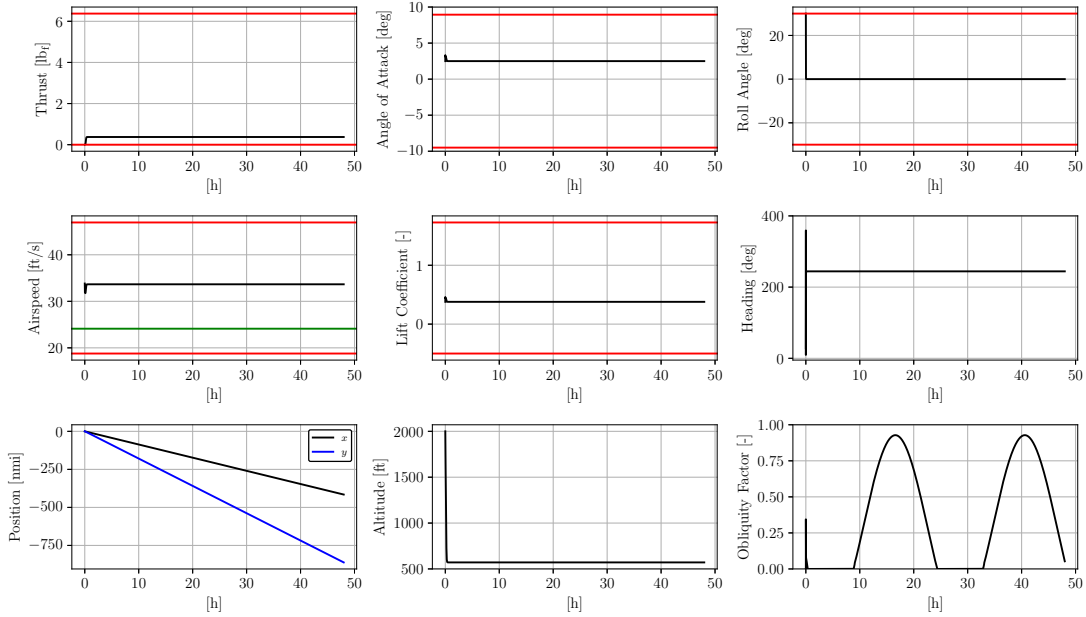


Figure 6.17: States and controls for a constant heading trajectory.

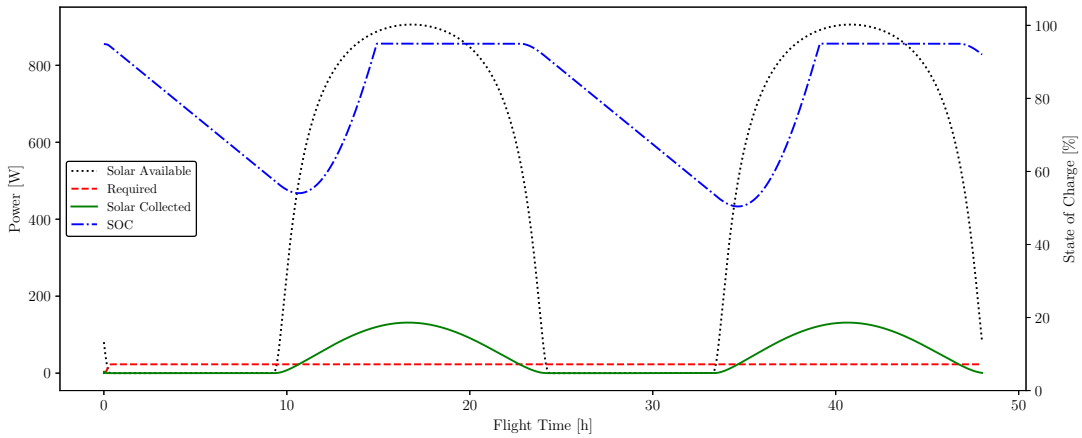


Figure 6.18: Power and SOC throughout the flight with the constant heading trajectory.

The connections of this network differ from the constant turning radius through the connections to the roll angle. In this network, the roll is strictly driven by the heading of the vehicle. The trajectory maintains a heading of 244 deg, effectively flying towards the direction of the sunset. As with the other networks, there are no connections to the airspeed, which does not allow the network to settle on the minimum drag airspeed

to minimize the power draw. The total fitness of this network is 63.40 h, the highest amongst all networks presented but still not as high as the projected fitness from the assumed trajectory from the vehicle optimization in Chapter 5. The difference in the minimum SOC from the first 24 h to the last 24 h is attributed to the glide segment at the beginning of the flight, which reaches a 54.0% SOC in the first night and 50.4% in the second night.

6.3 Summary of Trajectories

All three networks are capable of recharging the simulated solar UAV and maintaining flight through 48 h in the summer months. Although none of the resulting networks or flight strategies achieved a fitness as high as the projected fitness from the vehicle optimization, the results show promising strategies for achieving LALE solar-powered flight and that the proposed geometry in Chapter 5 is capable of multi-day flights.

None of the networks connect the airspeed to the controls nor did they settle at the minimum power airspeed. It is recognized that the fitness of each network is dependent on the airspeed and the comparison of the fitness values does not necessarily indicate that one trajectory is better than the other. The fitness of each presented network at different months is summarized in Table 6.6, where the diminishing fitness is attributed to shorter daylight hours and lower elevation angles. Each network was trained specifically for the irradiance and sun position for 21 June. The constant radius circular trajectory and the constant heading trajectory were not trained nor connected to any states to alter the path to the solar conditions of any other time of year. The variable radius circular trajectory was the only strategy that adjusted its path throughout the year due to the roll connection to the SOC. This made it more successful through the winter months, but did not properly consider the advantages of decreasing the flight speed to the minimum power speed. To properly compare the flight strategies, the vehicles should be flown at the same airspeed to properly account for the differences in charging rate and charge margin. The NEAT algorithm was used to determine viable flight strategies, not fully optimize the trajectory. The relevant conclusions of this chapter must be based in the practicality of implementing each strategy rather than the direct comparison of the various results.

Table 6.6: Network fitness in hours for different months of the year.

Month	Constant Radius	Variable Radius	Constant Heading
3	56.45	56.72	56.43
6	63.16	63.35	63.40
9	55.55	55.91	55.51
12	15.98	28.33	16.02

Traversing several hundred nautical miles is inconsistent with small solar UAV applications when communication range to its ground station is limited. The constant heading trajectory is not practical for surveillance nor applications requiring station-keeping.

It might be practical to adopt segments of the flight with a constant heading, but ultimately requiring the vehicle to return to its launch point. Constant radius circular trajectories can be similarly faulted if the radius is too great to service the required geographical location or remain within line of sight communication with its ground station. The constant radius trajectory at smaller radii is more practical for communications relay and can easily be adapted into environmental monitoring or surveillance applications. The circular trajectories resulting from the NEAT algorithm justifies the adoption of constant radius circular trajectories for the analyses in Chapters 4 and 5.

Ultimately, a variable radius circular trajectory was the best strategy to maximize the flight time, which achieved the highest minimum SOC overnight. In practice, it is less practical to constantly change the roll angle throughout the flight, which becomes dependent on the precision of a pilot or autopilot. To adopt the approach of the variable radius circular trajectory, it would be beneficial to have a low constant radius, *i.e.*, high bank angle, for low sun elevation angles and a high constant radius, *i.e.*, low bank angle, for higher sun elevations angles, with an instantaneous switch between the two rather than dynamically adjusting the bank angle across several hours.

The benefit of optimizing for increasing the charge margin is that the vehicle will have excess energy to overcome deviations from the idealized modelling and trajectories. The higher power draw from the networks as an artifact of not flying at the minimum power airspeed shows that the vehicle as modelled has energetic robustness worked into the design, which is promising for real applications. The energetic margins make it practical and feasible to deviate from ideal flight conditions, whether that is following non-optimized paths or flying with headwinds or some cloud cover. Ultimately the best path will be dependent on actual flight conditions and the objective of each flight of the vehicle. The resulting trajectories show that a constant heading or circular trajectories can be used for LALE solar-powered flight, where combinations of the two can be exploited for different applications.

7 Conclusions and Recommendations

The main objective of this thesis was to optimize the design of a LALE solar-powered UAV. This chapter provides concluding remarks, areas for improvement, and recommendations for future research and development on this topic.

7.1 Conclusions

Through a detailed analysis of various solar UAV configurations, a clean wing T-tail UAV is the best configuration for LALE solar-powered flight. The required balance between aerodynamic efficiency and onboard energy capacity is highly dependent on the payload and power draw of the vehicle. The ground handling requiring limitations in span and the manufacturing-induced chord length limit prevented the design from achieving a higher aspect ratio in the low payload case and a lower aspect ratio in the high payload case. This was an expected after the parametric comparison. The proposed design was similar in characteristics to the other successful LALE solar-powered UAVs with slim fuselages and high aspect ratios. The proposed vehicle is designed to accommodate two 3S6P battery packs as the upper limit in energy capacity to for a payload power draw of 6.275 W, with a typical mission requiring two 3S3P battery packs for a payload power draw of 2.45 W. The final vehicle has a 10 ft wingspan, 10 in diameter propeller, 2814-1250 brushless DC motor, and maximum takeoff weight of 9.3 lb_f.

A well-designed solar-powered UAV is only half the problem to achieving LALE solar-powered flight. No matter how optimal an aircraft design is, proper flight planning and controlling is critical for achieving the mission. Therefore, flight strategies and trajectories must be carefully considered to minimize the power consumption and maximize the solar cell alignment with the solar vector during the day. Linking the proposed solar UAV characteristics to the NEAT algorithm results shows that a circular trajectory with a variable turning radius is an effective flight strategy for extracting solar energy. Tighter radii at lower SOC's and low sun elevation angles help to align the solar cells to the solar vector for better charging, while higher radii at higher sun elevation angles have the same effect. The circular trajectory is the most practical for station-keeping, providing the UAV is able to overcome any headwinds to maintain its station of operations.

The performance and ability to achieve prolonged flight at any time of a the year is directly related to the payload and geographical location. The actual performance of the vehicle is difficult to simulate since there are various unaccounted electrical losses

within this analysis. The internal resistance of the assembled solar array, the power consumption of the MPPTs, and the power draw of the autopilot or servomotors were approximated through efficiency factors. The actual assembly of the vehicle and testing of electrical circuits would be required to predict more accurately the endurance and time of year that LALE solar-powered flight is possible with the proposed geometry. The flight time and charge margin throughout different times of the year allowed for the design of a solar-powered UAV that would be theoretically capable of prolonged flight anytime in at least half of the year.

7.2 Recommendations for Future Work

This research can be extended through the fabrication and testing of the proposed geometry. Minor modifications may be necessary for the integration of systems including sensors, solar cells, and cameras. More detailed design will be required for the integration of carbon fibre rods for structural stiffness as well as control surfaces.

There are several avenues that the optimization of the vehicle can be pursued further. First and foremost, an additional fuselage segment should be added to the geometry definition and included in the optimization variable. This would allow for the optimizer to explore solutions with longer fuselages at lower wetted areas with smaller control surfaces, which may have lower drag profiles than the results presented in Chapter 5. A different analysis should be performed with higher upper bounds on the chord lengths, wingspan, and fuselage length to allow a broader scope of $\mathcal{R}$ to be explored by the optimizer. This may show more agreement with the results of the parametric analysis, where it was expected that a higher power demand would result in a lower $\mathcal{R}$ than a vehicle with a lower power demand. Finally, the original optimization problem or with any of these modifications should be conducted at a higher latitude. A significant difference between the results at a higher latitude and the proposed geometry would suggest that there is no one-size-fits-all to achieving LALE solar-powered flight for all times of year.

There are several improvements that could be made to the linking of the NEAT algorithm and the modelled UAV. The optimization relies on propeller and motor data that are dependent on altitude and airspeed. The propeller and motor models should be integrated into the NEAT UAV model to accurately account for the changes in propulsive efficiency with airspeed rather than the constant propulsion efficiency used in this analysis. Penalties should be included in the fitness function to enforce more realistic continuity of the UAV's path, prevent jittery controls, and/or remain within a given range of the simulation's starting point. Furthermore, the networks should be evolved for various dates to develop a flexible controller capable of responding and maximizing the energy margins regardless of the time of year. With proper models, the NEAT algorithm could be used to consider thermal updrafts or wind gradients. These should be included in the trajectory planning for a LALE solar-powered UAV to explore additional means to extract energy from other natural phenomena and further prolong flight.

The best trajectory from the NEAT analysis was a variable radius circular trajectory. In practice, this approach requires dynamic updates to the bank angle and should be

reflective of the sun's position through a given day. Rather than determining the best angle for every instance in a day, the approach adopted by Edwards *et al.* should be used to determine the best bank angle for maximum solar power collection for discrete times of day, *i.e.*, for sunrise, for noon, and for sunset [68]. Three distinct bank angles would allow for better solar power collection throughout the day than a singular, constant angle, resulting in faster charging rates and higher energy margins.

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Appendices

A Solar Powered Aircraft

This appendix lists solar airplanes that have been built and tested in Table A. The non-comprehensive list primarily builds off of work from Noth [6, 12]. Solar cells and batteries have improved in efficiency and capacity while decreasing in cost, allowing solar flight to be explored by hobbyists which is not commonly recorded or published. Therefore, it is more difficult to record every solar aircraft that has flown in recent history. This tabulated history of solar powered aircraft consists of significant and documented aircraft. It is recognized that there may be missing aircraft from this summary, published or unpublished.

Table A.1: Non-comprehensive list of flown fixed-wing solar aircraft to date.

No.	Name	Year	Wingspan, b [ft]	Mean Chord, $\bar{c}$ [ft]	Length [ft]	Wing Area, S [ft ²]	Aspect Ratio, $\mathcal{R}$ [-]	Empty Weight, W_e [lb _f]	Total Weight, W_0 [lb _f]	Longest Recorded Flight, E [h]	Reference
1	Sunrise	1974	31.99	2.82	14.37	90.00	11.4	27.0		4	[12]
2	Sunrise II	1975	31.99	2.82	14.37	90.00	11.4	22.5			[12]
3	Solaris	1976	6.76	0.66		4.41	10.3	1.34		0.0417	[12]
4	Ra	1977	4.49	0.39	2.76	1.72	11.9	0.42			[12]
5	Utopie	1977	8.30	0.66	4.33	5.49	12.6	2.14			[12]
6	Solar-Student	1978	6.43	0.72	3.41	4.63	8.91	2.05			[12]
7	Solar One	1978	67.98	3.84	21.98	259.97	17.8	220			[12]
8	Solar-X4	1979	8.20	0.56	3.71	4.52	14.8	1.87			[12]
9	Solar Silberfuchs	1979	13.12	0.82	4.99	10.76	16	4.6			[12]
10	Solar Riser	1979	29.99	3.41	8.01	102.48	8.78	123.0	275		[12]
11	Solar-HB79	1980	9.19	0.79	4.76	7.21	11.7	3.3			[12]
12	Solair I	1980	52.50	4.53	17.72	236.83	14	264.6	441	5.68	[12]
13	Gossamer Penguin	1980	71.00	8.63		613.60	8.22	68.0	149		[12]
14	Solar-HB-80	1981	9.32	0.75	4.86	7.00	12.5	3.79			[12]
15	Solar Challenger	1981	46.67	4.86	30.25	235.00	9	220.0	337	5.38	[12]
16	Solus Solar	1984	10.50	0.95	2.89	10.01	11	4.85			[12]

No.	Name	Year	b [ft]	$\bar{c}$ [ft]	Length [ft]	S [ft ²]	$\mathcal{R}$ [-]	W_e [lb _f]	W_0 [lb _f]	E [h]	Reference
17	Poly	1986	10.63	0.95	2.89	10.44	10.8	5.47			[12]
18	Combi	1987	9.71	0.85	2.79	8.29	11.4	5.05			[12]
19	Solariane	1987	10.11	0.92	5.64	9.15	11.2	3.97			[12]
20	Helios (model)	1989	7.02	0.59		4.20	11.8	3.09			[12]
21	Blocj	1989	9.51	0.79		7.54	12	2.76			[12]
22	Grosholz	1989	10.07	0.62		6.46	15.8	4.08			[12]
23	Combi 2	1989	9.68	0.92	5.05	8.29	11.3	3.75			[12]
24	Ikaros	1989	8.20	0.75		6.24	10.8	3.97			[12]
25	Bleher	1989	6.56	0.79		5.27	8.18	3.02			[12]
26	Romarino	1989	6.56	0.66		4.31	10	3.97			[12]
27	Sol-e-moi	1989	9.84	0.56		5.38	18	4.63			[12]
28	Wolf	1989	9.84	0.69		6.78	14.3	3.53			[12]
29	WS-Solar	1989	8.20	0.72		5.92	11.3	3.42			[12]
30	Ariane Ultra	1990	6.50	0.69	3.74	4.41	11	6.66			[12]
31	Solar Voyager	1990	10.50	0.82		8.50	13	2.87			[12]
32	Mardini	1990	7.87	0.82		6.46	9.6	5.51			[12]
33	Sollisolar	1990	9.78	0.75		10.33	12.9	2.71			[12]
34	PB 26-FL	1990	8.53	0.72		6.24	11.8	5.07			[12]
35	Solarbaby	1990	5.58	0.52		3.01	10.4	2.76			[12]
36	Bleher	1990	6.56	0.72		4.74	9.03	3.42			[12]
37	Uccello	1990	8.86	0.75		6.78	11.5	4.19			[12]
38	Sole Florentino	1990	8.20	0.56		4.63	14.6	2.65			[12]
39	Soli	1990	6.82	0.59		4.09	11.5	3.31			[12]
40	Playboy	1990	7.87	0.62		4.84	12.8	2.98			[12]
41	WS-12 (then WS16)	1990	8.20	0.52	3.61	4.41	15.2	1.85			[12]

No.	Name	Year	b [ft]	$\bar{c}$ [ft]	Length [ft]	S [ft ²]	$\mathcal{R}$ [-]	W_e [lb _f]	W_0 [lb _f]	E [h]	Reference
42	Solar Flyer	1990	8.66	0.75	4.86	6.57	11.5	3.53			[12]
43	Blue Chip	1990	7.22	0.75	4.10	5.38	9.66	1.65			[12]
44	Solarmax	1990	11.42	0.98	1.94	11.20	11.6	5.60			[12]
45	Sollisolar 89-2	1990	9.78	0.75	4.40	7.32	13.1	2.73			[12]
46	Phonix	1990	8.60	0.69	4.23	6.03	12.2	2.60			[12]
47	Sunseeker	1990									[12]
48	Solar UHU	1991	7.55	0.75	3.94	5.71	10	3.2			[12]
49	Blue-Wing	1991	7.68	0.59	3.45	4.52	13	1.7			[12]
50	Solar Schilti 1	1991	5.71	0.62	3.81	3.66	9	1.5			[12]
51	Solar Schilti 2	1991	6.53	0.59	3.45	3.88	11.1	1.8			[12]
52	Solozo Solar	1991	7.38	0.69	4.27	5.06	10.7	2.4			[12]
53	Solix	1991	7.78	0.66	4.27	5.17	11.8	2.3			[12]
54	Solar mini challenger	1992	5.09	0.59		3.01	8.5	2.1			[12]
55	Rival-8 Solaris	1992	6.43	0.72	3.71	4.63	8.91	1.5			[12]
56	Pathfinder	1994	98.43	8.53	11.81	762	12.3	456.4	556		[3]
57	MikroSol	1995	3.71					0.419			[12]
58	Solair II	1996	65.62	2.82	20.08	183	23.5	308.6	507		[12]
59	Icare II	1996	82.03	3.38	25.26	277	24.3	595.2	794		[12]
60	Lo 120 Solar	1996	50.72	3.38		172	14.9				[12]
61	Solarflugzeug	1996	59.06	4.92		291	12	418.9	617		[12]
62	O sole mio	1996	65.62	4.04		264	16.3	286.6	485		[12]
63	Solar Solitude	1996	8.86	0.66		5.92	13.3	4.4			[12]
64	NanoSol	1996	3.64					0.353			[12]
65	Centurion	1997	202.77	7.87	11.81	1597	25.8	1175	1900		[12]
66	Trosollmuffel	1997	8.20	0.82		6.67	10.1	2.5			[12]

No.	Name	Year	b [ft]	$\bar{c}$ [ft]	Length [ft]	S [ft ²]	$\mathcal{R}$ [-]	W_e [lb _f]	W_0 [lb _f]	E [h]	Reference
67	Global Flyer	1997	8.20	0.75	3.94	6.14	11	2.3			[12]
68	Pathfinder Plus	1998	119.10	7.87	11.81	937.84	15.1	545.6	694		[12]
69	Solar Excel	1998	6.89	0.52	3.35	3.77	12.8	1.6		11.57	[12]
70	Solitair	1998	17.06								[12]
71	PicoSol	1998	3.25					0.287			[12]
72	LFMA	1998	6.23	0.82		5.06	7.76	2.6			[12]
73	Helios	1999	247.06	8.14	11.81	2008.74	30.4	1323	2050		[12]
74	Sunrazor (Sunriser)	2000	8.86	0.98		8.72	9.06	2.4		0.12	[12]
75	Goldcap 2	2001									[12]
76	Solarus	2001	7.55	0.62		4.74	12	1.06			[12]
77	FlyG	2002	19.69	1.97	8.86	38.75	10	22.04620			[12]
78	Solar Pleaser	2003	3.41	0.49	3.31	1.61	7	0.551			[12]
79	No Name	2003	0.46	0.05	0.39		9.33	0.00375			[12]
80	Solar Splinter	2003	14.01	1.15	6.99	16.15	12.2	9.9			[12]
81	Sol-Mite	2004	2.66	0.39		1.08	6.5	0.287			[12]
82	Sky-Sailor	2005	10.50	0.79	5.97	8.40	13.2	5.5		27	[12]
83	SoLong	2005	15.58	1.05		16.15	15	27.8		48.2	[10, 12]
84	Zephyr 4	2005	39.37					37.5		6	[14]
85	NanSun	2006	10.50	1.31	8.53	13.78	8	9.0			[12]
86	Howie Mark	2006	7.97	0.66		5.27	12.2	0.992			[12]
87	SunSailor	2006	13.78	1.05	7.22	14.53	13.1	7.9			[11, 12]
88	Aphelion	2006	10.27	0.72		7.54	14				[12]
89	Zephyr 5	2006	52.50					68.3		18	[14]
90	2.765 g Solar MAV	2007	0.46	0.13	0.49	0.06	3.41	0.00485			[12]
91	SolFly	2007	0.23								[12]

No.	Name	Year	b [ft]	$\bar{c}$ [ft]	Length [ft]	S [ft ²]	$\mathcal{R}$ [-]	W_e [lb _f]	W_0 [lb _f]	E [h]	Reference
92	Micro-Mite	2007	0.66	0.16		0.11	4	0.0209			[12]
93	Sun-Surfer I	2007	2.53		2.40	0.75	8.5	0.265			[12]
94	Sun-Surfer II	2007	2.56	0.36	2.43	0.97	7.03	0.414			[12]
95	Zephyr 6	2007	59.06					66.1		54	[15]
96	Zephyr 6	2008	59.06					66.1		82.6	[15]
97	SunLight Eagle	2009	114.00					173.0			[20]
98	Zephyr 7	2010	73.82					116.8		336	[16]
99	Solar Impulse 1	2012	208.02		71.69		19.7		3527	19.1	[22]
100	Sunseeker DUO	2013	72.18					639.3	1036	10	[8]
101	Solar Impulse 2	2016	236.23		71.69				5071	117.9	[21]
102	SolarStratos	2016	81.37		27.89			992			[112]
103	AtlantikSolar	2017	18.67	1.00	6.56	18.68		15.3		81.4	[1]
104	MARAAL	2017	17.55			21.96	14	37.5		8.5	[25]
105	Zephyr 8	2018	82.03					143.3		623.95	[17]
106	No Name	2019	12.69	0.96		12.16	13.24	14.2		0.55	[30]
107	Mini-Phantom	2020	10.50	0.98	5.12	10.05	10.6				[31]
108	Solar Flyer	2020	13.13		5.96	9.15	18.8	7.3		8	[32]
109	Sun	2021	6.56							0.12	[35]
110	CREATeV	2022	20.60					26.5		32	[33]
111	Zephyr 8	2022	82.03					143.3		1536	[18]
112	No Name	2024	16.54	1.02	7.02	16.82		11.0		21.8	[36]
113	Moon Flyer	2024	16.41					16.5		25.9	[37]
114	Skydweller	2025	236.23		71.69				5511.55	74	[41, 42]
115	Zephyr	2025	82.03					165.3		1614.9	[2]
116	PHASA-35	2025	114.84						330	24	[38]

No.	Name	Year	b [ft]	$\bar{c}$ [ft]	Length [ft]	S [ft ²]	$\mathcal{R}$ [-]	W_e [lb _f]	W_0 [lb _f]	E [h]	Reference
117	Sunglider	2025	255.92								[4, 40]
118	Kea Atmos Mk1b	2025	41.01						< 90	8.3	[44]

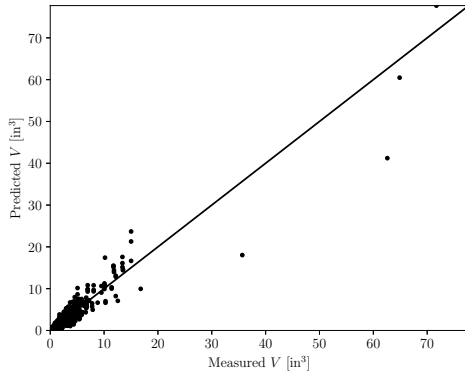
B Additional Size and Weight Correlation Statistics

This appendix summarizes additional statistical correlations not presented in Chapter 3.

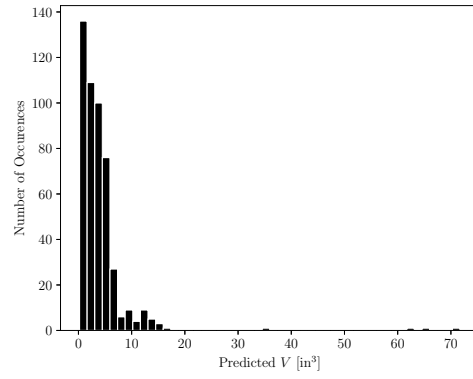
B.1 BLDC Motor Volume

The volume $\bar{V}$ in in^3 of BLDC motors can be predicted using Equation B.1. For this correlation, $N = 489$, $RMSE = 1.9448$, and Adjusted $R^2 = 0.8889$. Equation B.1 can reasonably be used to predict the volume for $\bar{V} < 18 \text{ in}^3$, corresponding to the data distribution shown in Figure B.1b.

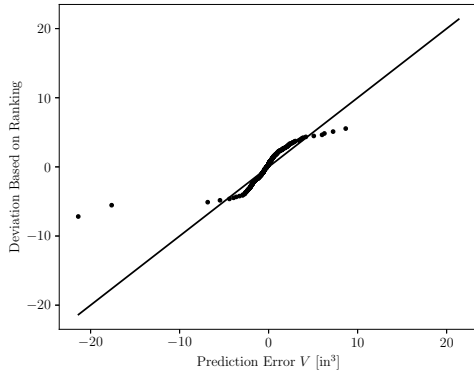
$$\bar{V} \text{ in}^3 = 0.10337 K_V^{-0.17411} I_0^{0.88025} R^{0.20472} P_{max}^{1.14016} I_{max}^{-0.69053} \quad (\text{B.1})$$



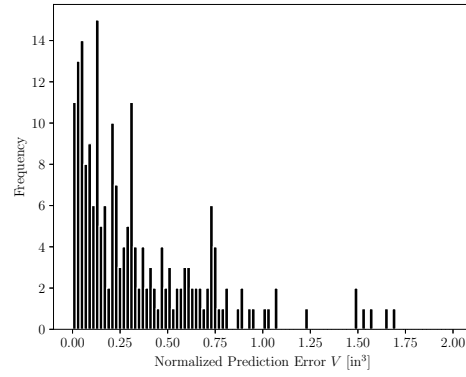
(a) Comparison of predicted to measured volume.



(b) Distribution of predicted volume.



(c) Statistical distribution of errors.

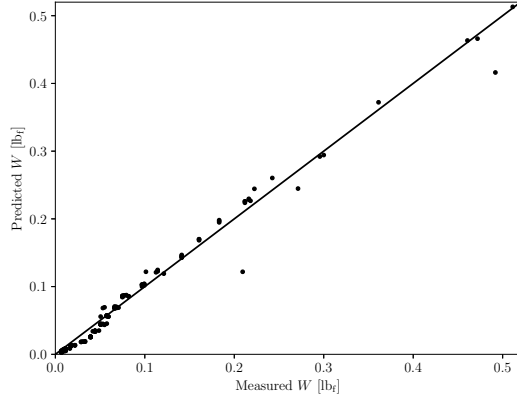


(d) Frequency distribution of errors.

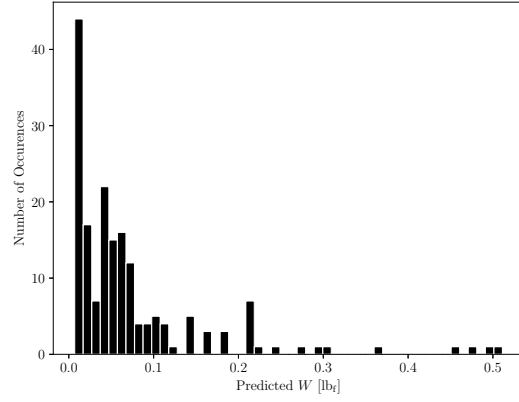
Figure B.1: Visualization of BLDC motor volume prediction.

B.2 Propeller Correlations

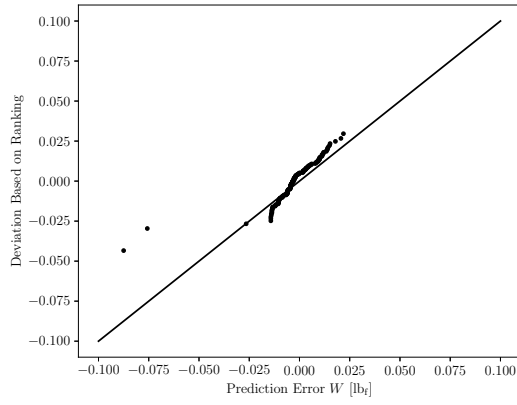
The remainder of this appendix summarizes the statistical analysis graphs for the propeller blade correlations found in Table 3.3. Figure B.2 shows the statistical analysis for the propellers categorized as electric.



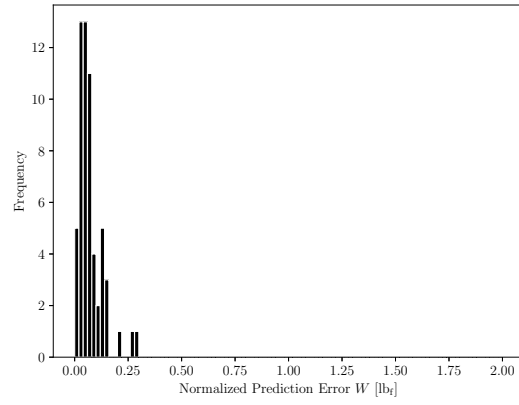
(a) Comparison of predicted to measured weight.



(b) Distribution of predicted weight.



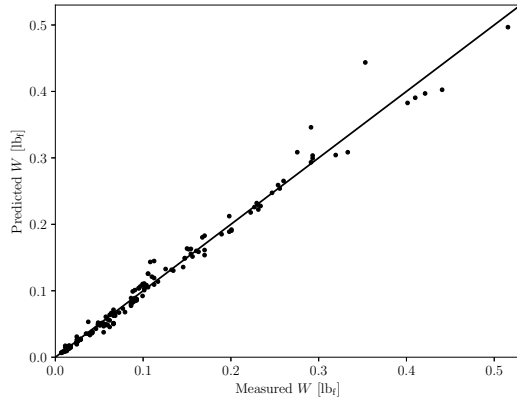
(c) Statistical distribution of errors.



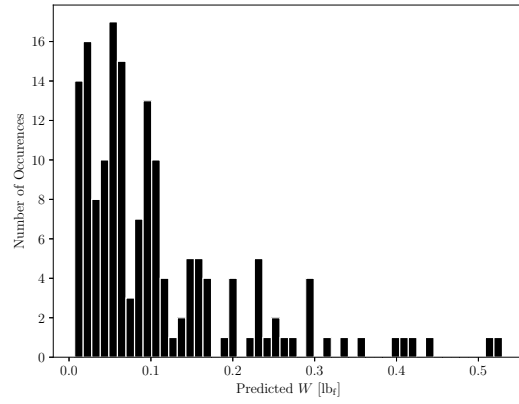
(d) Frequency distribution of errors.

Figure B.2: Statistical validation of 2-blade Electric APC Propeller prediction.

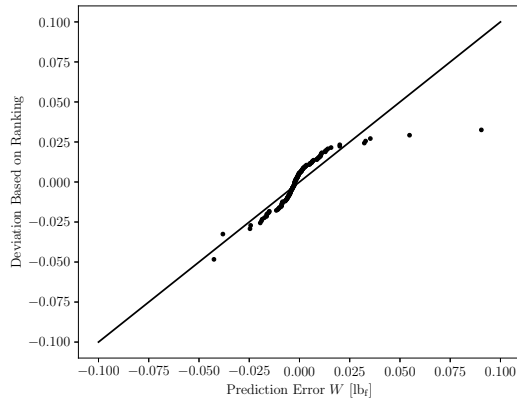
Figure B.3 shows the statistical analysis for the propellers categorized as regular.



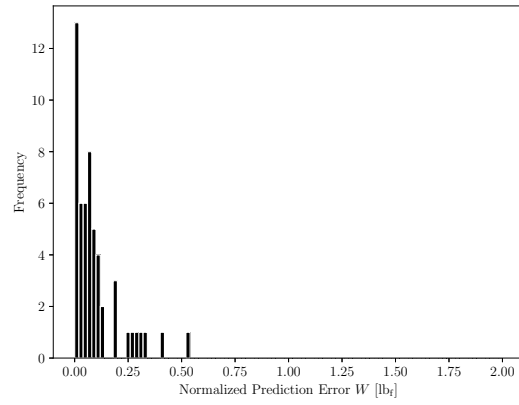
(a) Comparison of predicted to measured weight.



(b) Distribution of predicted weight.



(c) Statistical distribution of errors.



(d) Frequency distribution of errors.

Figure B.3: Statistical validation of 2-blade Regular APC Propeller prediction.

Figure B.4 shows the statistical analysis for the propellers categorized for slow flight.

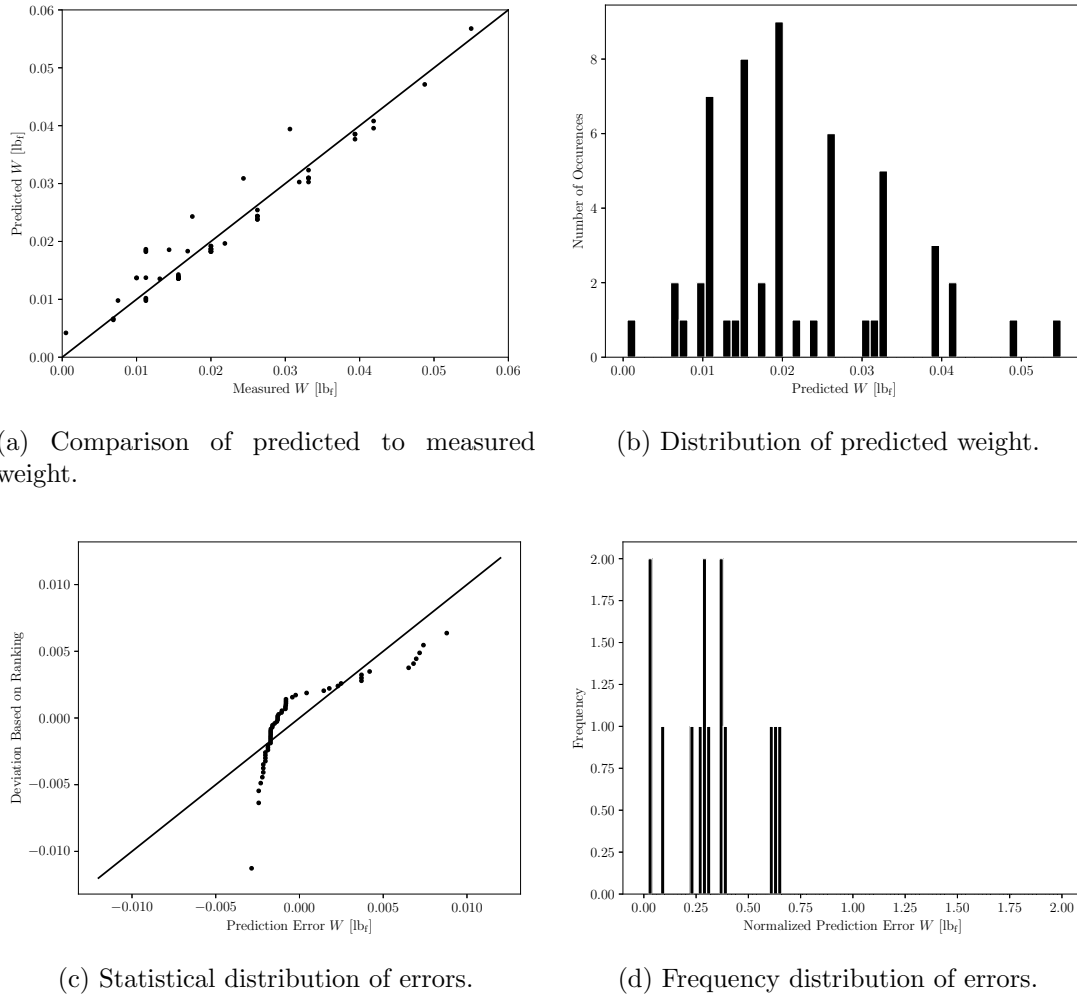
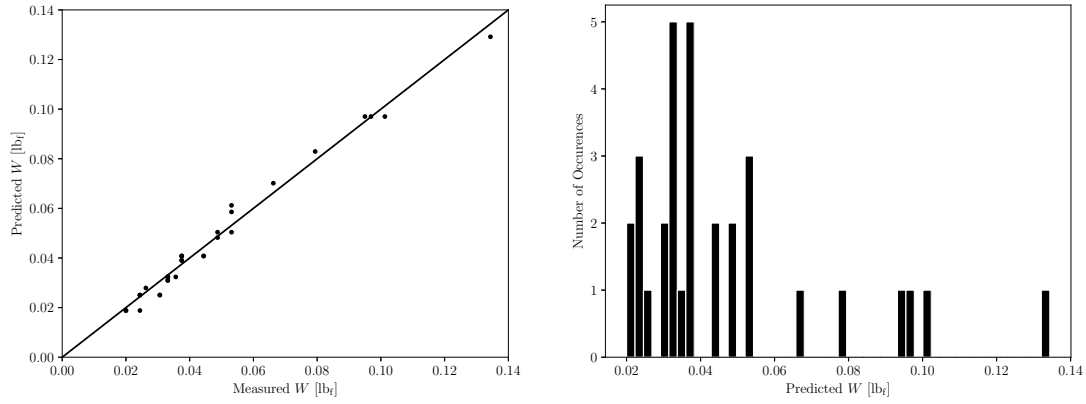


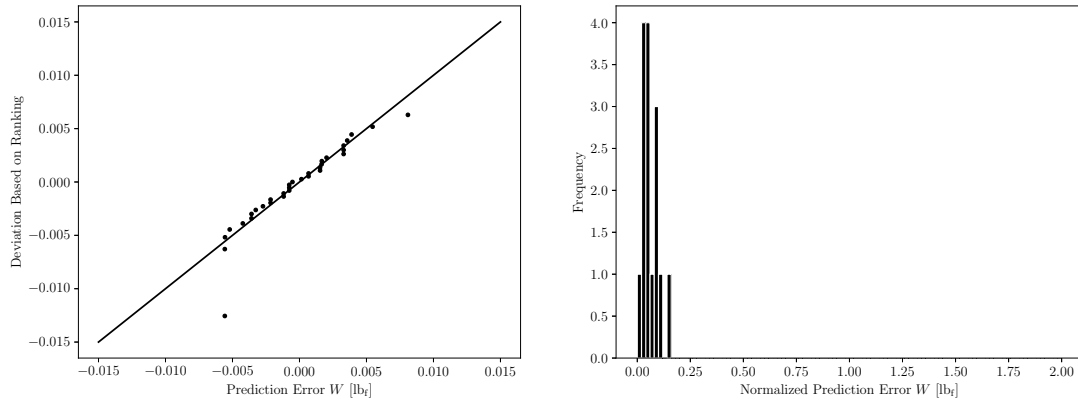
Figure B.4: Statistical validation of 2-blade Slow Flight APC Propeller prediction.

Figure B.5 shows the statistical analysis for the propellers categorized for multi rotors.



(a) Comparison of predicted to measured weight.

(b) Distribution of predicted weight.

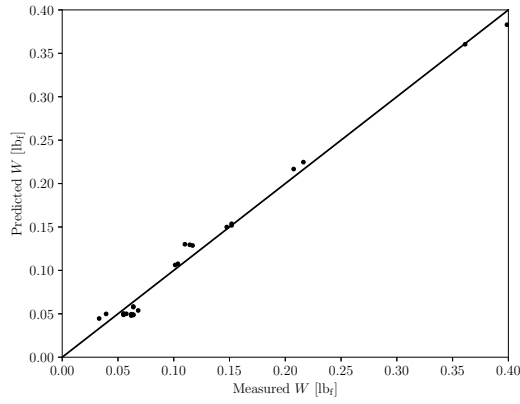


(c) Statistical distribution of errors.

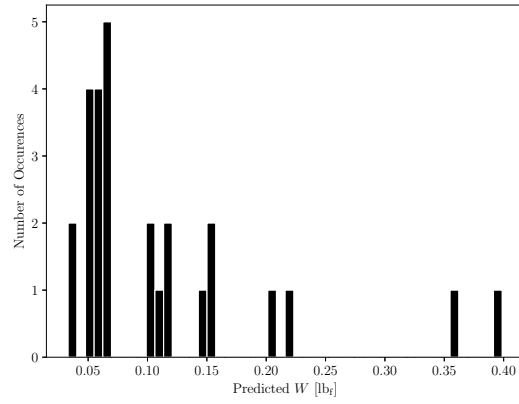
(d) Frequency distribution of errors.

Figure B.5: Statistical validation of 2-blade Multi Rotor APC Propeller prediction.

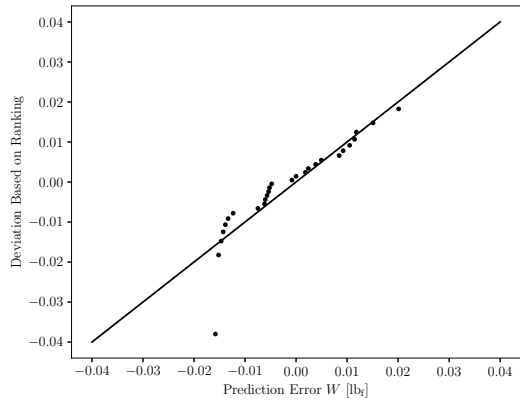
Figure B.6 shows the statistical analysis for the propellers categorized for racing.



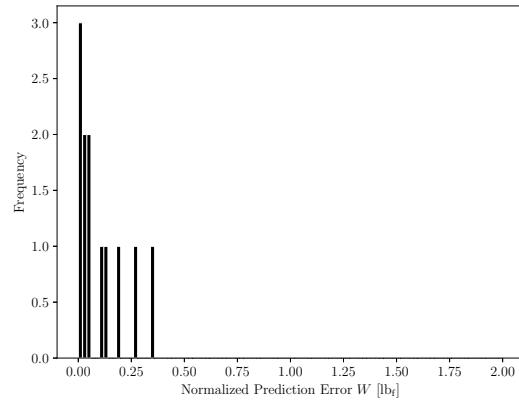
(a) Comparison of predicted to measured weight.



(b) Distribution of predicted weight.



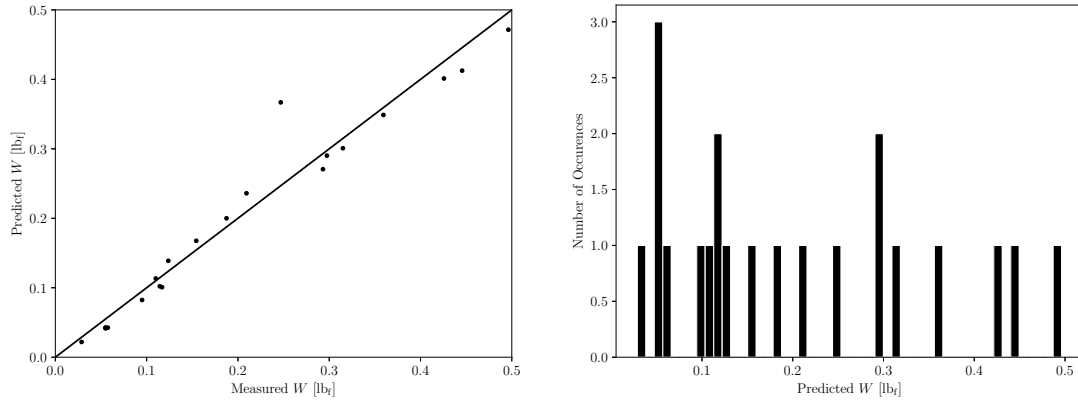
(c) Statistical distribution of errors.



(d) Frequency distribution of errors.

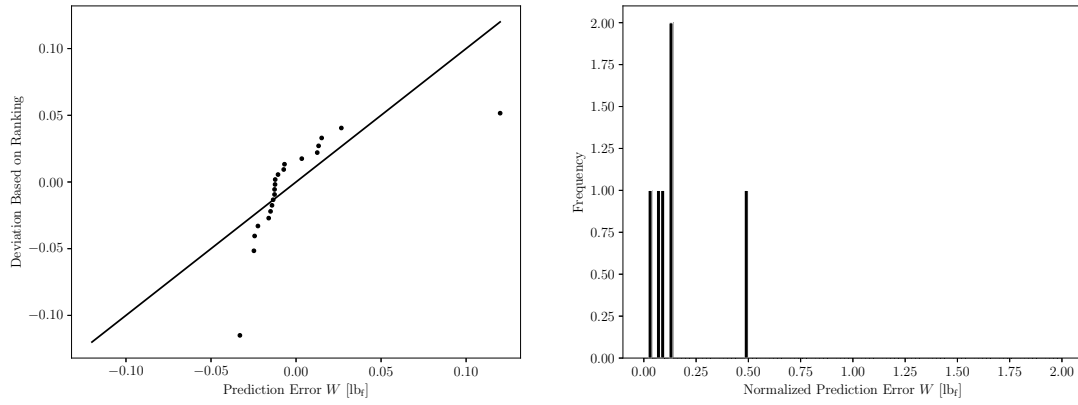
Figure B.6: Statistical validation of 2-blade Racing APC Propeller prediction.

Figure B.6 shows the statistical analysis for the propellers categorized as wide.



(a) Comparison of predicted to measured weight.

(b) Distribution of predicted weight.



(c) Statistical distribution of errors.

(d) Frequency distribution of errors.

Figure B.7: Statistical validation of 2-blade Wide APC Propeller prediction.

C Weight Method Validation

This appendix shows example cases of the weight method presented in Section 3.2.3. Table C.1 shows the geometric properties, weighed weight, and predicted weight of wings from UAVs used within the author's lab group. The measured wing weights have servos installed. The predicted wing structural weight is added to the servo weight for each wing and totalled such that $W_{b_{predicted}} + W_{servo} = W_{b_{predicted\ total}}$. The last row summarizes the relative difference in predicted weight with respect to the measured weight as a percent.

Table C.1: Wing weight validation.

	Eclipson A-XL	Eclipson XL	Volantex Phoenix V2	Volantex 2400
W_0 [lb _f]	3	1.6	2.2	2.7
b [ft]	4.62	3.3	6.56	7.87
S [ft ²]	3.23	1.72	3.88	4.9
$\mathcal{R}$ [-]	6.6	6.6	12.3	12.6
λ [-]	0.4	0.4	0.5	0.5
$\frac{t}{c}$ [-]	0.14	0.13	0.115	0.115
$\frac{\rho}{\sigma}$ [$\frac{1}{ft}$]	4.58×10^{-5}	4.58×10^{-5}	2.69×10^{-5}	2.69×10^{-5}
$W_{b_{predicted}}$ [lb _f]	0.6702	0.2756	0.8863	1.3691
W_{servo} [lb _f]	4×0.0287	2×0.0287	4×0.0198	4×0.0198
$W_{b_{predicted\ total}}$ [lb _f]	0.7850	0.3330	0.9655	1.4483
$W_{b_{actual}}$ [lb _f]	0.7782	0.3351	0.9744	1.4330
Δ [%]	+0.9	-0.6	-0.9	+1.1

Table C.2 summarizes the calculated and actual weights of two commercially available UAVs. The payload in the calculated weights are assumed values.

Table C.2: UAV weight validation.

Weights [lb _f]	RQ 11		Kahu	
	Calculated	Actual	Calculated	Actual
MTOW	4.0 (-4.8%)	4.2	8.7 (+1.2%)	8.6
OEW	2.1		5.2	
ZPW	3.6		6.2	
Batteries	1.5		1.0	
Payload	0.4	0.4	2.5	
Propulsion	0.5		0.6	
Structures	1.5		4.4	
Systems	0.1		0.3	
Reference		[113]		[114]

D Aerodynamics Method Validation

This appendix shows the lift and drag coefficients for a Kahu UAV. The aerodynamic data calculated from *pyACDT* considers the laminar transition to turbulent flow at 50% of the length of all surfaces to account for the low Reynold's Number and smooth skin surface due to the carbon layup of the vehicle. Wind tunnel testing and an analytical analysis using the USAF Digital Datcom are digitized from Shafer et al [115]. Figure D.1 shows the lift and drag coefficients, C_L and C_D , respectively, as calculated by *pyACDT*, Datcom, and experimental wind tunnel testing data. Of the various methods used in the analysis by Shafer et al. for the analysis of the Kahu, the methods integrated into *pyACDT* have the closest agreement, supporting the use of the analysis tool for small UAVs.

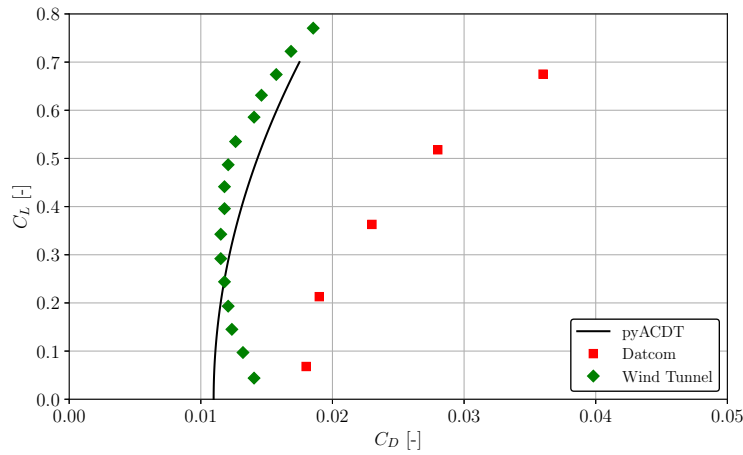


Figure D.1: Kahu drag polar.

E Parametric Analysis Solar Cell Orientation

This appendix presents the method and algorithm used to generate all possible solar array combinations as inputs to the parametric analysis in Chapter 4.

Beginning with $b_{PV1} = n_{PV,wing}/2$, cells are systematically relocated one at a time to rows further downstream while satisfying the constraints above. All possible combinations of cell orientations are used as inputs for the parametric analysis. Table E.1 lists the possible cell orientations corresponding to Figure 4.4 where $b_{PV1} = 12$, and the sequential case where $b_{PV1} = 11$ for $n_{PV,wing} = 42$. The procedure for generating the different solar cell orientations on the wing is described by Algorithm 1, where each combination is stored in the list *PVconfigs*.

Table E.1: Wing solar cell orientations at $b_{PV1} = 11$ and $b_{PV1} = 12$ for $n_{PV,wing}/2 = 21$, or 42 total.

b_{PV1}	b_{PV2}	b_{PV3}	b_{PV4}	b_{PV1}	b_{PV2}	b_{PV3}	b_{PV4}
12	9	0	0	11	10	0	0
12	7	2	0	11	8	2	0
12	6	3	0	11	7	3	0
12	5	4	0	11	6	4	0
12	5	2	2	11	6	2	2
12	4	3	2	11	5	5	0
				11	5	3	2
				11	4	4	2
				11	4	3	3

Algorithm 1 GENERATECELLORIENTATIONS($n_{PV,wing}$)

Require: $n_{PV,wing}$: total solar cells on the wing

Ensure: $PVconfigs$: list of all valid non-increasing distributions $[b_{PV1}, b_{PV2}, b_{PV3}, b_{PV4}]$ summing to $n_{PV,wing}/2$, where each $b_{PV_i} \in \{0\} \cup \{2, 3, 4, \dots\}$

```
1:  $PVconfigs \leftarrow []$ 
2:  $b_{PV1} \leftarrow n_{PV,wing}/2$ 
3:  $b_{PV2} \leftarrow 0$ 
4:  $b_{PV3} \leftarrow 0$ 
5:  $b_{PV4} \leftarrow 0$ 
6: while  $b_{PV1} \geq b_{PV2}$  do
7:   if  $b_{PV2} \neq 1$  then
8:     Append  $[b_{PV1}, b_{PV2}, 0, 0]$  to  $PVconfigs$ 
9:   end if
10:   $\hat{b}_2 \leftarrow b_{PV2}$   $\triangleright$  snapshot: inner loops allocate from  $\hat{b}_2$  without modifying  $b_{PV2}$ 
11:   $i_3 \leftarrow 2$   $\triangleright i_3$ : panels to allocate across the  $b_{PV3}+b_{PV4}$  tier
12:  while  $i_3 \leq \hat{b}_2 - 2$  do  $\triangleright$  require at least 2 panels remaining in  $b_{PV2}$ 
13:     $b_{PV2}^* \leftarrow \hat{b}_2 - i_3$   $\triangleright$  panels remaining in row 2 after allocating  $i_3$  downstream
14:     $b_{PV3}^* \leftarrow i_3$   $\triangleright$  all  $i_3$  panels go to row 3; row 4 empty for this sub-entry
15:    if  $b_{PV2}^* \geq b_{PV3}^*$  and  $b_{PV3}^* \neq 1$  then
16:      Append  $[b_{PV1}, b_{PV2}^*, b_{PV3}^*, 0]$  to  $PVconfigs$ 
17:    end if
18:     $i_4 \leftarrow 2$   $\triangleright i_4$ : panels to reallocate from the  $i_3$  budget into row 4
19:    while  $i_4 \leq i_3 - 2$  do  $\triangleright$  require at least 2 panels remaining in  $b_{PV3}$ 
20:       $b_{PV3}^{**} \leftarrow i_3 - i_4$ 
21:       $b_{PV4}^{**} \leftarrow i_4$ 
22:      if  $b_{PV2}^* \geq b_{PV3}^{**} \geq b_{PV4}^{**}$  and  $b_{PV3}^{**} \neq 1$  and  $b_{PV4}^{**} \neq 1$  then
23:        Append  $[b_{PV1}, b_{PV2}^*, b_{PV3}^{**}, b_{PV4}^{**}]$  to  $PVconfigs$ 
24:      end if
25:       $i_4 \leftarrow i_4 + 1$ 
26:    end while
27:     $i_3 \leftarrow i_3 + 1$ 
28:  end while
29:  if  $b_{PV2} = 0$  then
30:     $b_{PV1} \leftarrow b_{PV1} - 2$ ;  $b_{PV2} \leftarrow b_{PV2} + 2$ 
31:  else
32:     $b_{PV1} \leftarrow b_{PV1} - 1$ ;  $b_{PV2} \leftarrow b_{PV2} + 1$ 
33:  end if
34: end while
35: return  $PVconfigs$ 
```

F Extended Parametric Analysis Results

Annex to show trends from Chapter 4 apply to other spans

Any gaps in data are due to the limitations of the geometry generation in the empenage sizing or limitations of *pyACDT*.

Figures F.1 through F.3 show consistent trends at different b_{PV1} in support of the analysis of Figure 4.9 in Section 4.5 with zero payload.

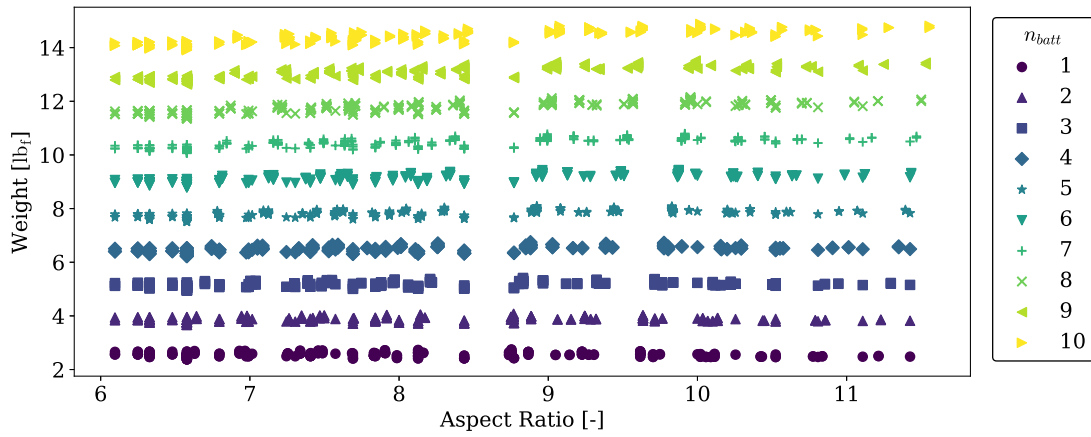


Figure F.1: Effects of $\mathcal{R}$ on total weight for $b_{PV1} = 10$, identified by n_{batt} .

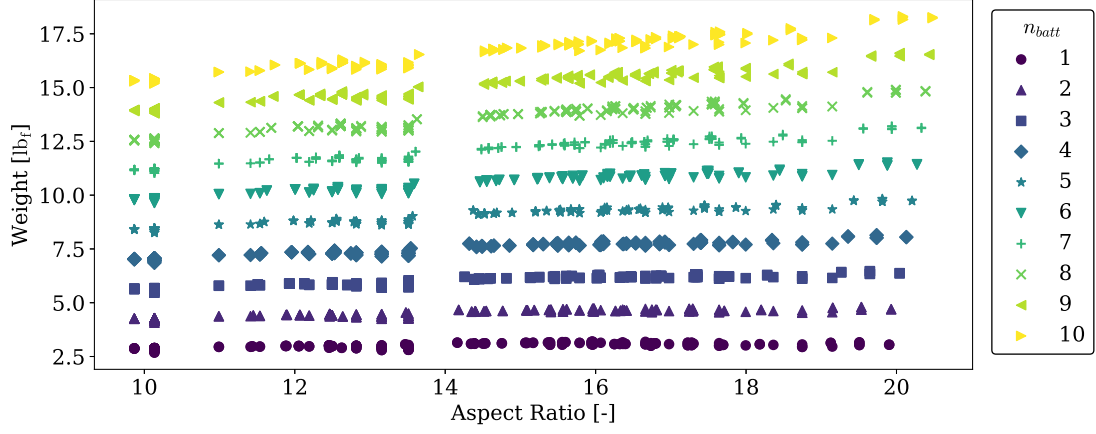


Figure F.2: Effects of $\mathcal{R}$ on total weight for $b_{PV1} = 15$, identified by n_{batt} .

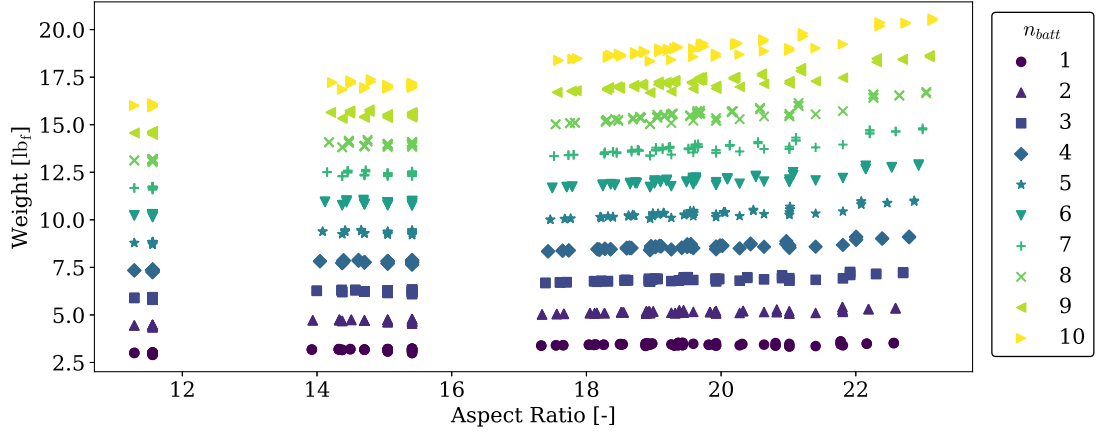


Figure F.3: Effects of $\mathcal{R}$ on total weight for $b_{PV1} = 17$, identified by n_{batt} .

Figures F.4 through F.6 show consistent trends at different b_{PV1} in support of the analysis of Figure 4.11 in Section 4.5 with zero payload.

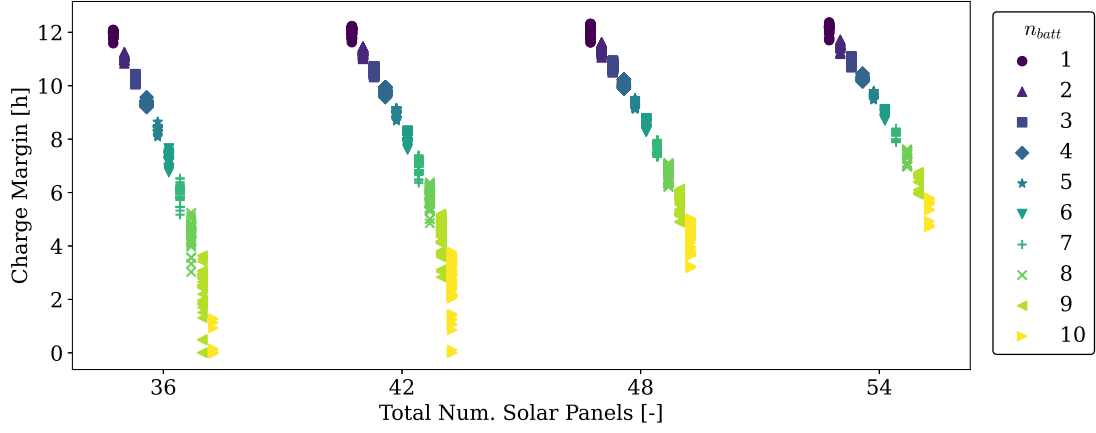


Figure F.4: Charge margin as a function of n_{PV} , identified by n_{batt} with no payload for $b_{PV1} = 10$.

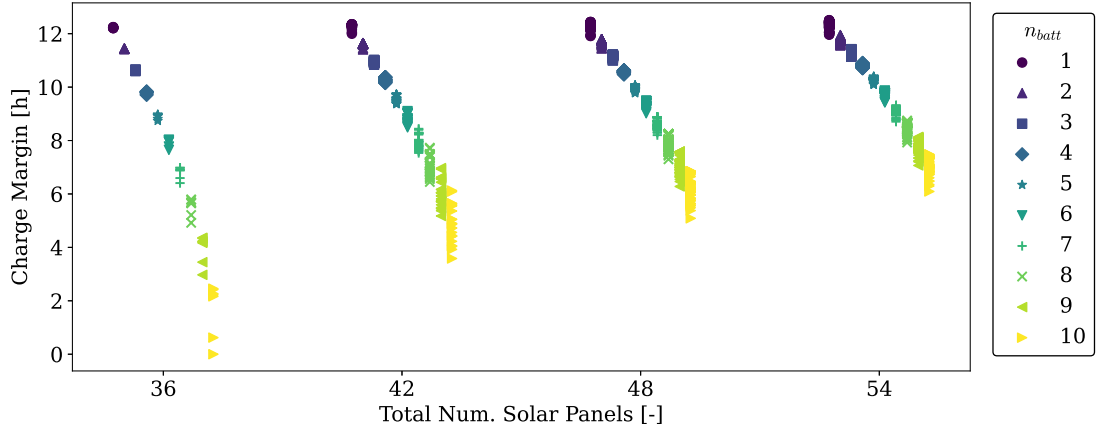


Figure F.5: Charge margin as a function of n_{PV} , identified by n_{batt} with no payload for $b_{PV1} = 15$.

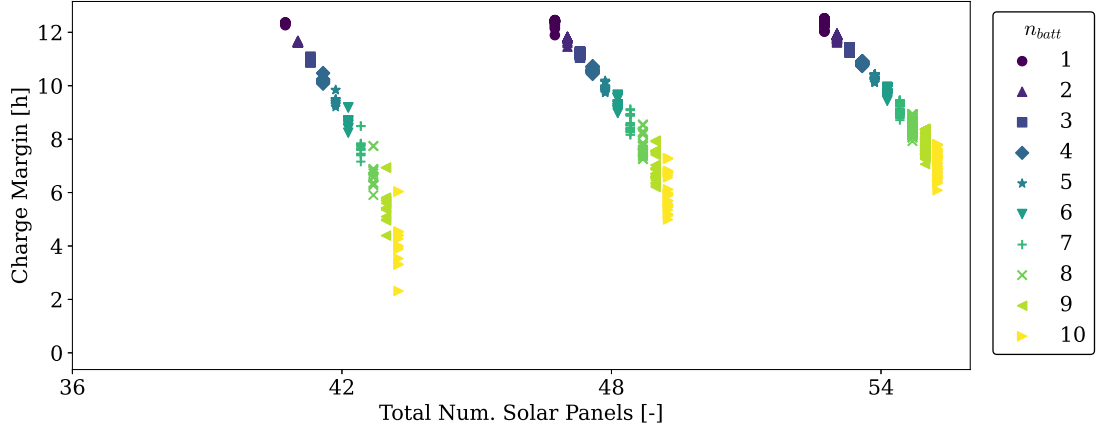


Figure F.6: Charge margin as a function of n_{PV} , identified by n_{batt} with no payload for $b_{PV1} = 17$.

Figures F.7 through F.10 show consistent trends at different b_{PV1} in support of the analysis of Figure 4.11 in Section 4.5 with payload option 2.

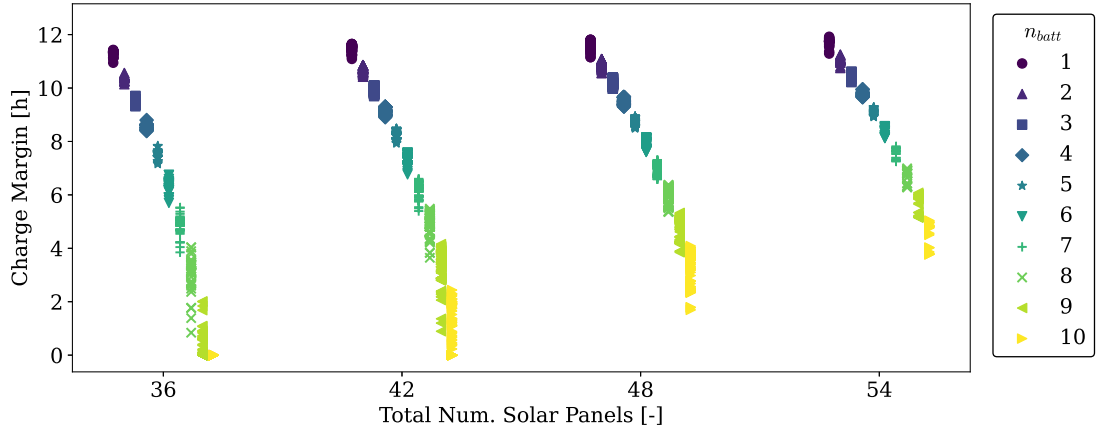


Figure F.7: Charge margin as a function of n_{PV} , identified by n_{batt} with payload option 2 for $b_{PV1} = 10$.

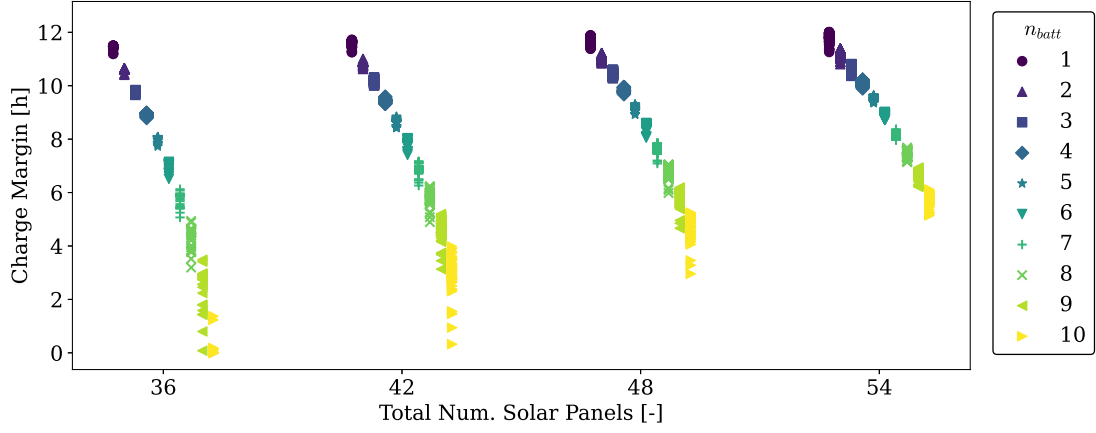


Figure F.8: Charge margin as a function of n_{PV} , identified by n_{batt} with payload option 2 for $b_{PV1} = 12$.

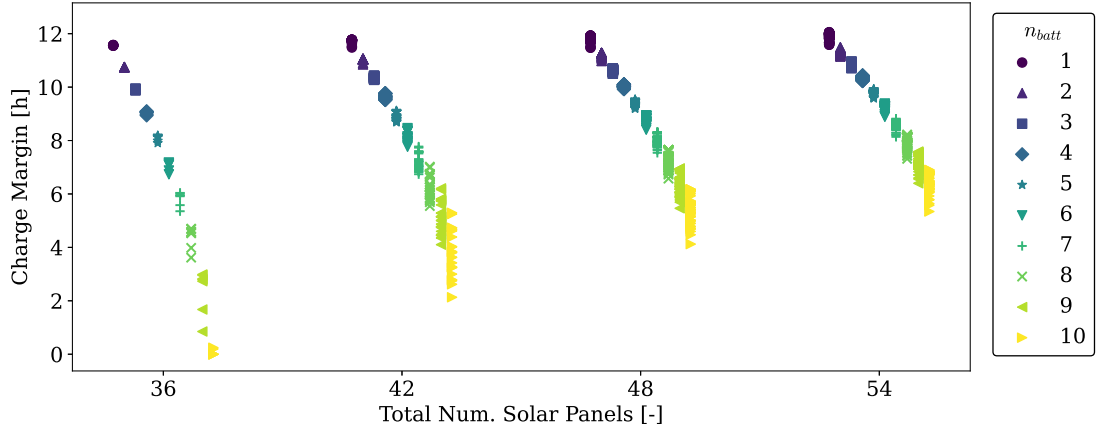


Figure F.9: Charge margin as a function of n_{PV} , identified by n_{batt} with payload option 2 for $b_{PV1} = 15$.

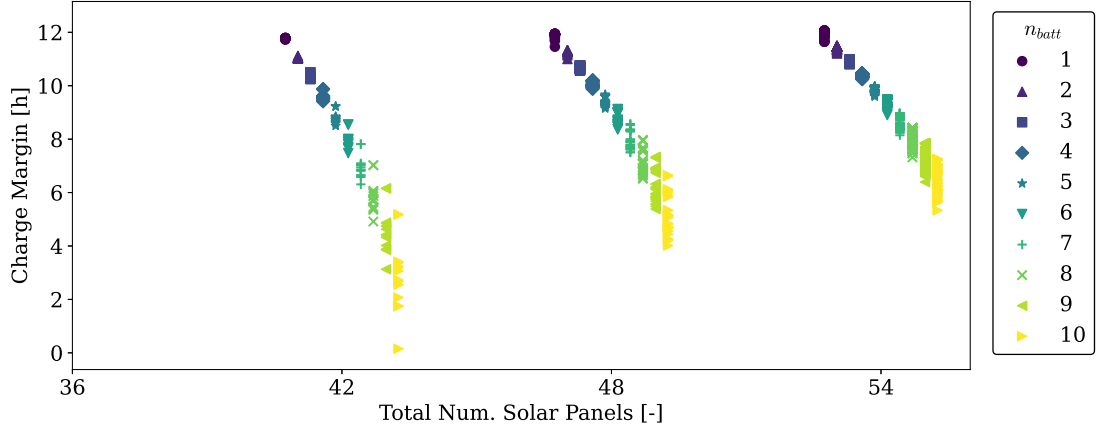


Figure F.10: Charge margin as a function of n_{PV} , identified by n_{batt} with payload option 2 for $b_{PV1} = 17$.

Figures F.11 through F.13 show consistent trends at different b_{PV1} in support of the analysis of Figure 4.13 in Section 4.5.

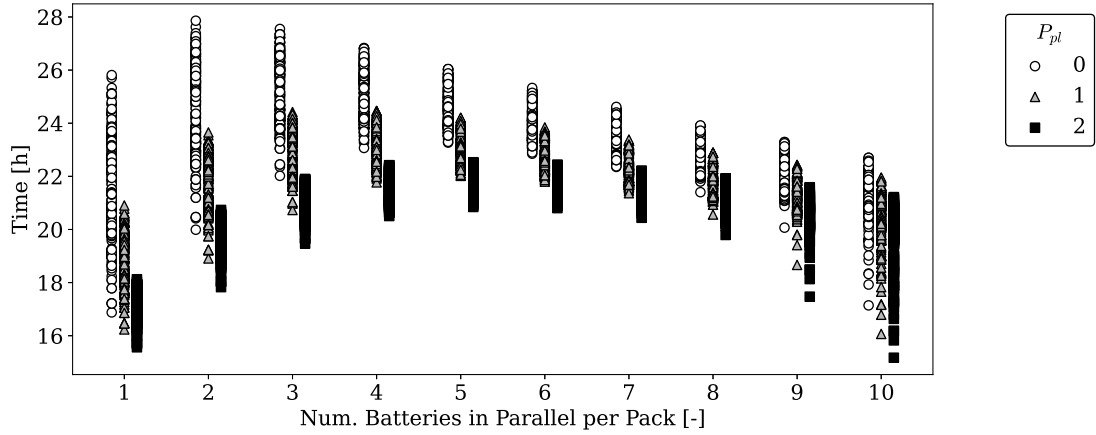


Figure F.11: Flight time as a function of n_{batt} , identified by payload option at $b_{PV1} = 10$.

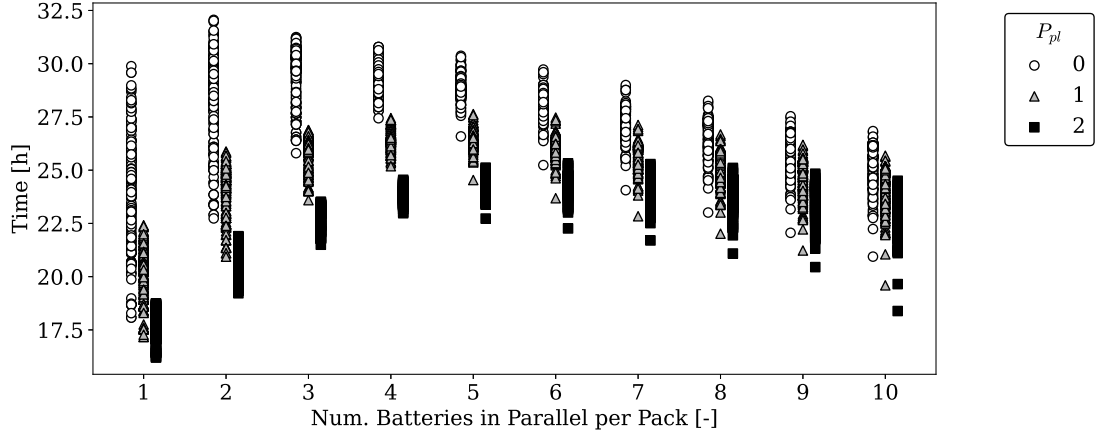


Figure F.12: Flight time as a function of n_{batt} , identified by payload option at $b_{PV1} = 15$.

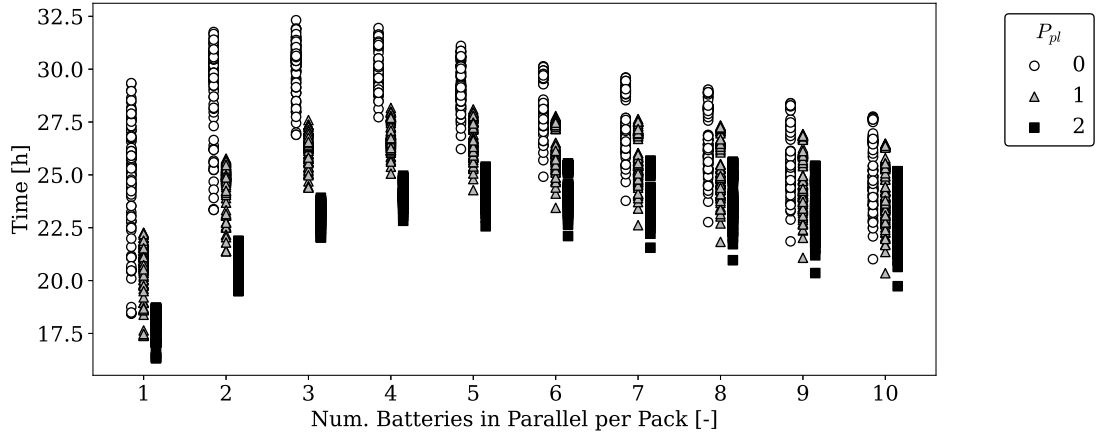


Figure F.13: Flight time as a function of n_{batt} , identified by payload option at $b_{PV1} = 17$.

Figures F.14 through F.16 show consistent trends at different b_{PV1} in support of the analysis of Figure 4.14 in Section 4.5 with zero payload. Note that the scales are not constant in the y -axis or the colour bar. Figures F.17 through F.20 show the same trends but for payload option 2.

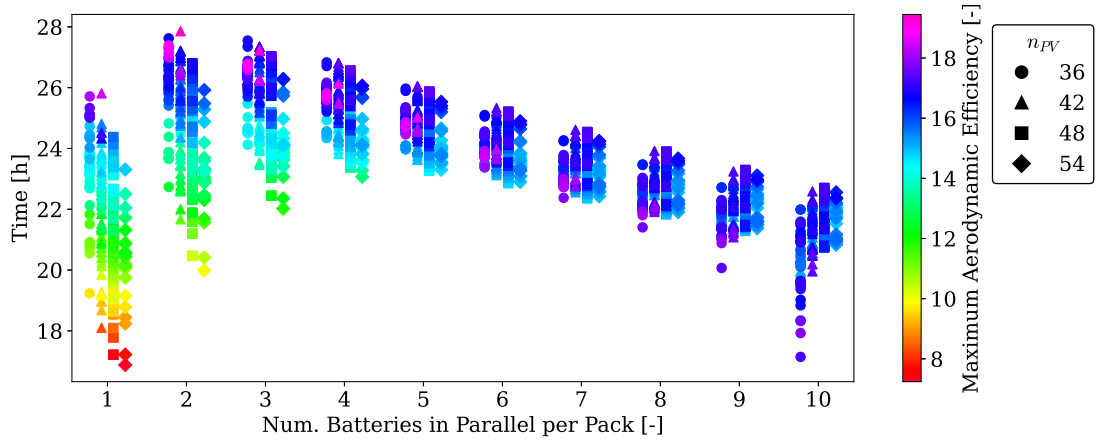


Figure F.14: Flight time as a function of n_{batt} , identified by n_{PV} for zero payload at $b_{PV1} = 10$.

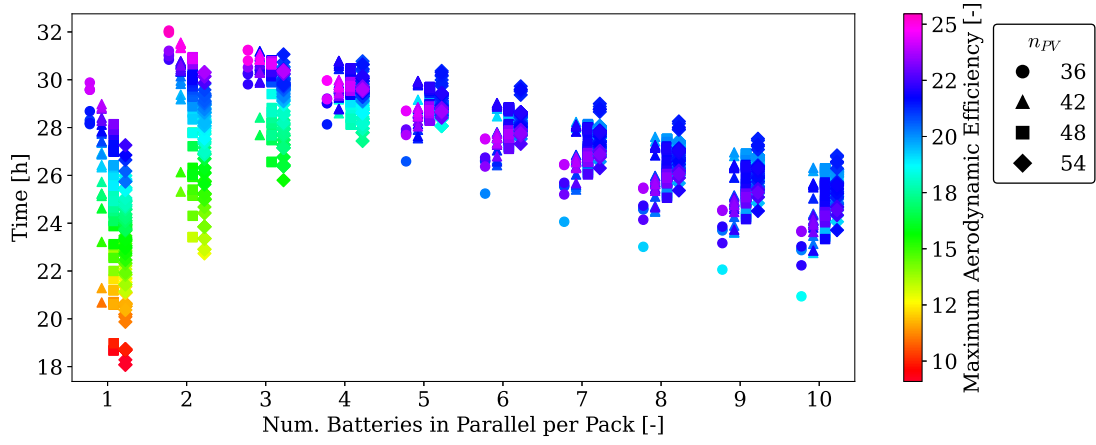


Figure F.15: Flight time as a function of n_{batt} , identified by n_{PV} for zero payload at $b_{PV1} = 15$.

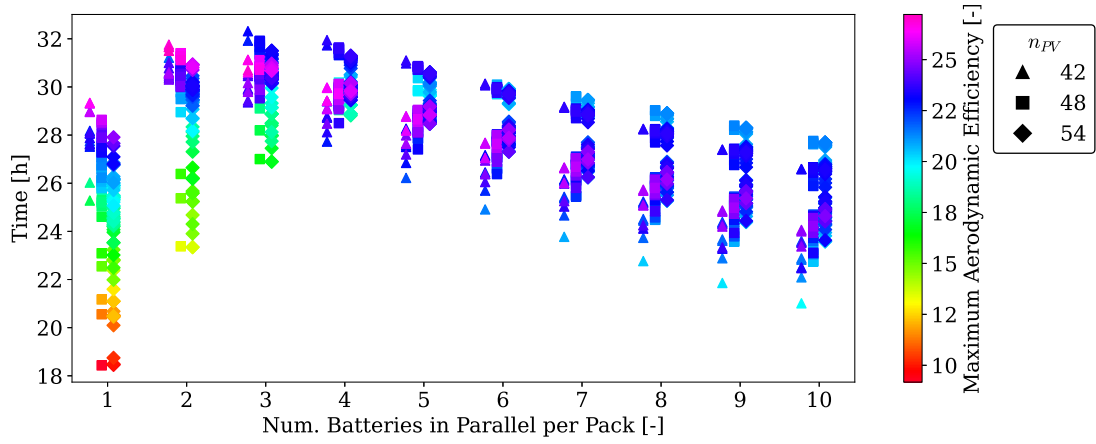


Figure F.16: Flight time as a function of n_{batt} , identified by n_{PV} for zero payload at $b_{PV1} = 17$.

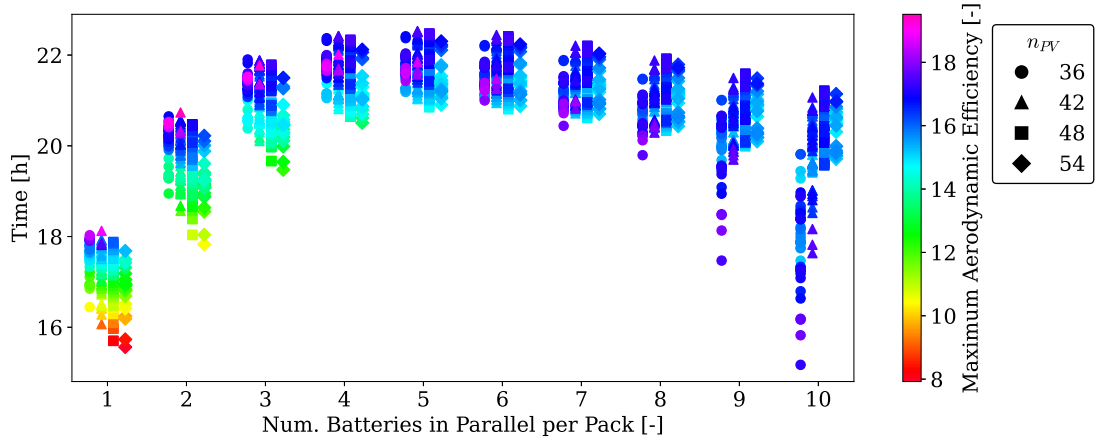


Figure F.17: Flight time as a function of n_{batt} , identified by n_{PV} for payload option 2 at $b_{PV1} = 10$.

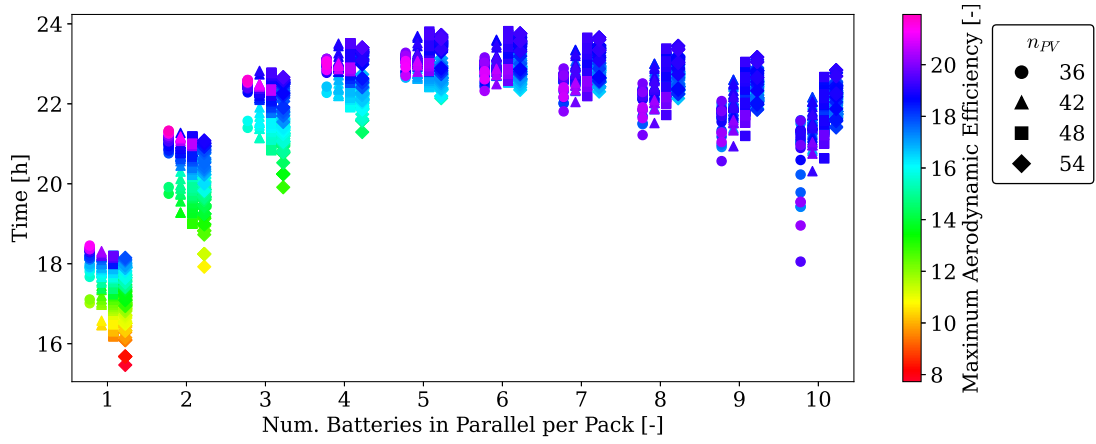


Figure F.18: Flight time as a function of n_{batt} , identified by n_{PV} for payload option 2 at $b_{PV1} = 12$.

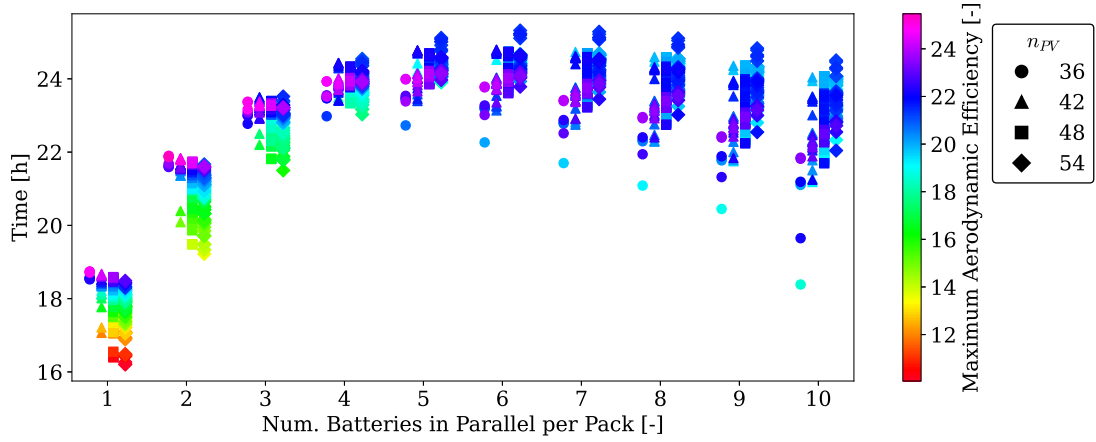


Figure F.19: Flight time as a function of n_{batt} , identified by n_{PV} for payload option 2 at $b_{PV1} = 15$.

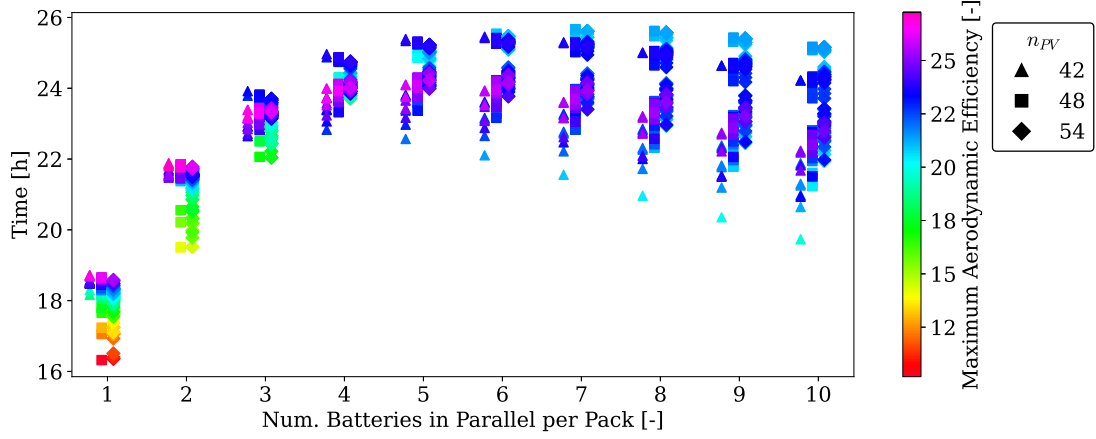


Figure F.20: Flight time as a function of n_{batt} , identified by n_{PV} for payload option 2 at $b_{PV1} = 17$.

Figures F.21 through F.27 show consistent trends at different b_{PV1} in support of the analysis of Figure 4.15 in Section 4.5 with and without payload. Note that the scales are not constant in the y -axis or the colour bar.

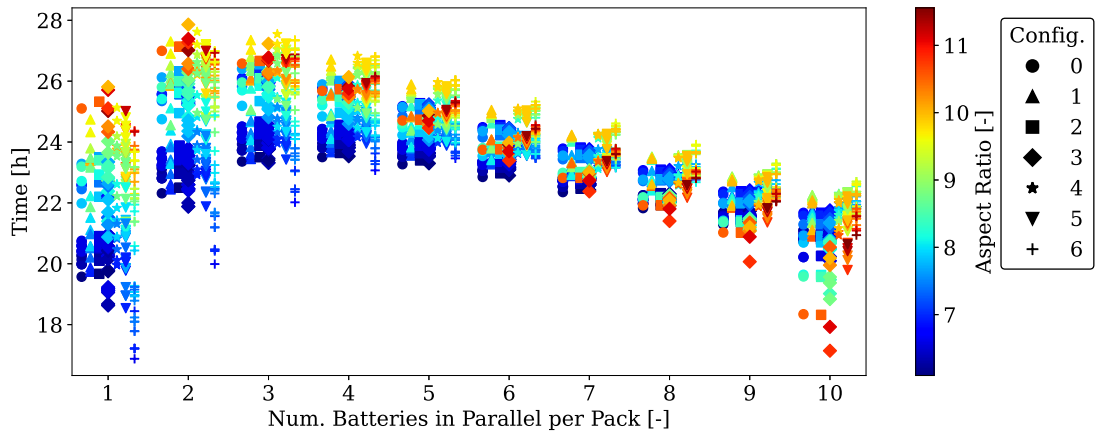


Figure F.21: Flight time as a function of n_{batt} , coloured by $\mathcal{R}$, identified by Configuration for no payload and $b_{PV1} = 10$.

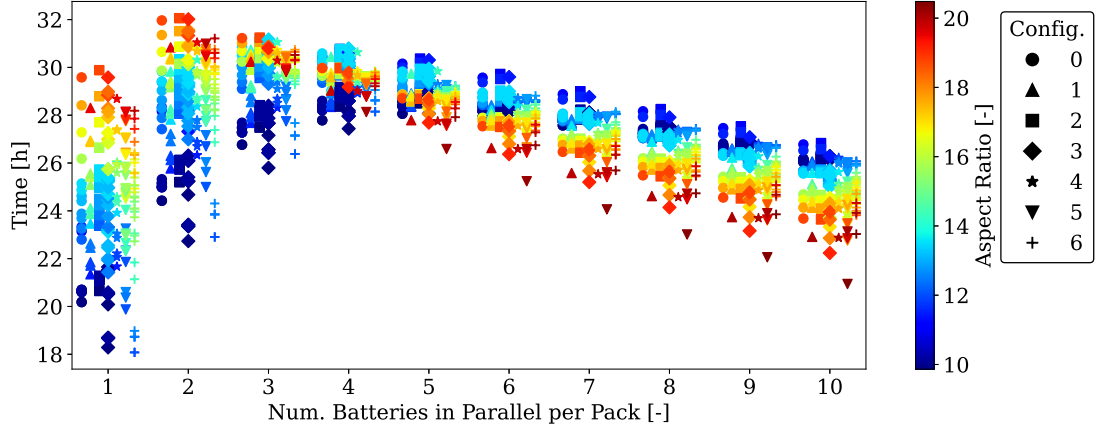


Figure F.22: Flight time as a function of n_{batt} , coloured by $\mathcal{R}$, identified by Configuration for no payload and $b_{PV1} = 15$.

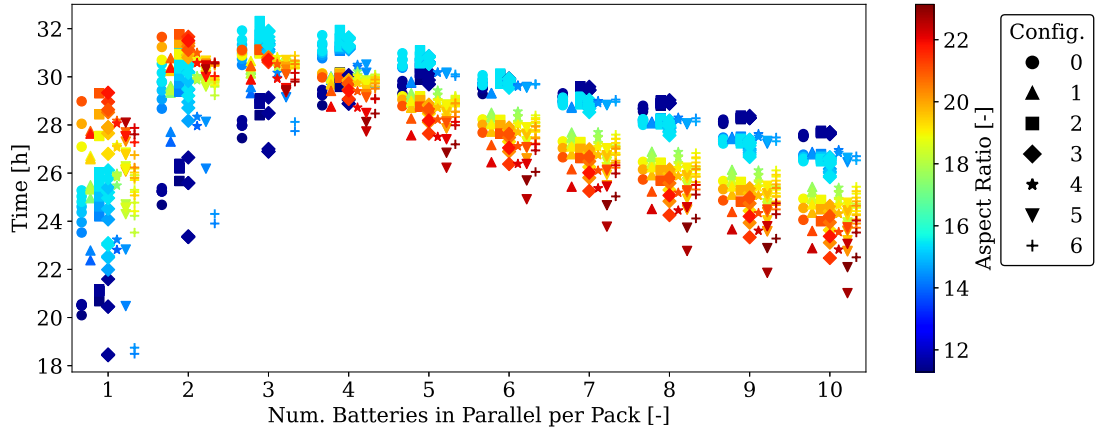


Figure F.23: Flight time as a function of n_{batt} , coloured by $\mathcal{R}$, identified by Configuration for no payload and $b_{PV1} = 17$.

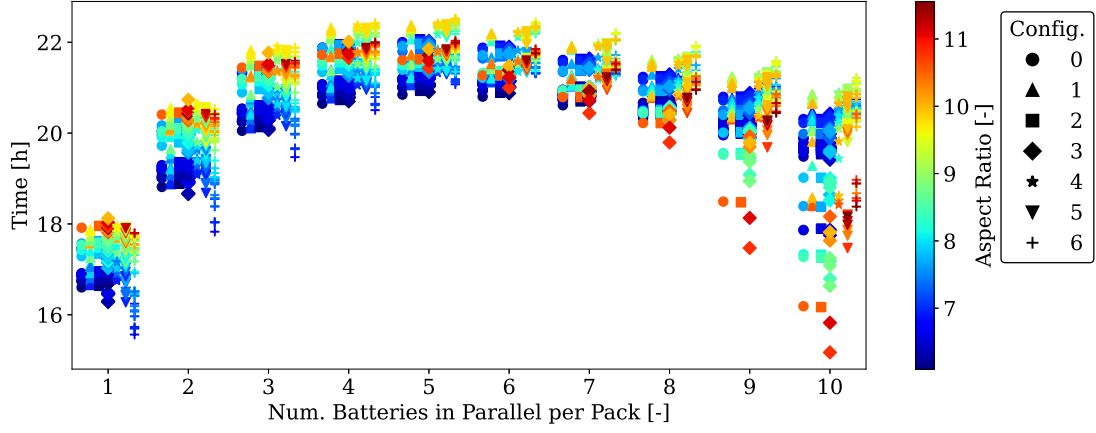


Figure F.24: Flight time as a function of n_{batt} , coloured by $\mathcal{R}$, identified by Configuration for payload option 2 and $b_{PV1} = 10$.

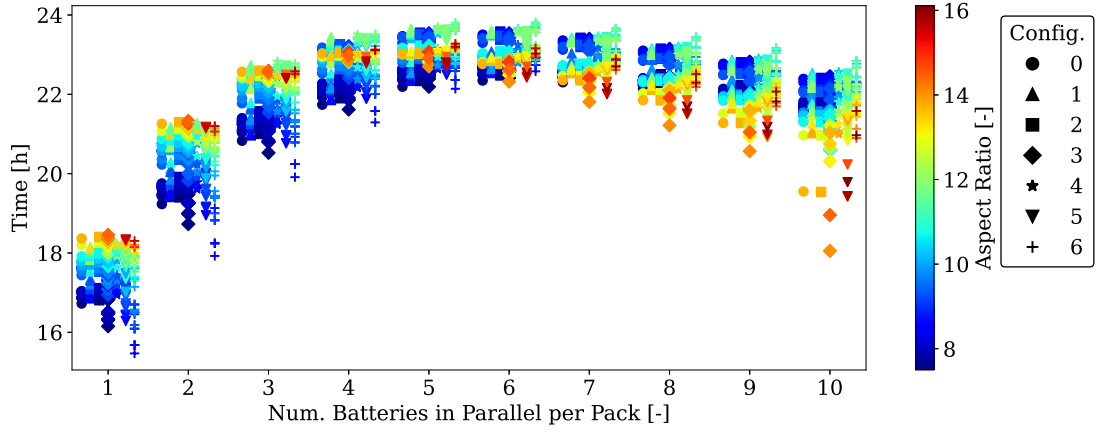


Figure F.25: Flight time as a function of n_{batt} , coloured by $\mathcal{R}$, identified by Configuration for payload option 2 and $b_{PV1} = 12$.

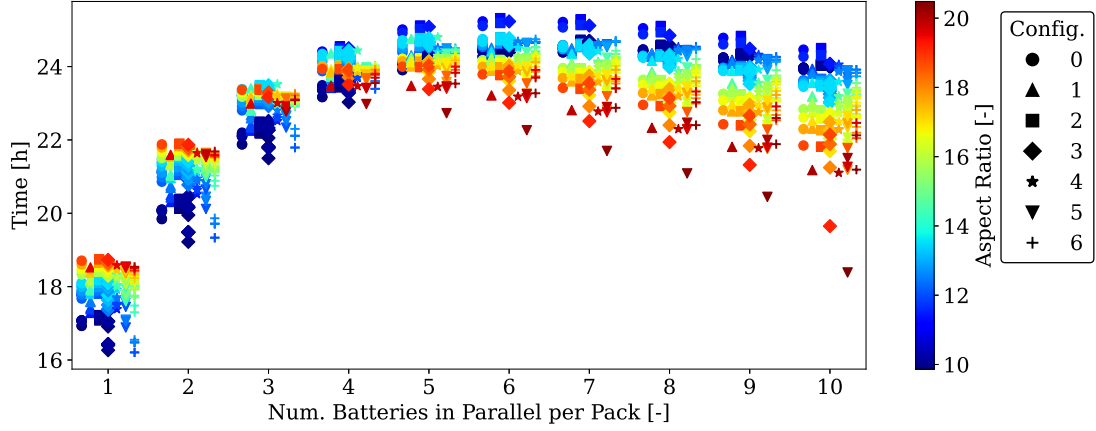


Figure F.26: Flight time as a function of n_{batt} , coloured by $\mathcal{R}$, identified by Configuration for payload option 2 and $b_{PV1} = 15$.

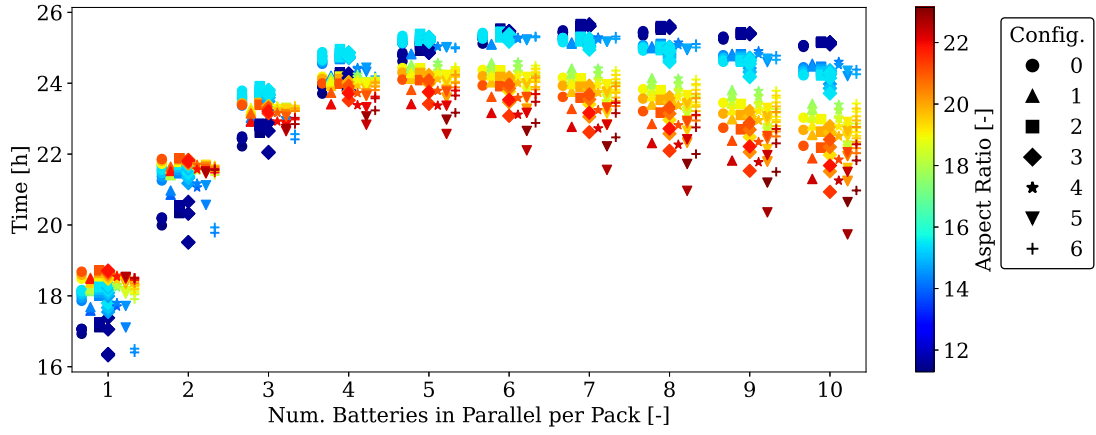


Figure F.27: Flight time as a function of n_{batt} , coloured by $\mathcal{R}$, identified by Configuration for payload option 2 and $b_{PV1} = 17$.

Figures F.28 through F.34 show consistent trends at different b_{PV1} in support of the analysis of Figure 4.17 in Section 4.5 with and without payload. Note that the scales are not constant in the y -axis.

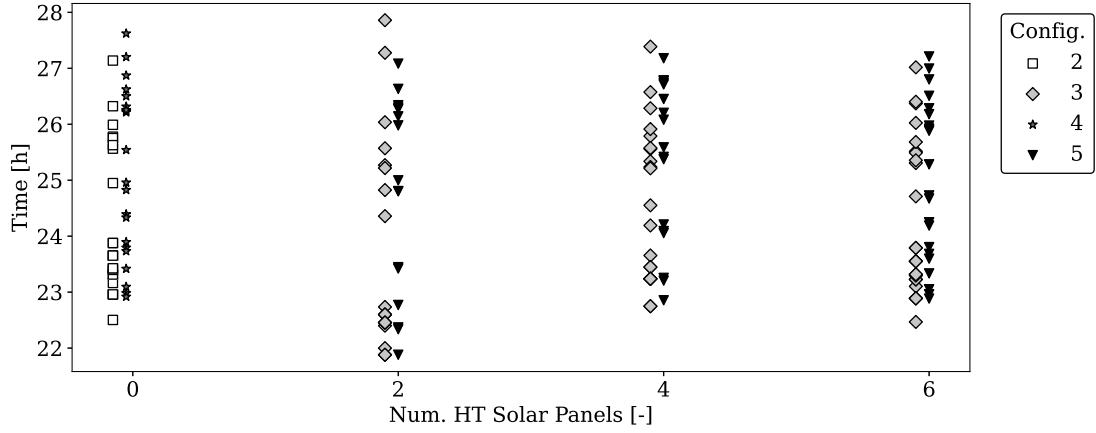


Figure F.28: Flight time as a function of the number of solar cells on the horizontal stabilizer for Configurations 2, 3, 4, and 5, where $b_{PV1} = 10$ and $n_{batt} = 2$, without payload.

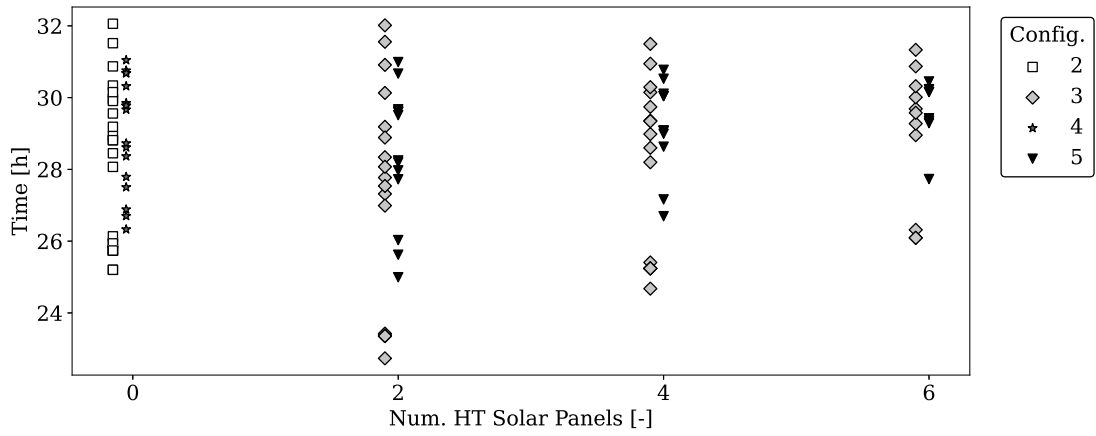


Figure F.29: Flight time as a function of the number of solar cells on the horizontal stabilizer for Configurations 2, 3, 4, and 5, where $b_{PV1} = 15$ and $n_{batt} = 2$, without payload.

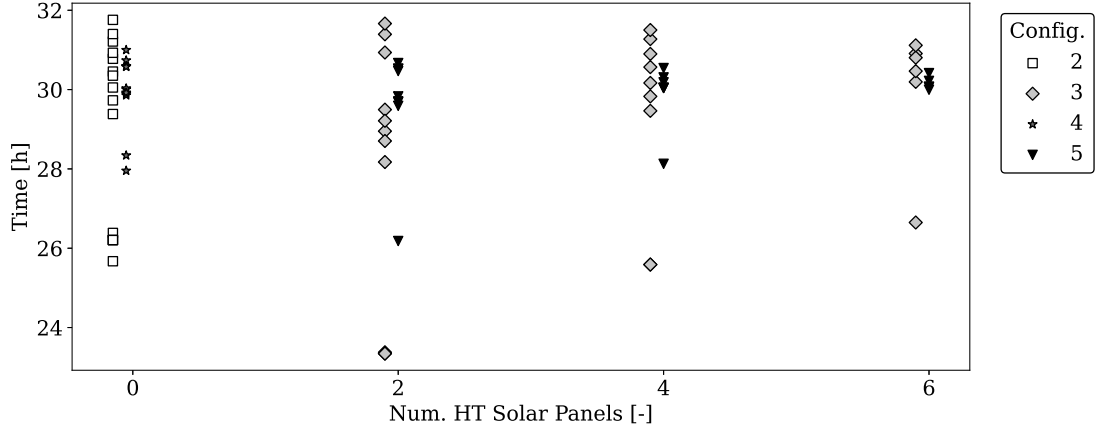


Figure F.30: Flight time as a function of the number of solar cells on the horizontal stabilizer for Configurations 2, 3, 4, and 5, where $b_{PV1} = 17$ and $n_{batt} = 2$, without payload.

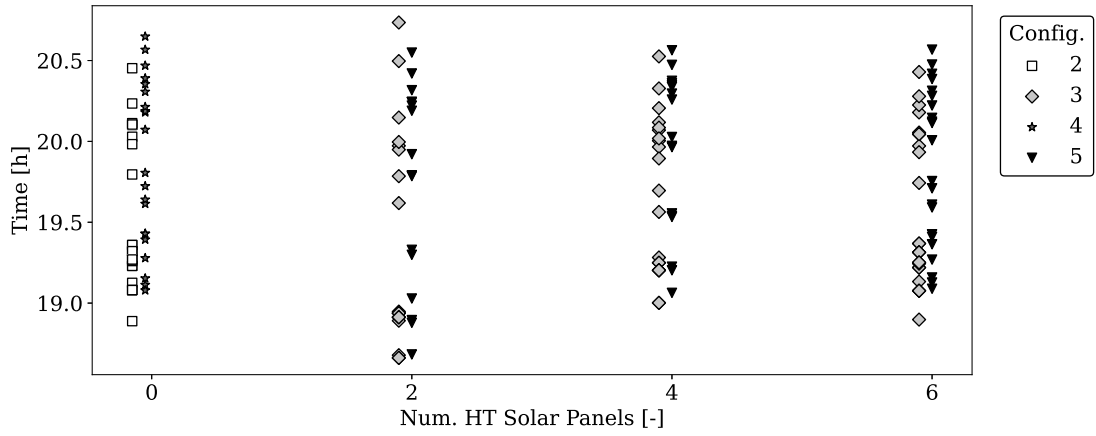


Figure F.31: Flight time as a function of the number of solar cells on the horizontal stabilizer for Configurations 2, 3, 4, and 5, where $b_{PV1} = 10$ and $n_{batt} = 2$, with payload option 2.

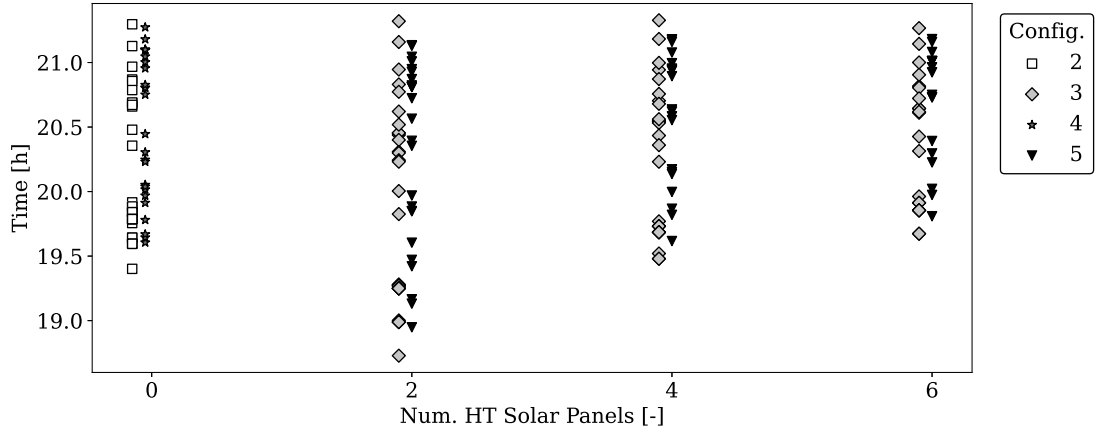


Figure F.32: Flight time as a function of the number of solar cells on the horizontal stabilizer for Configurations 2, 3, 4, and 5, where $b_{PV1} = 12$ and $n_{batt} = 2$, with payload option 2.

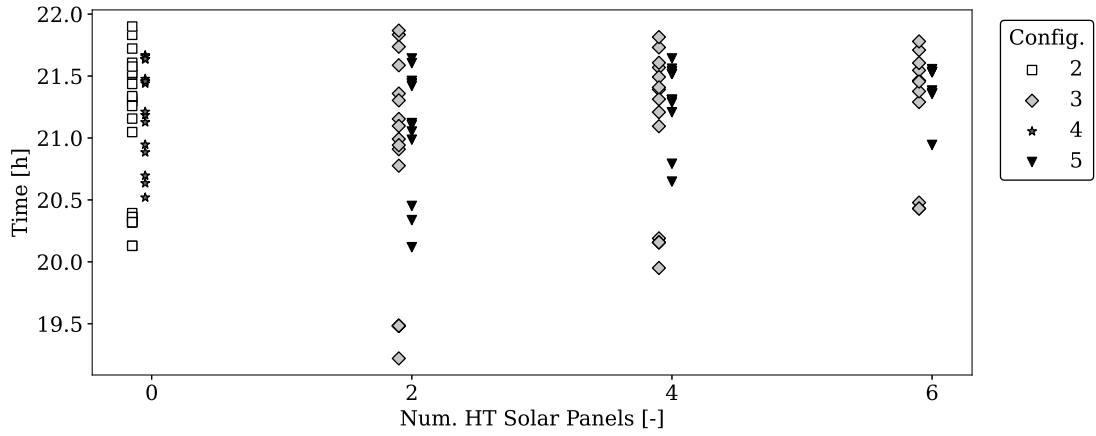


Figure F.33: Flight time as a function of the number of solar cells on the horizontal stabilizer for Configurations 2, 3, 4, and 5, where $b_{PV1} = 15$ and $n_{batt} = 2$, with payload option 2.

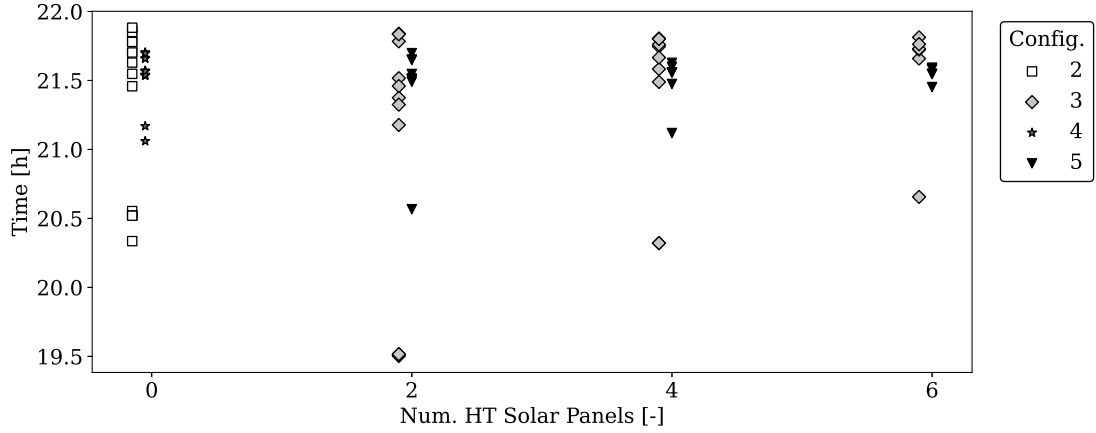


Figure F.34: Flight time as a function of the number of solar cells on the horizontal stabilizer for Configurations 2, 3, 4, and 5, where $b_{PV1} = 17$ and $n_{batt} = 2$, with payload option 2.

Figures F.35 and F.36 shows the data for all months, filling in the gaps in Figures 4.18 and 4.19. Figures F.37 and F.38 show the same trends with payload option 2.

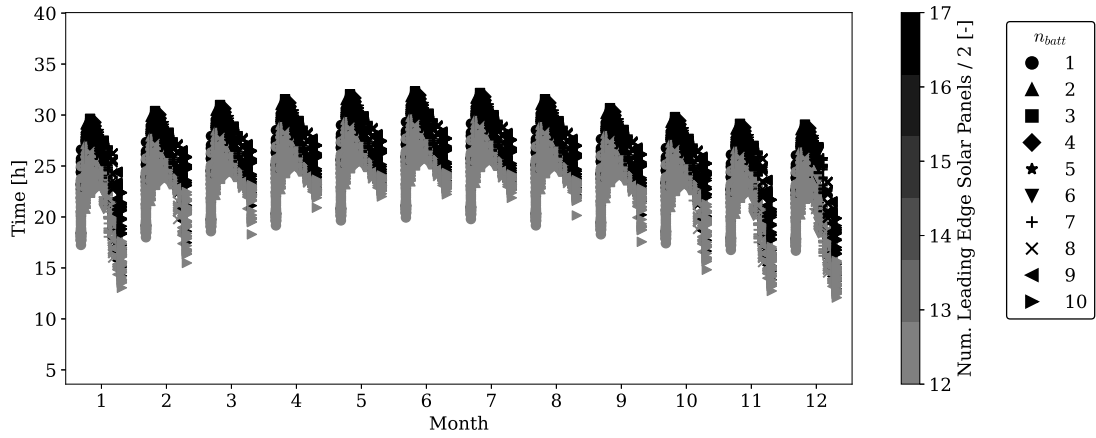


Figure F.35: Flight time as a function of time of year, identified by n_{batt} for zero payload in Kingston, ON.

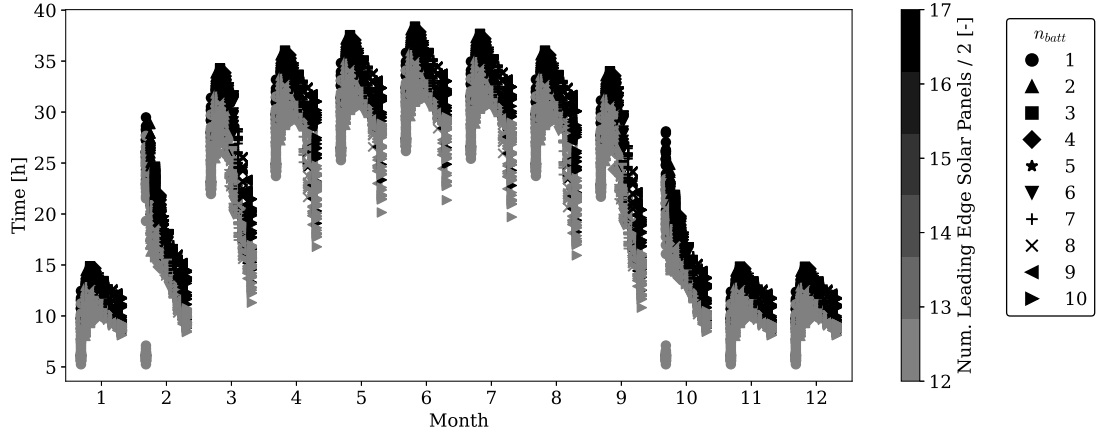


Figure F.36: Flight time as a function of time of year, identified by n_{batt} for zero payload in Inuvik, NT.

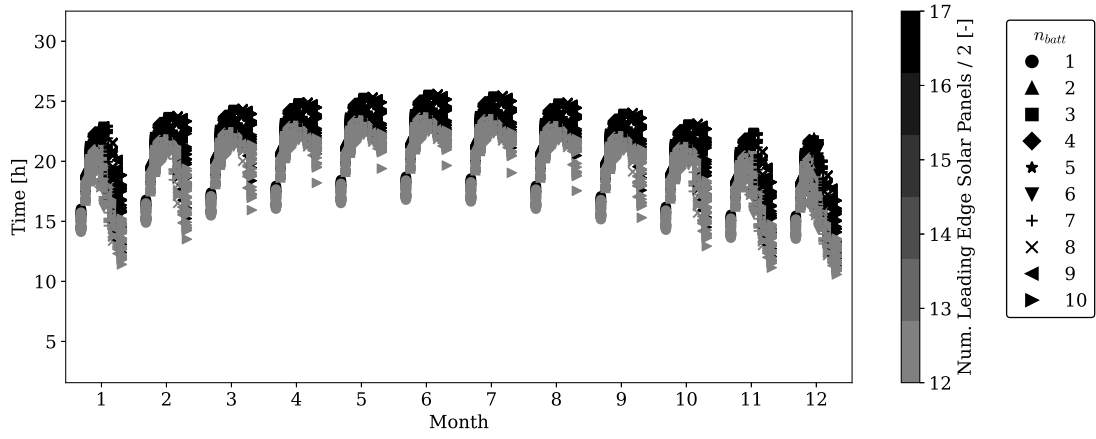


Figure F.37: Flight time as a function of time of year, identified by n_{batt} for payload option 2 in Kingston, ON.

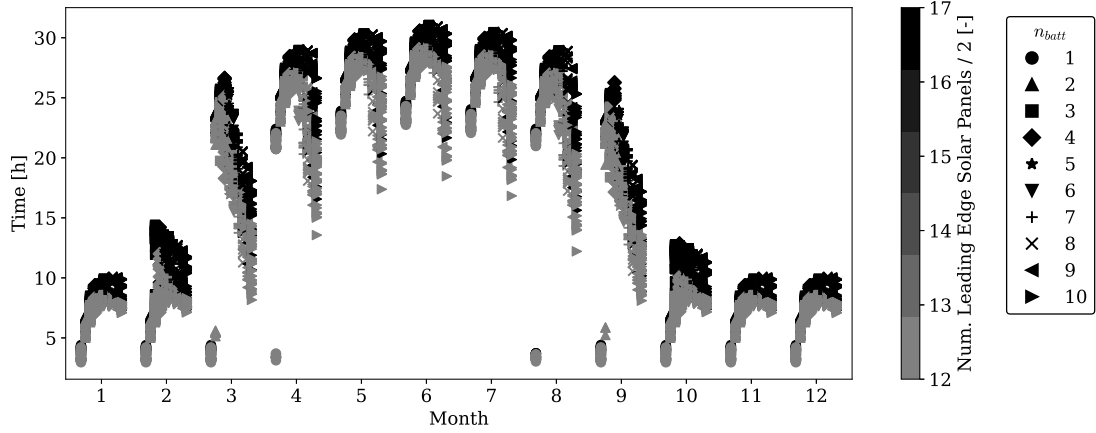


Figure F.38: Flight time as a function of time of year, identified by n_{batt} for payload option 2 in Inuvik, NT.

Figures F.39 through F.41 shows the insensitivity of the charge margin's preference to a clean wing or nacelle with variations in payload and latitude in support of the analysis with Figure 4.23.

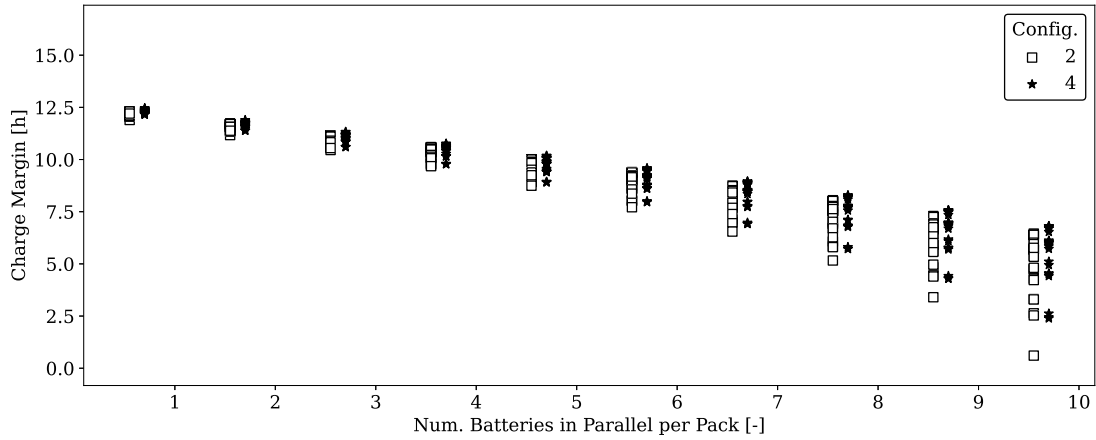


Figure F.39: Charge margin as a function of n_{batt} , identified by Configuration for zero payload, $b_{PV1} = 12$, month 6, flown in Kingston.

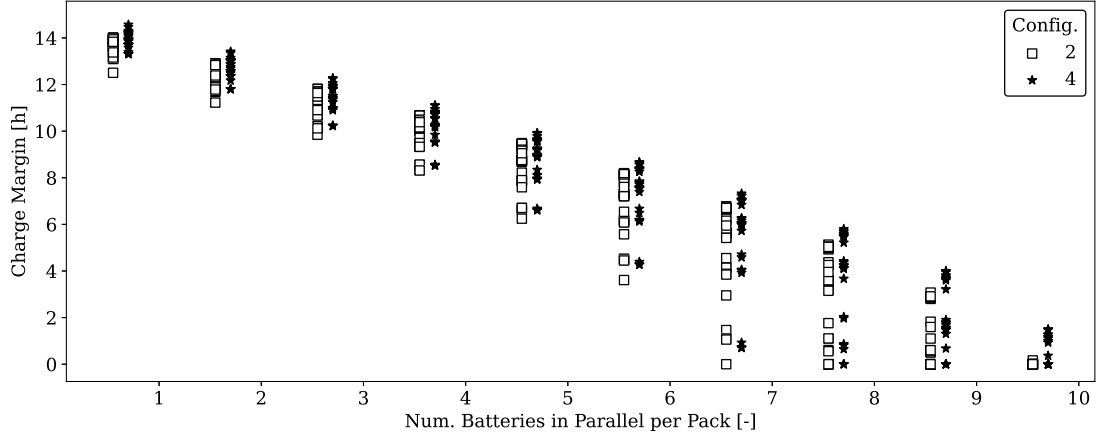


Figure F.40: Charge margin as a function of n_{batt} , identified by Configuration for payload option 2, $b_{PV1} = 12$, month 6, flown in Inuvik.

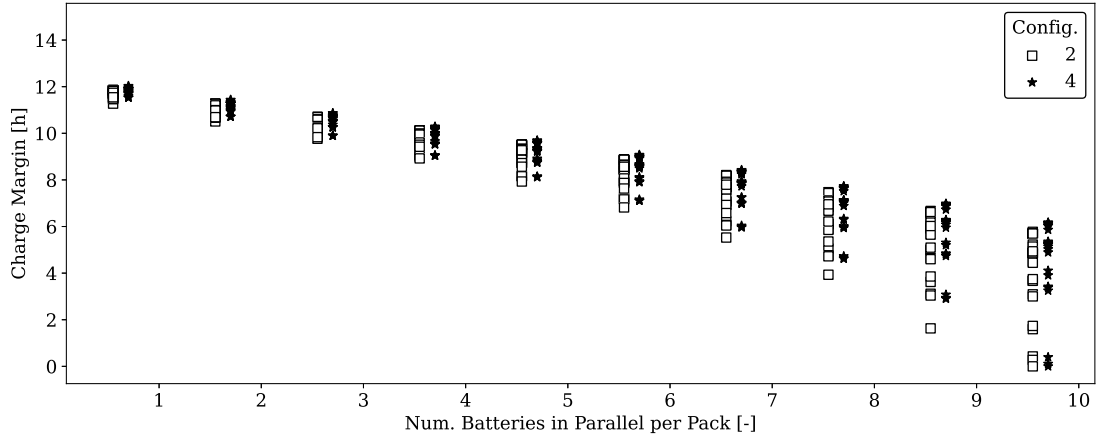


Figure F.41: Charge margin as a function of n_{batt} , identified by Configuration for payload option 2, $b_{PV1} = 12$, month 6, flown in Kingston.

Figures F.42 through F.48 show flight time as a function of n_{batt} , coloured by maximum aerodynamic efficiency, with markers indicating n_{PV} for zero payload at different months, first in Inuvik then in Kingston in support of the analysis of Figure 4.26.

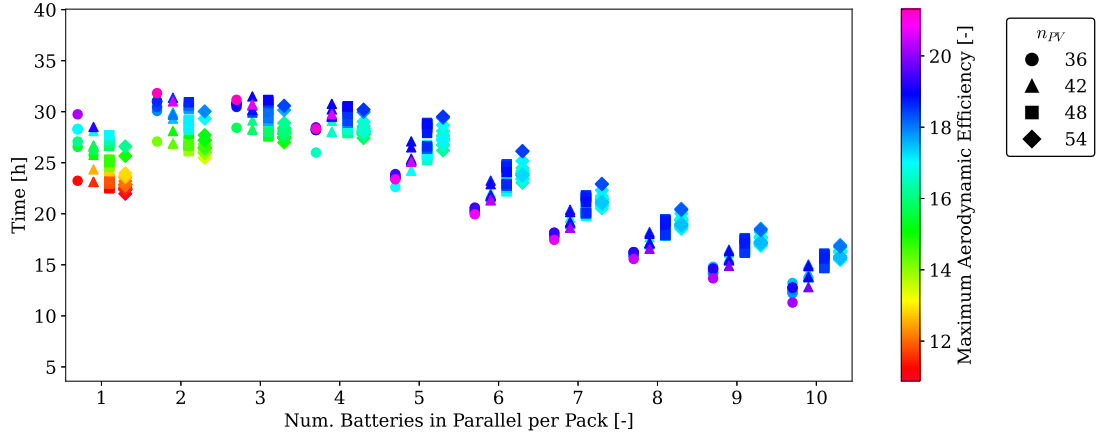


Figure F.42: Flight time as a function of n_{batt} , identified by n_{PV} for zero payload at $b_{PV1} = 12$, month 3, flown in Inuvik.

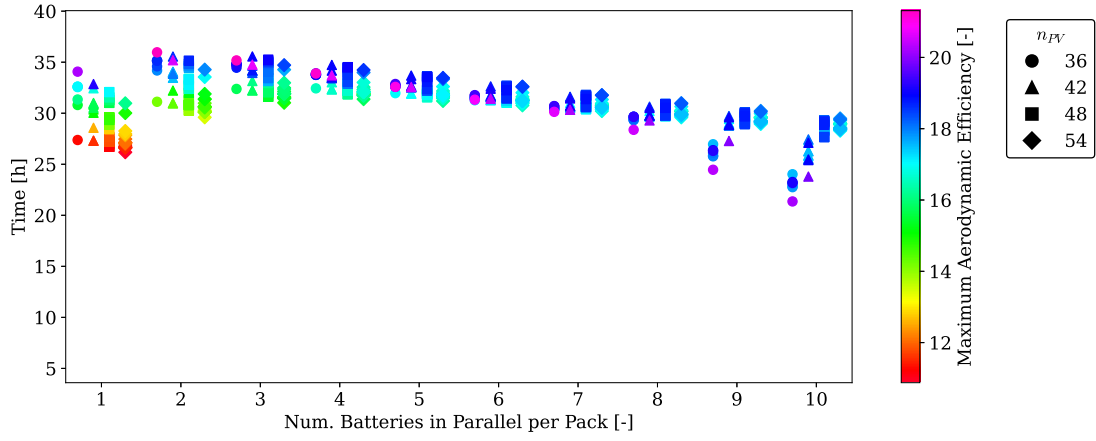


Figure F.43: Flight time as a function of n_{batt} , identified by n_{PV} for zero payload at $b_{PV1} = 12$, month 6, flown in Inuvik.

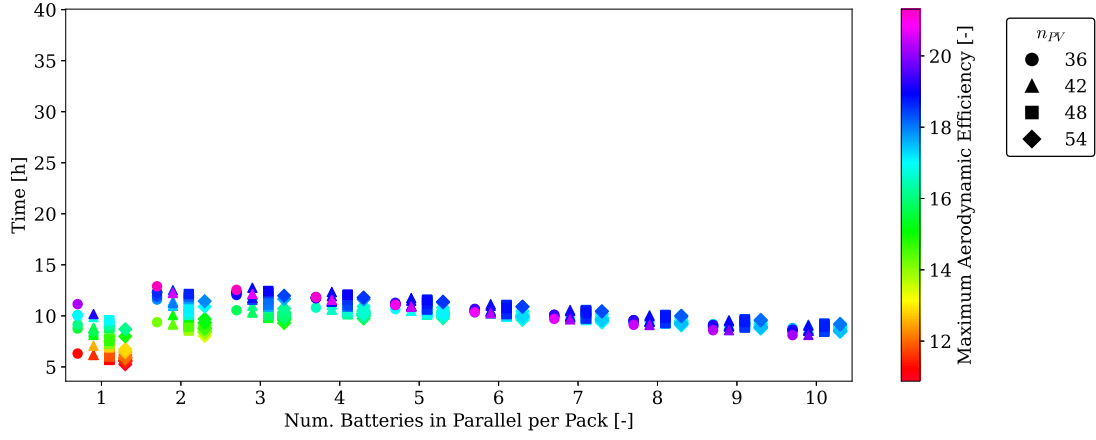


Figure F.44: Flight time as a function of n_{batt} , identified by n_{PV} for zero payload at $b_{PV1} = 12$, month 12, flown in Inuvik.

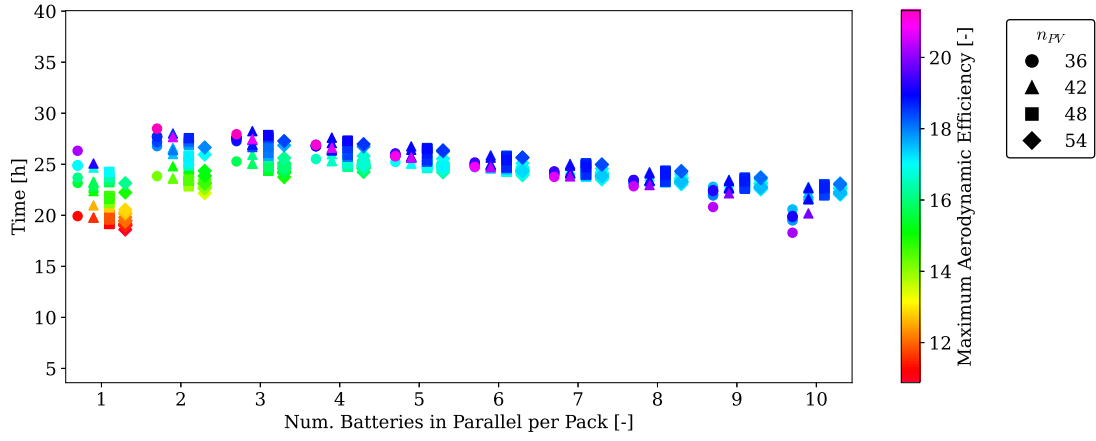


Figure F.45: Flight time as a function of n_{batt} , identified by n_{PV} for zero payload at $b_{PV1} = 12$, month 3, flown in Kingston.

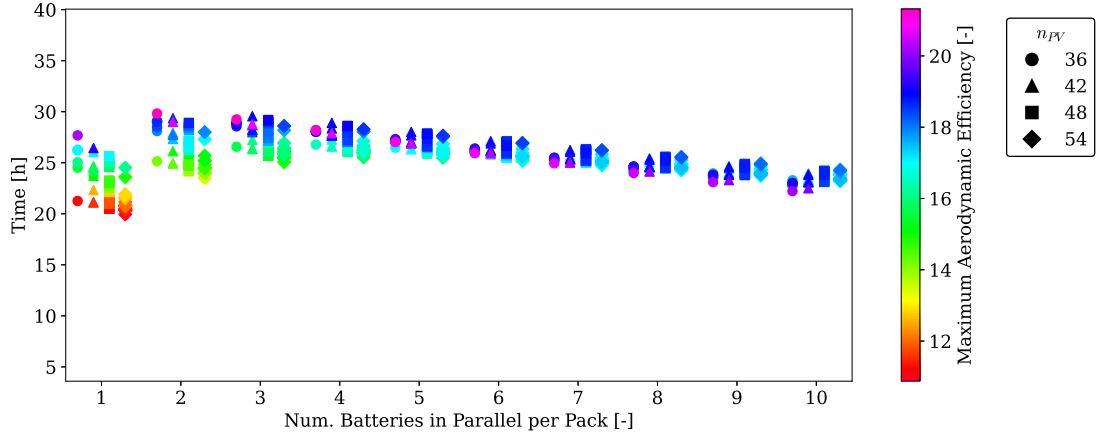


Figure F.46: Flight time as a function of n_{batt} , identified by n_{PV} for zero payload at $b_{PV1} = 12$, month 6, flown in Kingston.

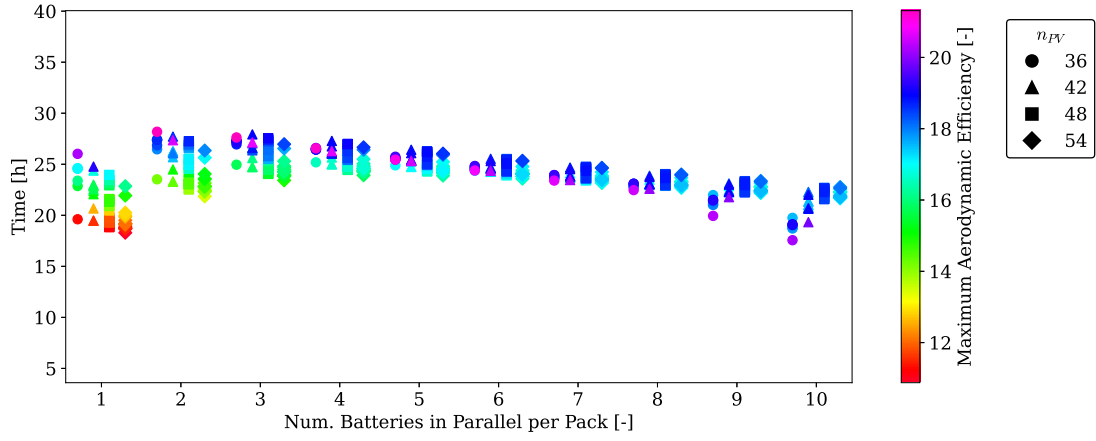


Figure F.47: Flight time as a function of n_{batt} , identified by n_{PV} for zero payload at $b_{PV1} = 12$, month 9, flown in Kingston.

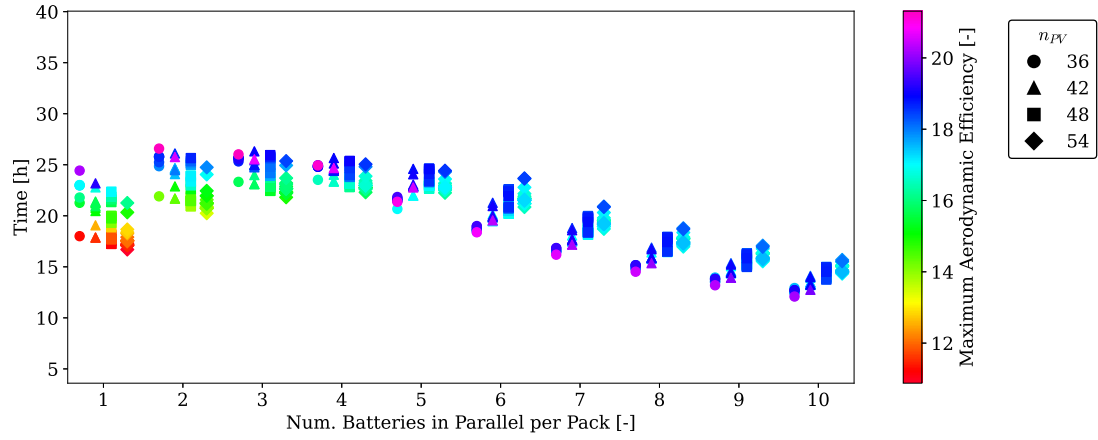


Figure F.48: Flight time as a function of n_{batt} , identified by n_{PV} for zero payload at $b_{PV1} = 12$, month 12, flown in Kingston.

G Extended Solar UAV Optimization Results

This appendix contains extended results from the UAV optimization in Chapter 5. The sections mimic the structure of Chapter 5.

G.1 Initial Results

G.1.1 Clean Configuration with Payload

Figure G.1 shows the constraints for the clean configuration with payload optimization.

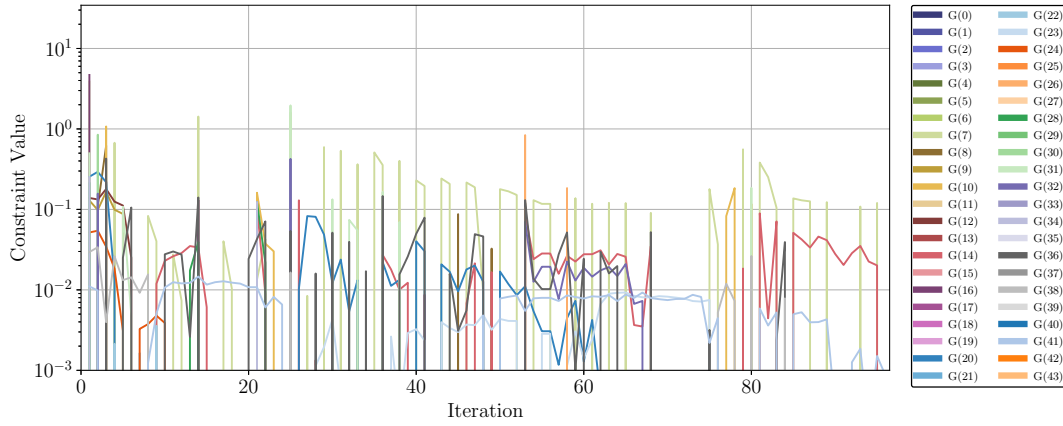


Figure G.1: Constraints not meeting the tolerance threshold and their values for the clean configuration with payload.

G.1.2 Nacelles without Payload

Figures G.2 and G.3 show the constraints and scatter plots for all 250 iterations for the nacelle configuration with no payload. These complement and complete the dataset from Figure 5.18.

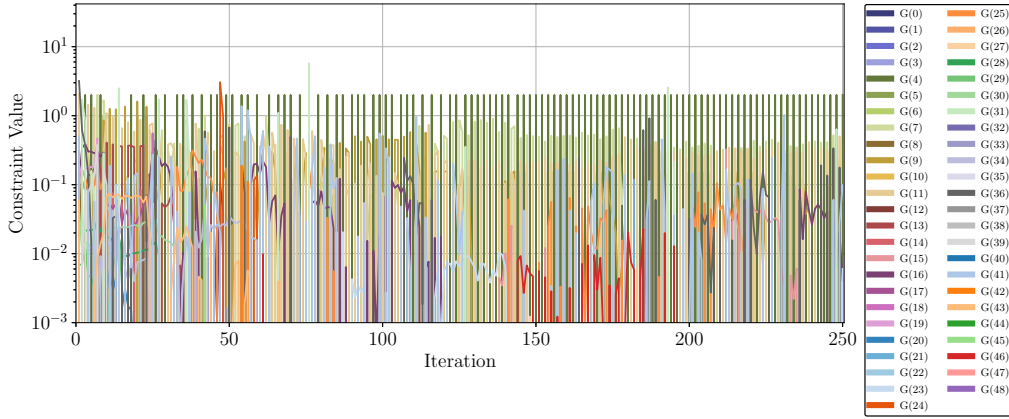


Figure G.2: Constraints not meeting the tolerance threshold and their values for the nacelle configuration, no payload.

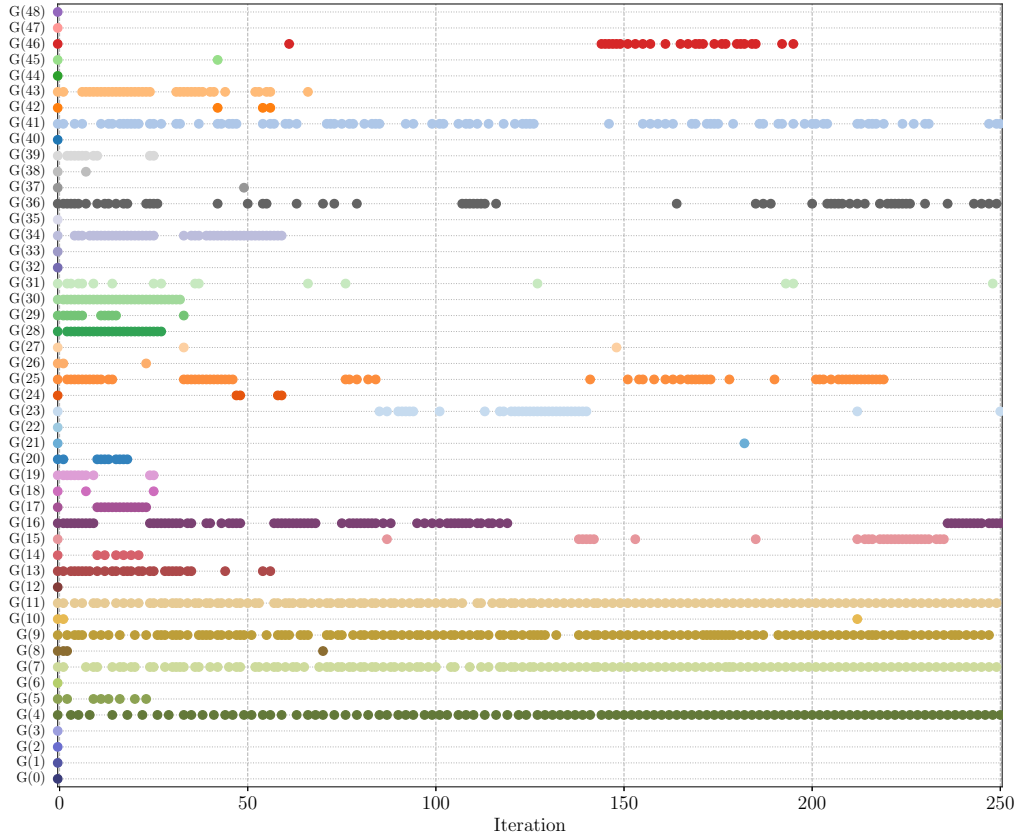


Figure G.3: Scatter of constraints not meeting the tolerance threshold for the nacelle configuration, no payload.

Figure G.4 shows the parallel lines plot for all 250 iterations of the optimization of the nacelle configuration without payload.

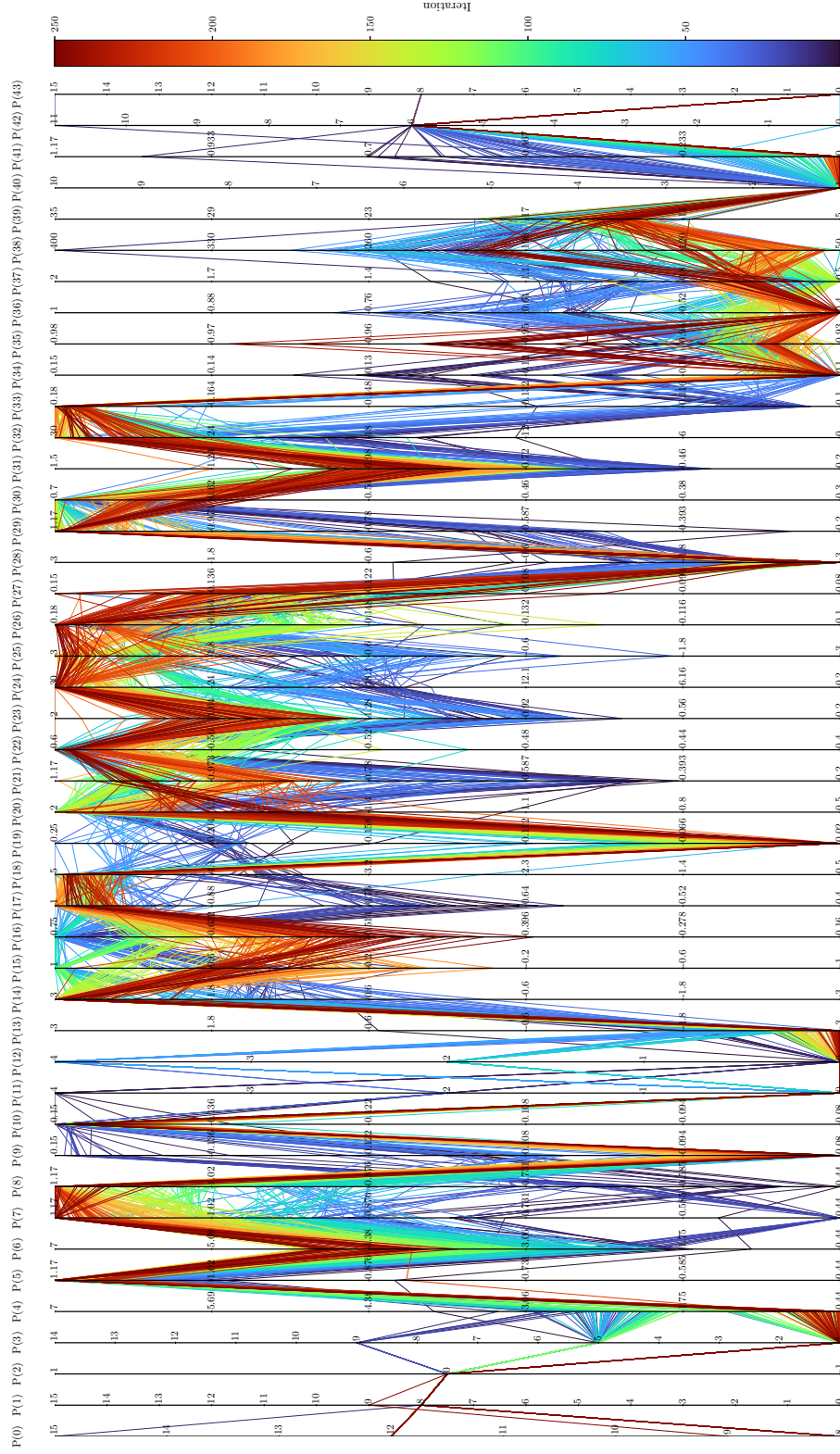


Figure G.4: Design variables for the nacelle configuration, no payload.

G.1.3 Nacelles with Payload

Figures G.5 and G.6 show the constraints and scatter plots for all 250 iterations of the nacelle configuration with payload. These complete and complement the dataset from Figure 5.23.

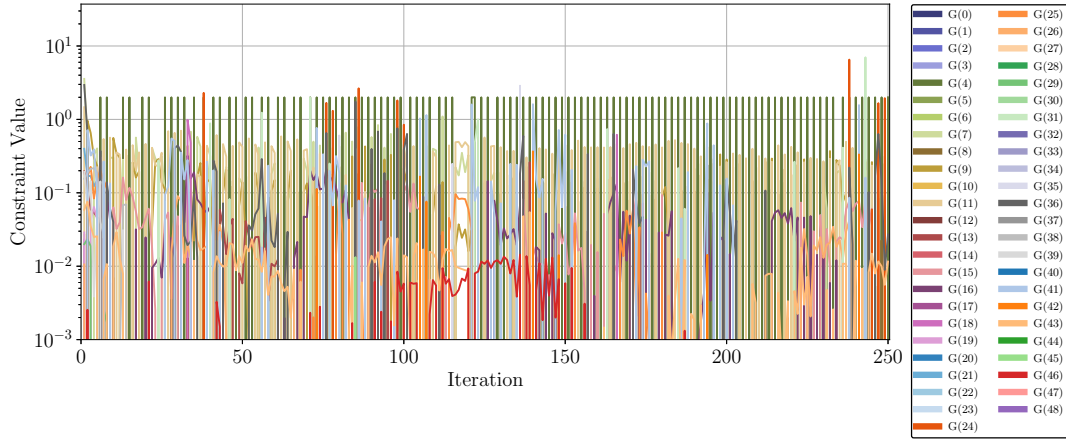


Figure G.5: Constraints not meeting the tolerance threshold and their values for the nacelle configuration with payload.

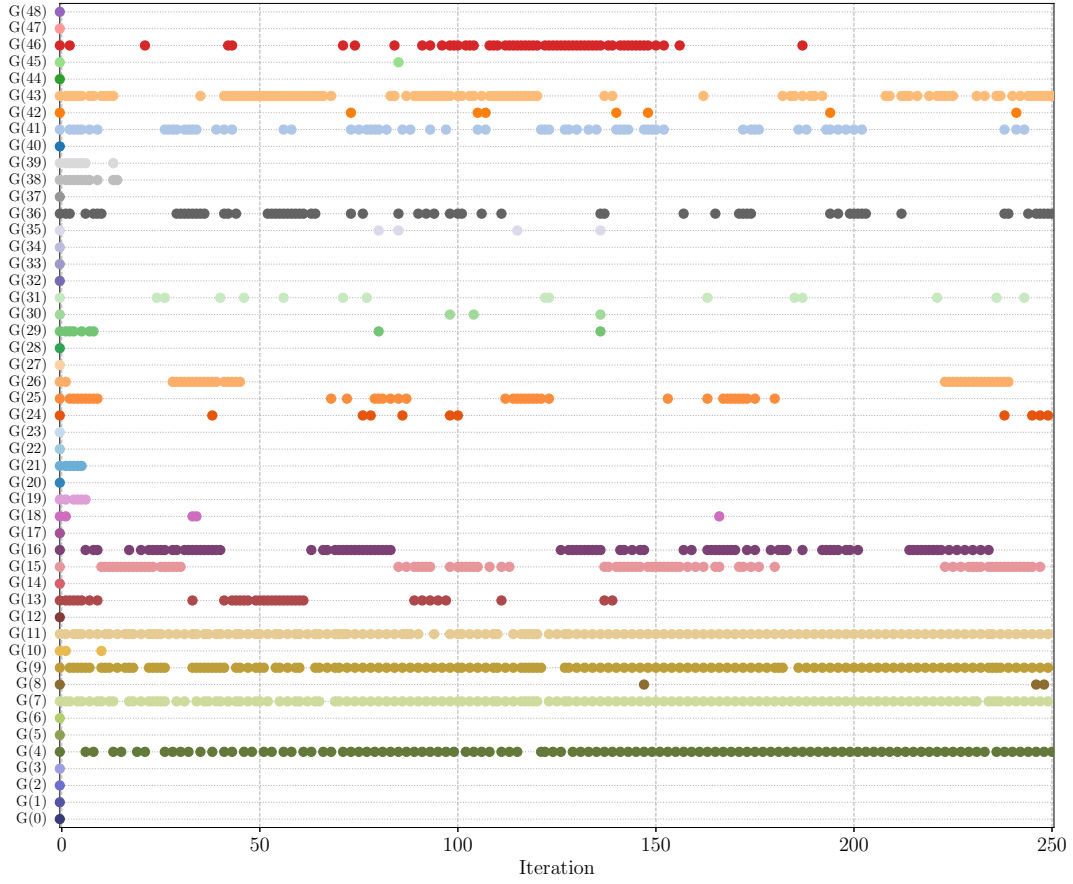


Figure G.6: Scatter of constraints not meeting the tolerance threshold for the nacelle configuration with payload.

Figure G.7 shows the parallel lines plot for all 250 iterations of the optimization of the nacelle configuration with payload.

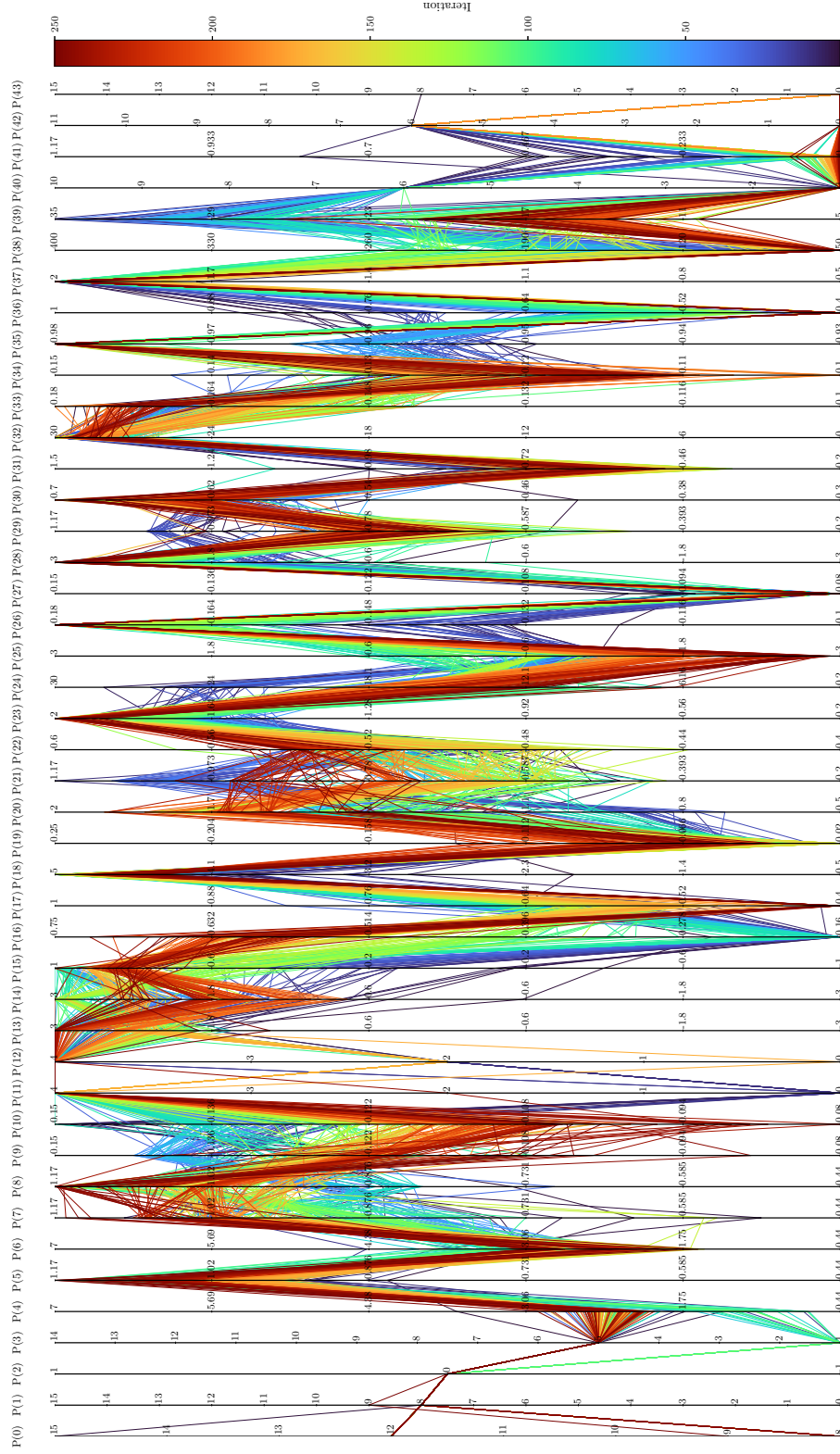


Figure G.7: Design variables for the nacelle configuration with payload.

G.2 Refined Results

G.2.1 Clean Configuration without Payload

Figure G.8 shows the values of the constraints not meeting the tolerance threshold for the refined optimization of the clean configuration with no payload.

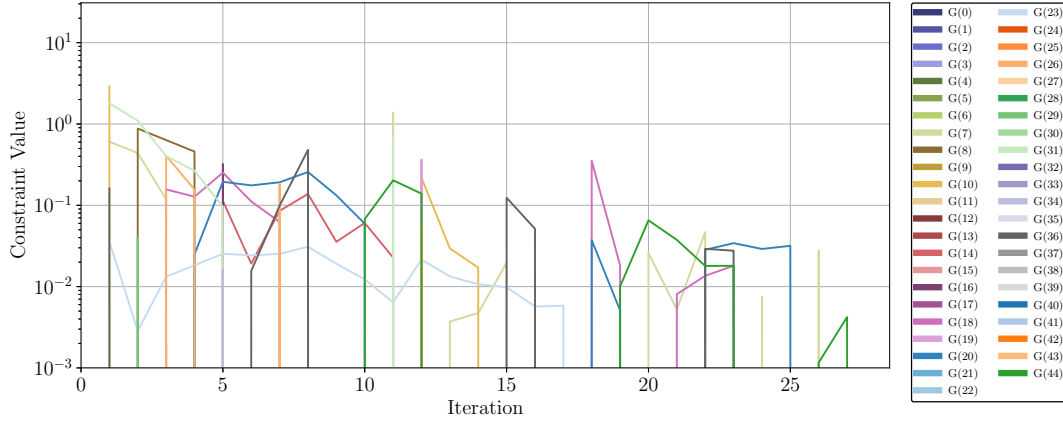


Figure G.8: Constraints not meeting the tolerance threshold and their values for the refined clean configuration without payload optimization.

Figure G.9 shows the parallel lines plot for the refined clean configuration without payload optimization.

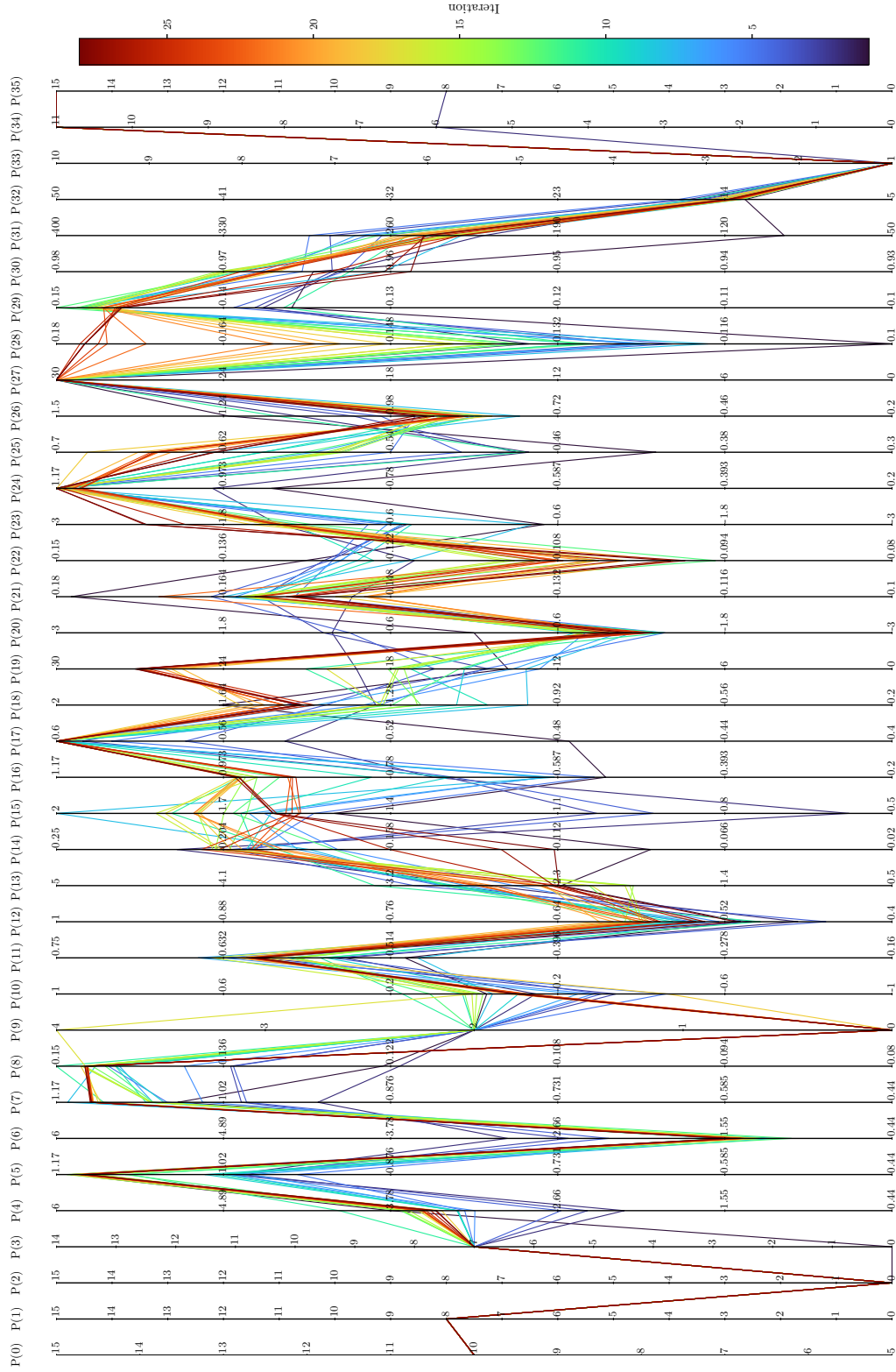


Figure G.9: Design variables for the refined clean configuration without payload optimization.

G.2.2 Clean Configuration with Payload

Figure G.10 shows the values of the constraints not meeting the tolerance threshold for the refined optimization of the clean configuration with payload.

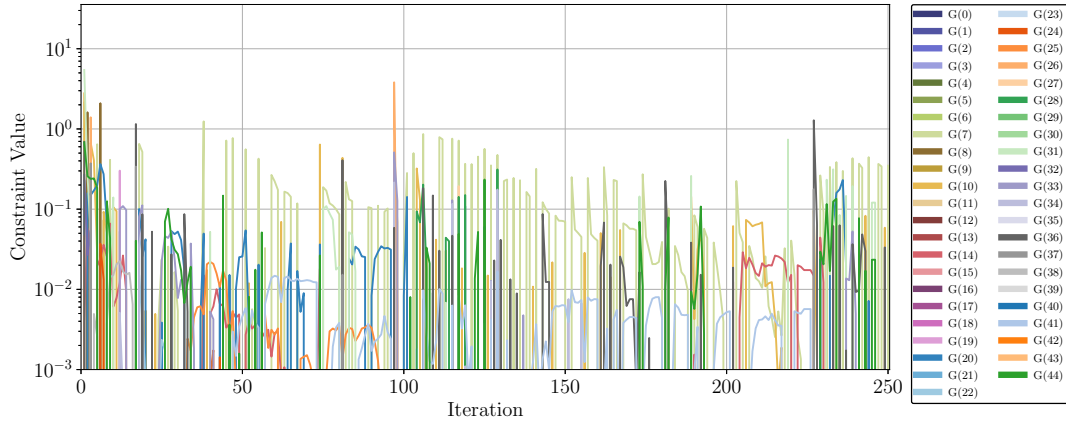


Figure G.10: Constraints not meeting the tolerance threshold and their values for the refined clean configuration with payload optimization.

Figure G.11 shows the parallel lines plot for the refined clean configuration with payload optimization.

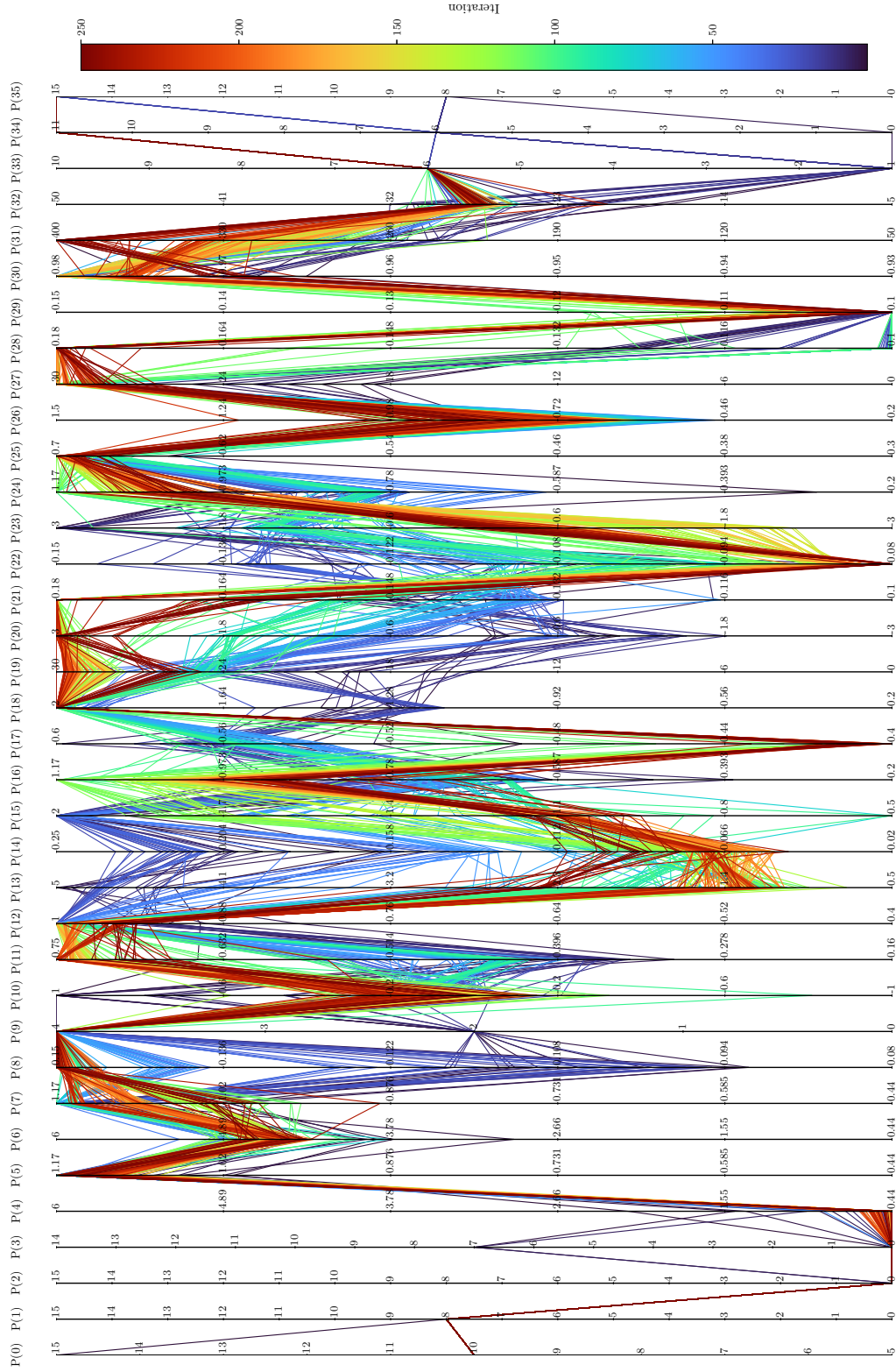


Figure G.11: Design variables for the refined clean configuration with payload optimization.

G.2.3 Nacelles without Payload

Figure G.12 shows the value of the constraints not meeting the tolerance threshold for the refined optimization of the nacelle configuration with no payload.

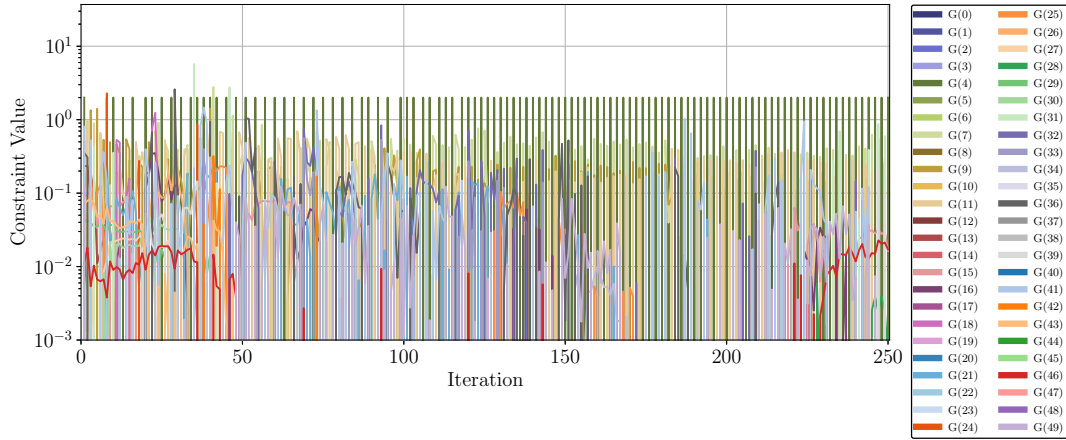


Figure G.12: Constraints not meeting the tolerance threshold and their values for the refined nacelle configuration without payload optimization.

Figure G.13 shows the parallel lines plot for the refined nacelle configuration without payload optimization.

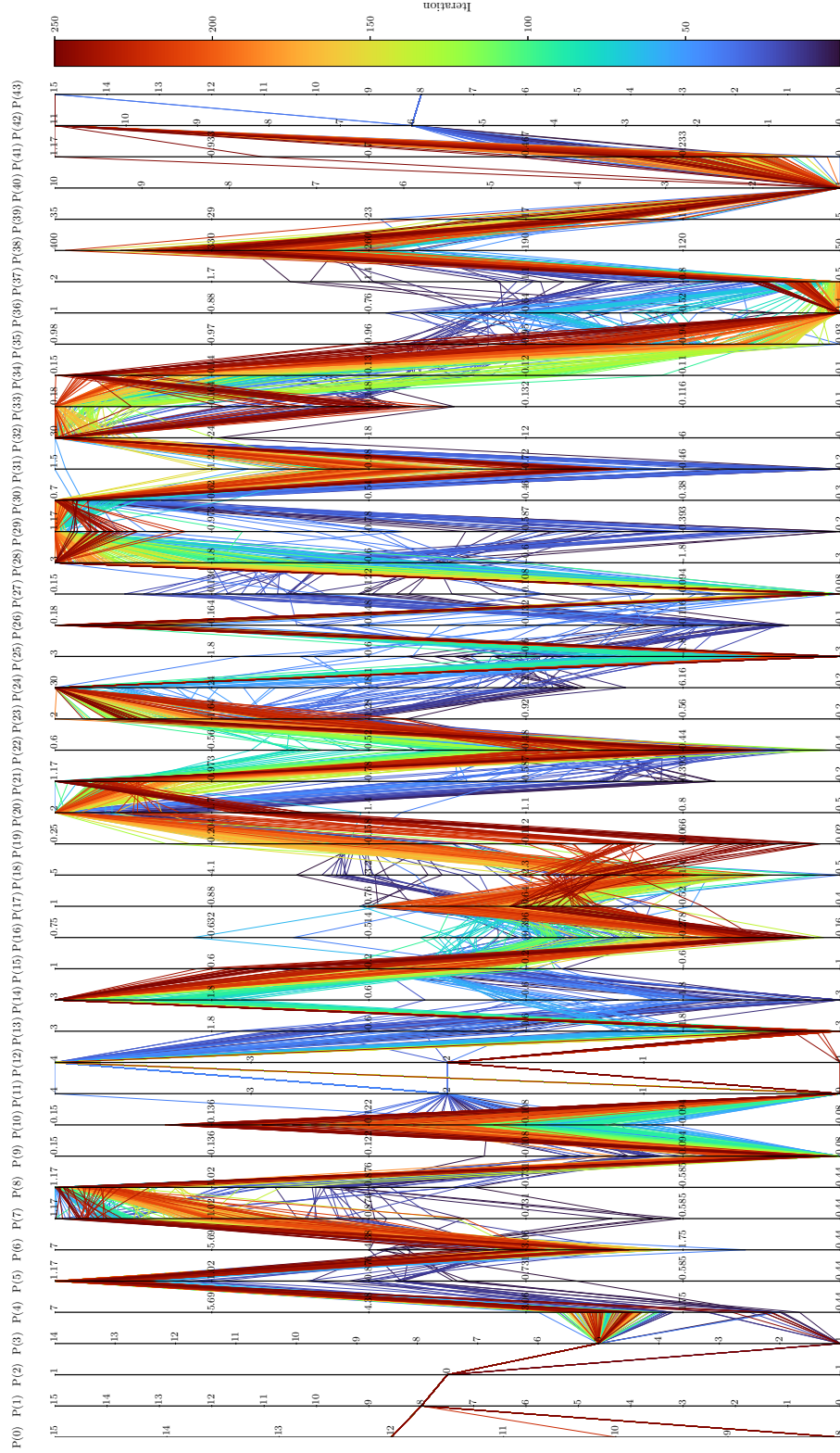


Figure G.13: Design variables for the refined nacelle configuration without payload optimization.

G.2.4 Nacelles with Payload

Figure G.14 shows the value of the constraints not meeting the tolerance threshold for the refined optimization of the nacelle configuration with payload.

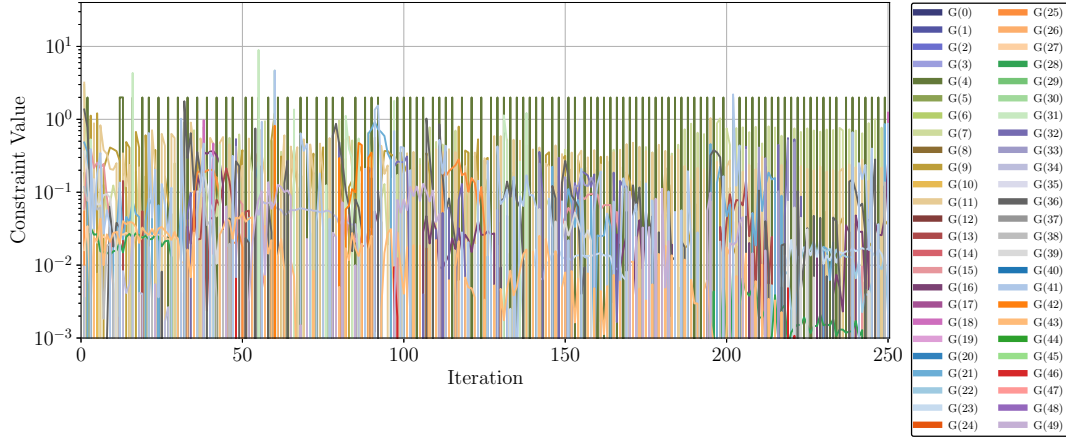


Figure G.14: Constraints not meeting the tolerance threshold and their values for the refined nacelle configuration with payload optimization.

Figure G.15 shows the parallel lines plot for the refined nacelle configuration with payload optimization.

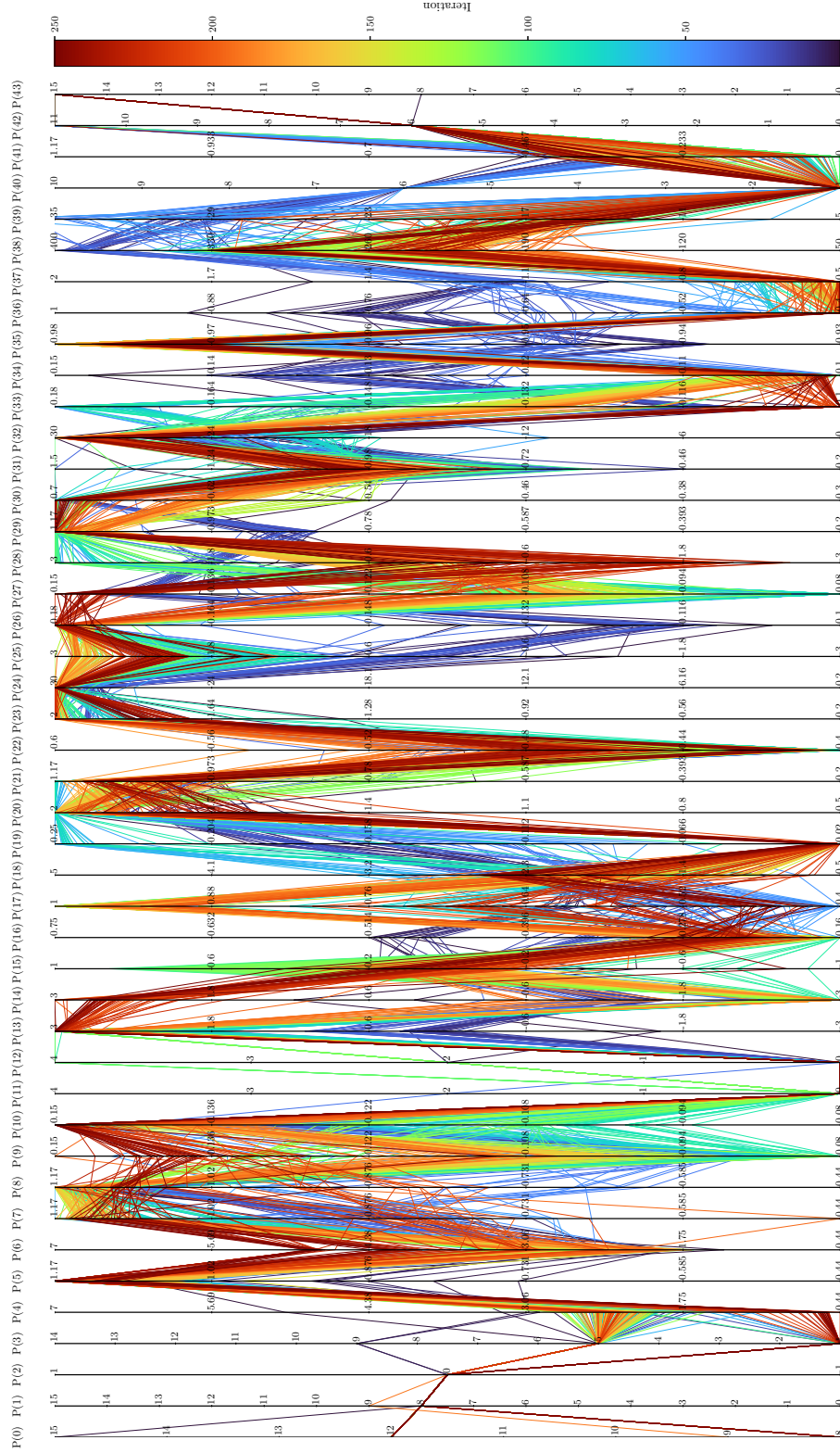


Figure G.15: Design variables for the refined nacelle configuration with payload optimization.

G.2.5 Proposed Geometry

The individual design variables for the proposed geometry are summarized in Table G.1. The listed airspeeds and numbers of battery cells in parallel correspond to the three payload combinations outlined in Table 3.5 where the lowest power draw corresponds to the slowest airspeed and least number of battery cells in parallel, with all parameters increasing with increasing power draw.

Table G.1: Variables for the proposed clean configuration geometry.

Label	Design Variable Name	a_i
P(0)	b_{PV1} [-]	11
P(1)	b_{PV2} [-]	9
P(2)	b_{PV3} [-]	0
P(3)	b_{PV_i} [-]	0 5
P(4)	b_i [ft]	2.20
P(5)	c_i [ft]	1.12
P(6)	b_o [ft]	2.80
P(7)	c_t [ft]	1.12
P(8)	Wing thickness to chord ratio [-]	0.147
P(9)	Wing airfoil ID [-]	0
P(10)	Wing x offset [ft]	0.183
P(11)	Fuselage nose section length [ft]	0.75
P(12)	Fuselage squareness factor [-]	1.0
P(13)	Fuselage tail length [ft]	1.5
P(14)	Aft fuselage radius [ft]	0.08
P(15)	Fuselage height to width ratio [-]	1.08
P(16)	Horizontal tail root chord [ft]	0.95
P(17)	Horizontal tail taper ratio [-]	0.4
P(18)	Horizontal tail semi-span [ft]	2
P(19)	Horizontal tail sweep [deg]	28
P(20)	Horizontal tail dihedral [deg]	3
P(21)	Horizontal tail root thickness to chord ratio [-]	0.18
P(22)	Horizontal tail tip thickness to chord ratio [-]	0.08
P(23)	Horizontal tail twist [deg]	-0.4
P(24)	Vertical tail root chord [ft]	1.03
P(25)	Vertical tail taper ratio [-]	0.7
P(26)	Vertical tail span [ft]	0.75
P(27)	Vertical tail sweep [deg]	30
P(28)	Vertical tail root thickness to chord ratio [-]	0.18
P(29)	Vertical tail tip thickness to chord ratio [-]	0.10
P(30)	Vertical tail TE position [% of fuselage length]	0.97
P(31)	Turn radius [ft]	337
P(32)	Cruising airspeed [$\frac{\text{ft}}{\text{s}}$]	15.2, 21.0, 27.0
P(33)	Number of batteries in parallel [-]	1, 3, 6
P(34)	Propeller ID [-]	11
P(35)	BLDC motor ID [-]	15

H Solar UAV Optimization with Relaxed Constraints

This appendix summarizes the results of the optimization with more relaxed constraints. The modified constraints for the clean configuration are summarized in Table H.1. The modified constraints for the nacelle configuration are summarized in Table H.2. As with the original optimization problem, each configuration was optimized without an upper constraint on the static margin. The results from that initial optimization are used as a starting point for a refined optimization with an upper constraint on the static margin. This appendix only presents the final results of the optimization including the upper static margin constraint.

Table H.1: Updated constraints for clean configuration optimization.

Label	Constraint	Value
G(7)	Wing span [ft]	≤ 12
G(31)	Vertical tail aspect ratio [-]	≤ 2.5
G(34)	Static margin [% MAC]	≥ 5.0
G(44)	Static margin [% MAC]	≤ 25

Table H.2: Updated constraints for nacelle configuration optimization.

Label	Constraint	Value
G(7)	Wing span [ft]	≤ 12
G(36)	Vertical tail aspect ratio [-]	≤ 2.5
G(39)	Static margin [% MAC]	≥ 5.0
G(49)	Static margin [% MAC]	≤ 25

H.1 Clean Configuration without Payload

This section outlines the results for the clean configuration optimization with no additional payload with the relaxed and refined constraints outlined in Table H.1. The optimization with the refined static margin constraints converged in 30 iterations with a fitness of 131.04 h. The fitness for each iteration is plotted in Figure H.1. The decrease in fitness is due to the optimizer finding a solution that satisfies more constraints, which

is preferred by the algorithm to a design with high fitness and several constraints outside of the tolerance threshold. The next iteration is a design with improved constraint values and fitness.

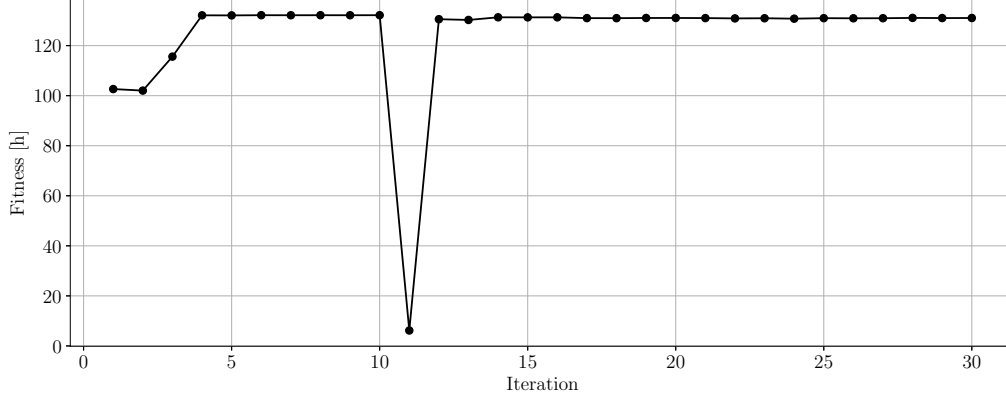


Figure H.1: Fitness convergence with relaxed constraints for the clean configuration, no payload.

The constraints not meeting the tolerance threshold are plotted in Figure H.2. The scatter plot of these constraints are shown in Figure H.3. The final geometry is shown in Figure H.4 in blue, with the final geometry from the refined constraints optimization shown in grey.

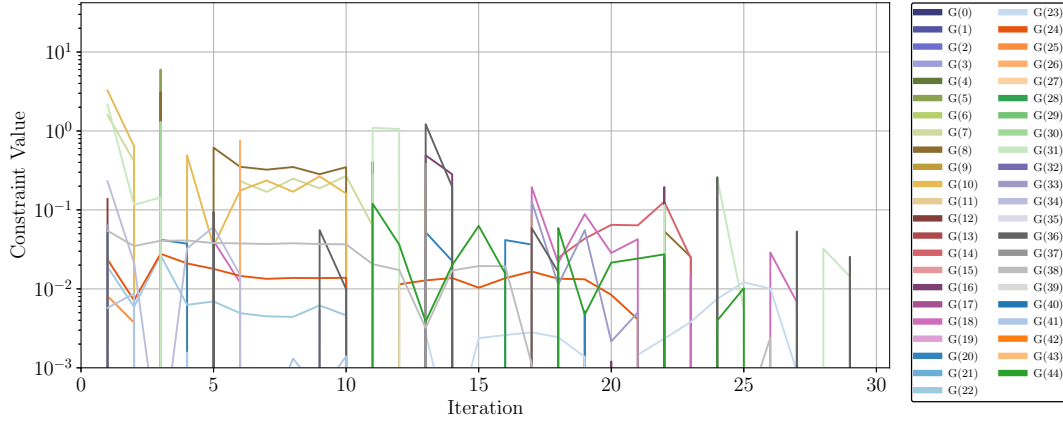


Figure H.2: Constraints violated and their values for the relaxed constraints for the clean configuration, no payload.

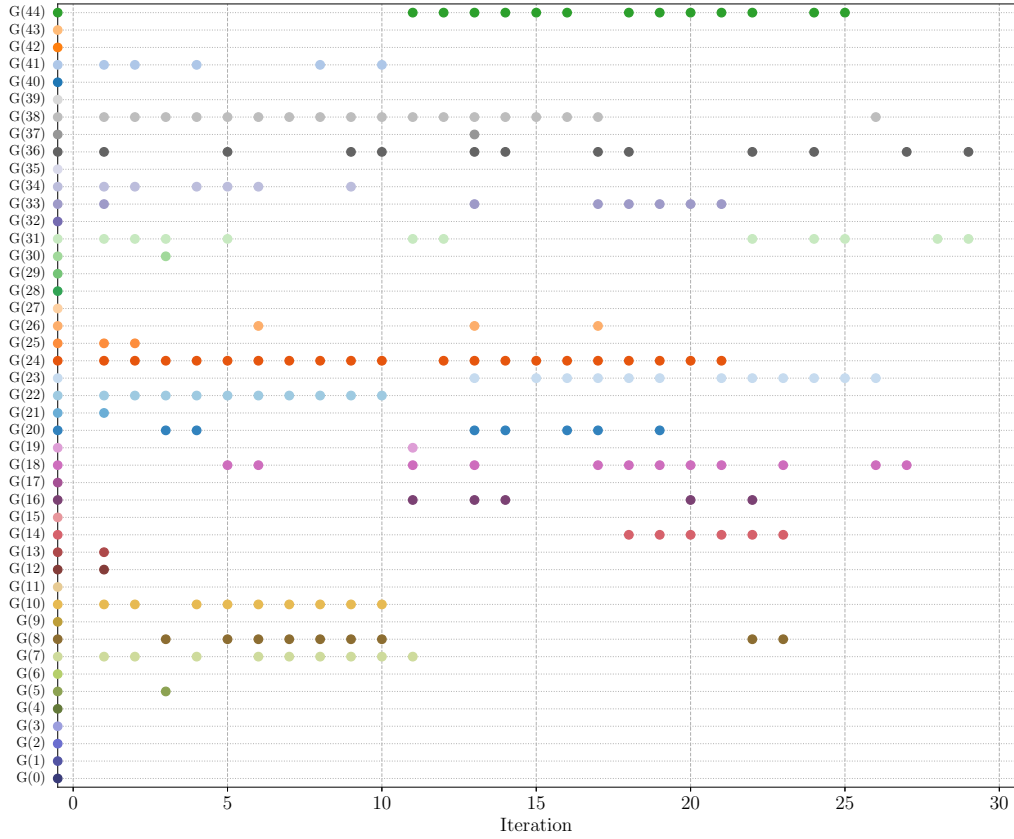


Figure H.3: Scatter of constraints not meeting the tolerance threshold for the relaxed constraints for the clean configuration, no payload.

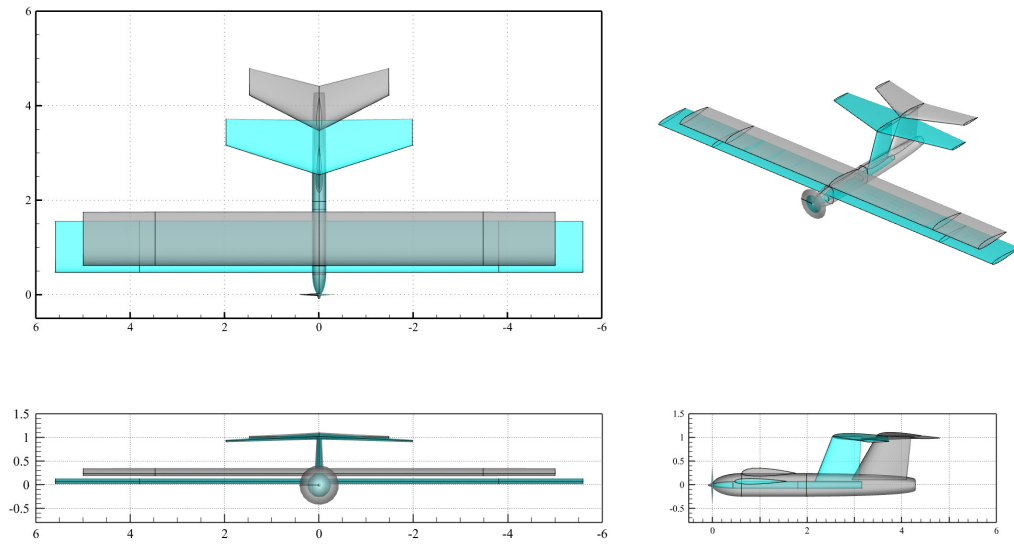


Figure H.4: Solution for the clean configuration without payload with relaxed constraints shown in blue, the original vehicle shown in grey. Axes are in feet.

The evolution of the design variables is shown in Figure H.5. This result does not maximize the number of solar cells on the wings, which is contrary to other results. Without payload, the clean configuration is able to obtain a high fitness value without greater increases to span or aspect ratio. By increasing the number of panels from $P(0)=10$ and $P(1)=8$ to $P(0)=P(1)=12$, the fitness increases from 131.04 h to 131.89 h. Some of the vehicle properties of the clean configuration without payload vehicle from Chapter G and this vehicle are shown in Table H.3, which are referred to as the original and new UAVs, respectively. The new vehicle has a thinner and shorter fuselage, larger wing, higher $\mathcal{R}$ and larger span, all properties which the parametric analysis suggests should result in better performance. The different propulsion systems results in the new vehicle having a higher power draw and therefore lower performance. With the same propulsion system (adopting that of the original vehicle), the new vehicle's fitness increases to 131.64 h, remaining marginally lower than the original vehicle. Without payload, the increased span has not proven effective at outperforming the original vehicle that was optimized for a shorter span.

Table H.3: Clean configuration without payload vehicle properties for different span constraints.

Parameter	Unit	$G(7) \leq 10$	$G(7) \leq 12$
n_{PV}	[-]	36	36
S	[ft ²]	11.40	12.24
$\bar{c}$	[ft]	1.14	1.10
b	[ft]	9.99	11.17
λ	[-]	0.99	0.99
$\mathcal{R}$	[-]	8.76	10.19
w_{fuse}	[ft]	0.31	0.31
h_{fuse}	[ft]	0.50	0.16
$\mathcal{R}_{HT}$	[-]	3.86	4.59
S_{HT}	[ft ²]	2.25	3.40
$\mathcal{R}_{VT}$	[-]	1.95	2.47
S_{VT}	[ft ²]	1.75	1.45
x_{CG}	[% MAC]	33	45
W_{batt}	[lb _f]	1.2	1.2
W_{struc}	[lb _f]	0.2	0.3
W_{prop}	[lb _f]	0.5	0.2
W_{pl}	[lb _f]	0.8	0.5
OEW	[lb _f]	0.7	0.5
W_0	[lb _f]	2.7	2.2
SM	[% MAC]	24	14
n_z	[-]	1.00	1.00
C_{D_0}	[-]	0.0207	0.0193
K	[-]	0.038	0.033
$C_{L_{max}}$	[-]	1.73	1.64
V	$[\frac{ft}{s}]$	13.8	13.0
V_{stall}	$[\frac{ft}{s}]$	11.0	10.0
$V_{P_{min}}$	$[\frac{ft}{s}]$	12.8	11.1
$C_{l,\beta}$	[rad ⁻¹]	-0.078	-0.064
$C_{n,\beta}$	[rad ⁻¹]	0.066	0.043
T_{max}	[lb _f]	6.6	0.7
E	[Wh]	93.2	93.2
$P_p + P_{pl}$	[W]	3.1	3.3
fitness	[h]	131.93	131.04

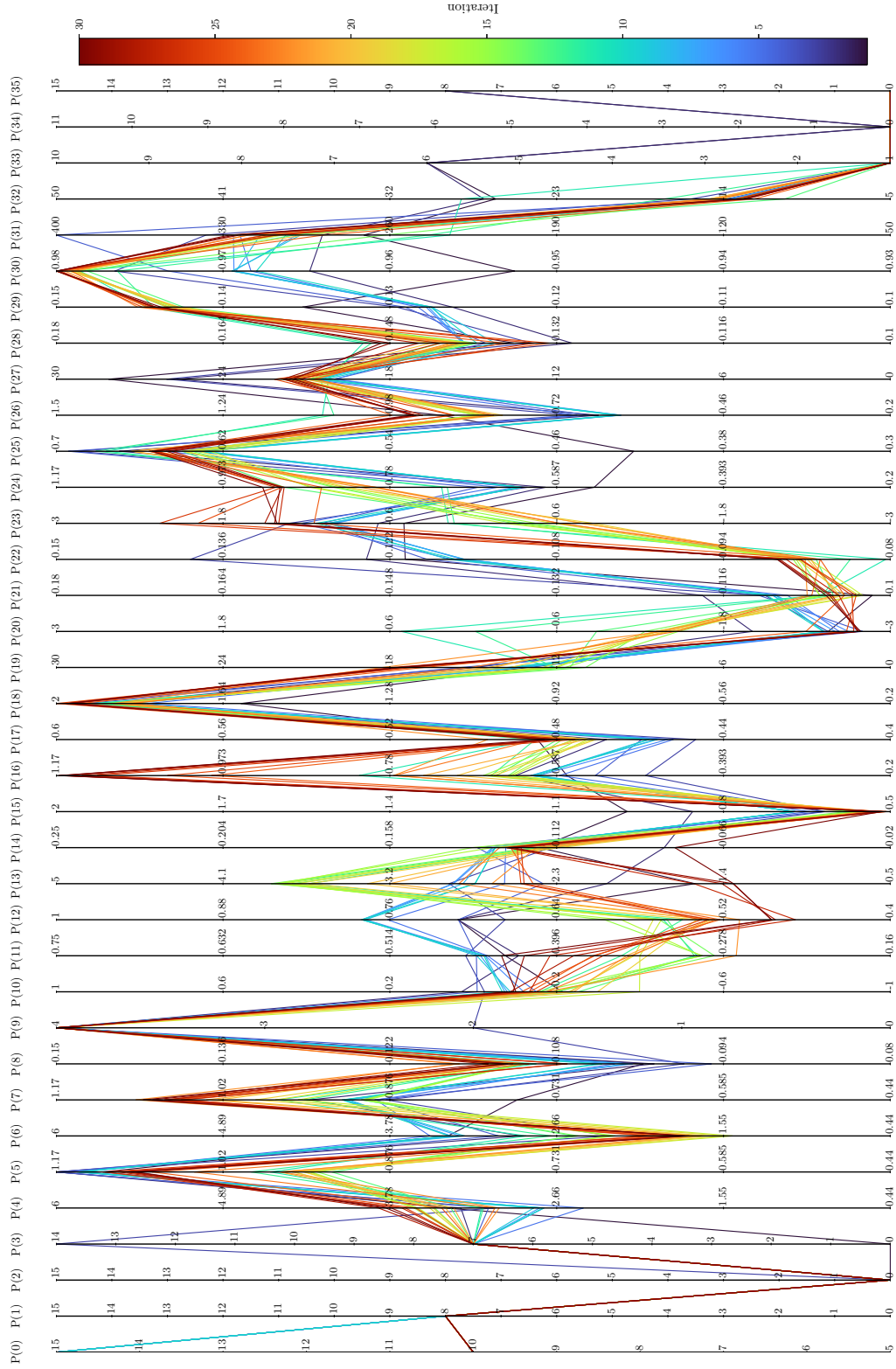


Figure H.5: Design variables for the clean configuration without payload with relaxed constraints.

H.2 Clean Configuration with Payload

This section outlines the results of the clean configuration optimization with payload, with relaxed constraints detailed in Table H.1. The optimization was manually stopped after 133 iterations as the design space had found solutions not in violation of any constraints and began oscillating between designs in violation of the span constraint and being within the feasible design space with no significant increase in fitness. The fitness is shown in Figure H.6. The fitness must be considered in conjunction with the constraints to identify the solution to this optimization. The values of the constraints not meeting the tolerance threshold are shown in Figure H.7. The scatter plot of these constraints is shown in Figure H.8.

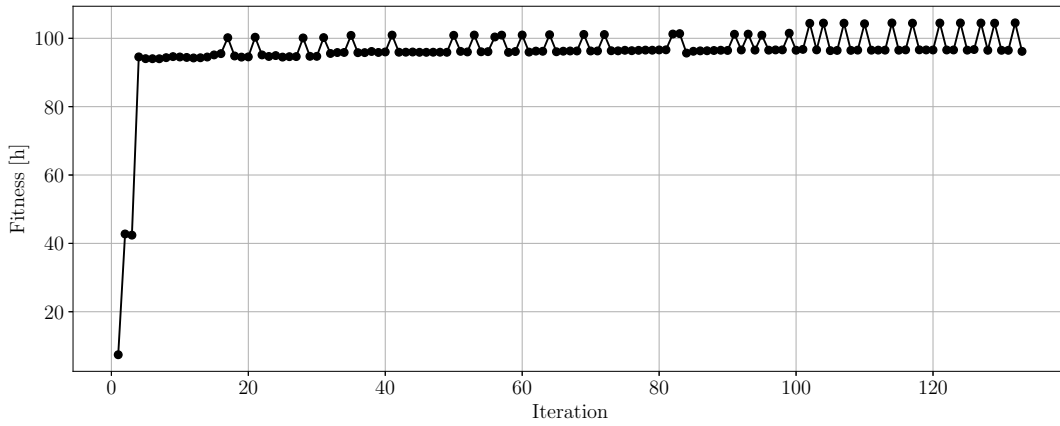


Figure H.6: Fitness convergence with relaxed constraints for the clean configuration, with payload.

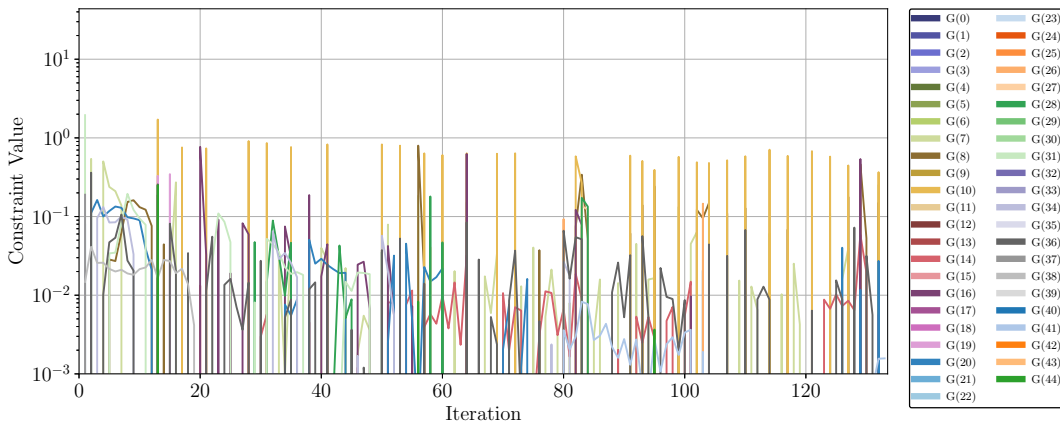


Figure H.7: Constraints violated and their values for the relaxed constraints for the clean configuration, with payload.

Iterations 105, 106, 108, 120, and 122 satisfy the constraints and have the most promis-

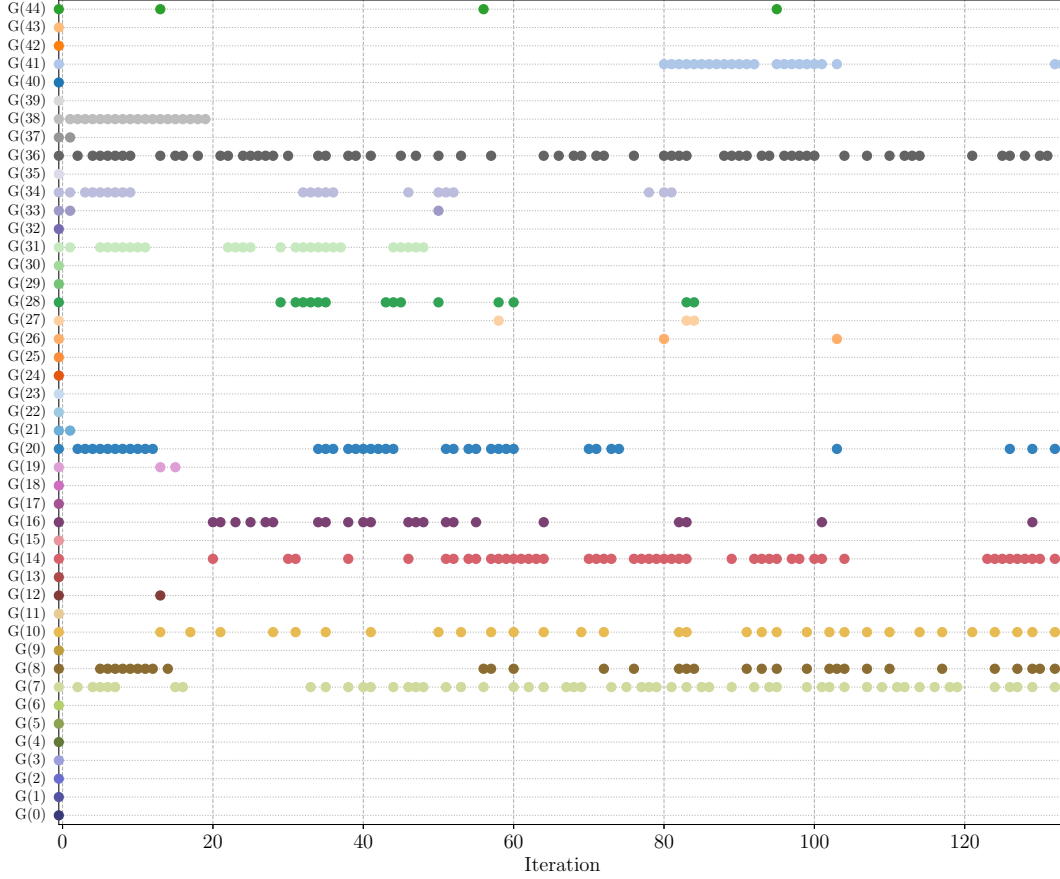


Figure H.8: Scatter of constraints not meeting the tolerance threshold for the relaxed constraints for the clean configuration, with payload.

ing fitness values. Iteration 122 is considered the solution for this optimization with the highest fitness of 96.58 h. This fitness corresponds to an ability to recharge batteries in months 6, 3, and 9, with $T_{cm} = 1.63$ h in month 9. The fitness at the larger span constraint is much more stable than at the reduced constraint in the original optimization, supporting that larger spans are required to achieve LALE solar-powered flight when the power draw increases with payload. Unlike the no payload optimization, this optimization saturates the allocated span, remaining on the border of the span constraint. The evolution of the design variables are shown in Figure H.9.

This design also does not maximize the number of solar cells that fit on the wing. By increasing $P(0)=10$ and $P(1)=8$ to $P(0)=P(1)=13$, the fitness increases from 96.58 h to 102.96 h, creating larger energetic margins for months that it is capable of recharging. The final geometry is shown in Figure H.10 in blue, with the original geometry from Chapter 5 shown in grey. Some of the vehicle properties of this optimization are compared to the original vehicle are summarized in Table H.4. The larger span and higher $\mathcal{R}$ coupled with lower drag coefficients and therefore lower power output significantly improves the performance of the UAV. The time of year that LALE solar-powered flight can be achieved has a larger window with the relaxed span constraint.

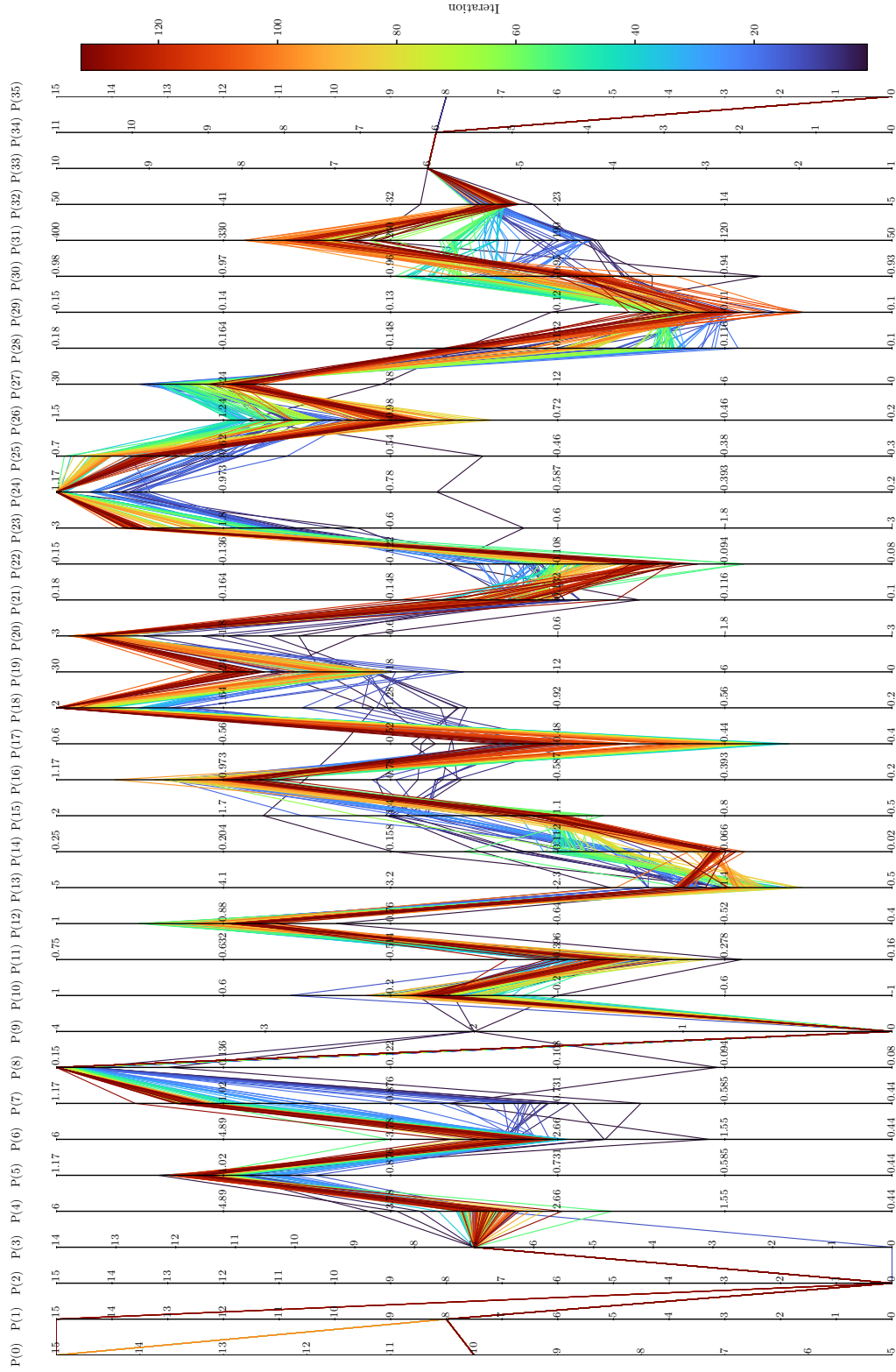


Figure H.9: Design variables for the clean configuration, with payload with relaxed constraints.

Table H.4: Clean configuration with payload vehicle properties for different span constraints.

Parameter	Unit	$G(7) \leq 10$	$G(7) \leq 12$
n_{PV}	[-]	36	36
S	[ft ²]	11.02	12.71
$\bar{c}$	[ft]	1.11	1.06
b	[ft]	9.91	12.00
λ	[-]	0.97	0.97
$\mathcal{R}$	[-]	8.91	11.32
w_{fuse}	[ft]	0.37	0.35
h_{fuse}	[ft]	0.40	0.39
$\mathcal{R}_{HT}$	[-]	6.01	5.52
S_{HT}	[ft ²]	2.66	2.88
$\mathcal{R}_{VT}$	[-]	1.70	1.96
S_{VT}	[ft ²]	1.31	1.83
x_{CG}	[% MAC]	29	39
W_{batt}	[lb _f]	7.2	7.2
W_{struc}	[lb _f]	0.7	1.1
W_{prop}	[lb _f]	0.5	0.2
W_{pl}	[lb _f]	0.8	0.8
OEW	[lb _f]	1.2	1.3
W_0	[lb _f]	9.2	9.3
SM	[% MAC]	25	16
n_z	[-]	1.00	1.00
C_{D_0}	[-]	0.0195	0.0190
K	[-]	0.035	0.029
$C_{L_{max}}$	[-]	1.73	1.73
V	[$\frac{\text{ft}}{\text{s}}$]	27.3	25.7
V_{stall}	[$\frac{\text{ft}}{\text{s}}$]	20.8	19.5
$V_{P_{min}}$	[$\frac{\text{ft}}{\text{s}}$]	24.1	21.7
$C_{l,\beta}$	[rad ⁻¹]	-0.070	-0.056
$C_{n,\beta}$	[rad ⁻¹]	0.034	0.031
T_{max}	[lb _f]	6.2	1.6
E	[Wh]	559.4	559.4
$P_p + P_{pl}$	[W]	27.5	25.9
fitness	[h]	43.37	96.58

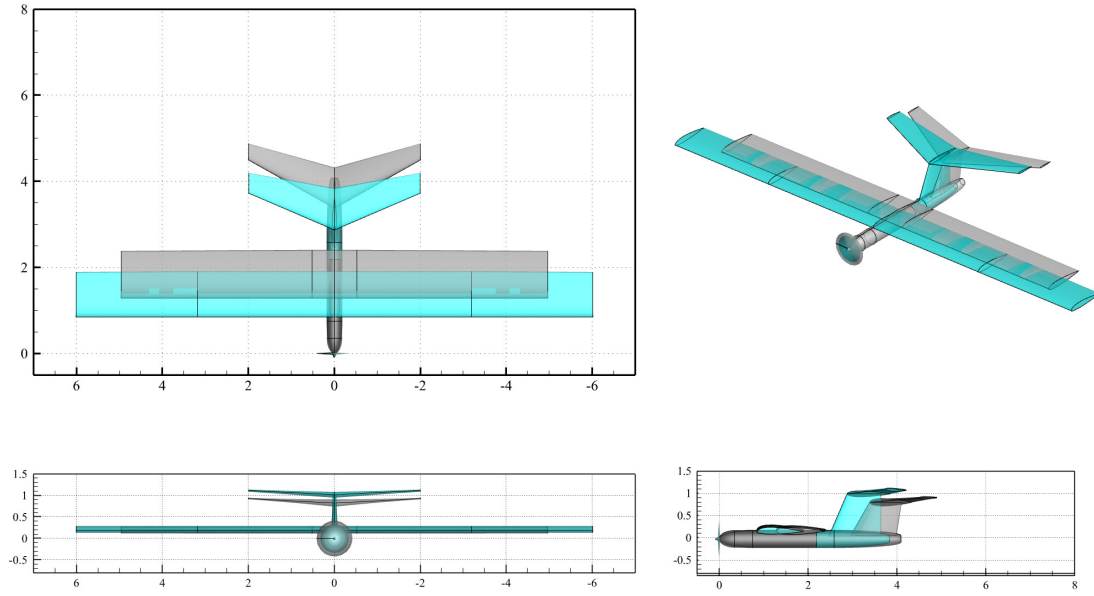


Figure H.10: Solution for the clean configuration, with payload with relaxed constraints shown in blue, the original vehicle shown in grey. Axes are in feet.

H.3 Nacelles without Payload

This section outlines the results of the optimization with relaxed constraints for the nacelle configuration with no payload. The optimization converged after 60 iterations with the final solution being fully compliant to every constraint. The fitness throughout the iterations is shown in Figure H.11. The final vehicle has a fitness of 132.54 h.

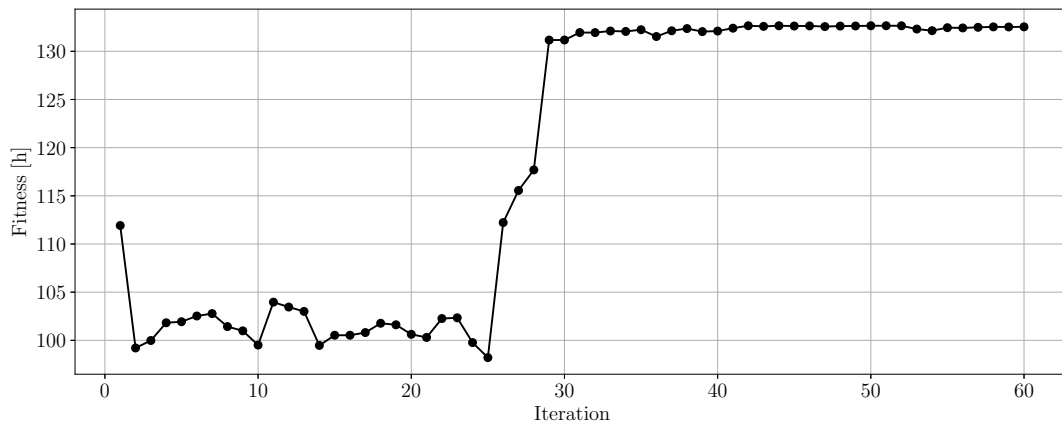


Figure H.11: Fitness convergence with relaxed constraints for the nacelle configuration, no payload.

The values of the constraints not meeting the tolerance threshold are plotted in Fig-

ure H.12. The scatter plot of these constraints in each iteration is in Figure H.13. The violation of the span constraint (G(7)) indicates the solution tends towards maximizing the span with as low of a flight speed as permissible, indicated by the consistent violation of the C_L constraint (G(41)). The evolution of the design variables is shown in Figure H.15.

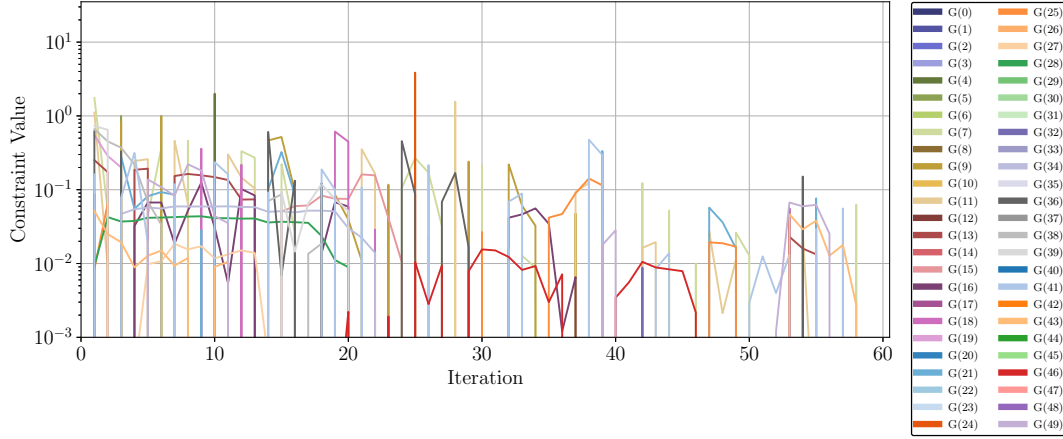


Figure H.12: Constraints violated and their values for the relaxed constraints for the nacelle configuration, no payload.

As with several other results, this configuration does not maximize the number of solar cells on the wing. This is the only wing that does not tend towards an entirely rectangular planform, yet there remains area on the wing that could be filled with solar cells. If the number of solar cells was changed to $P(0)=P(1)=12$, the fitness increases from 132.54 h to 132.95 h, showing that the optimization converged in a local optimum. The final geometry is shown in Figure H.14 in blue, with the original vehicle shown in grey.

Table H.5 compares vehicle properties of the original vehicle to the vehicle with the relaxed constraints. A larger span, larger number of solar cells, lower drag coefficients, and overall lower power draw contributes to a higher fitness, indicating larger energetic margins than the original vehicle with the smaller wingspan. The lower power draw is attributed to a different motor-propeller combination than the original vehicle, which also contributes to the lower maximum thrust value.

Table H.5: Nacelle configuration without payload vehicle properties for different span constraints.

Parameter	Unit	$G(7) \leq 10$	$G(7) \leq 12$
n_{PV}	[-]	36	40
S	[ft ²]	11.07	13.02
$\bar{c}$	[ft]	1.12	1.09
b	[ft]	9.86	11.99
λ	[-]	1.00	0.74
$\mathcal{R}$	[-]	8.79	11.04
w_{nac}	[ft]	0.30	0.37
h_{nac}	[ft]	0.17	0.18
w_{fuse}	[ft]	0.18	0.20
h_{fuse}	[ft]	0.33	0.40
$\mathcal{R}_{HT}$	[-]	3.30	5.27
S_{HT}	[ft ²]	2.08	2.61
$\mathcal{R}_{VT}$	[-]	1.55	2.35
S_{VT}	[ft ²]	1.36	1.75
x_{CG}	[% MAC]	22	26
W_{batt}	[lb _f]	1.2	1.2
W_{struc}	[lb _f]	0.3	0.2
W_{prop}	[lb _f]	0.5	0.2
W_{pl}	[lb _f]	0.5	0.6
OEW	[lb _f]	0.7	0.4
W_0	[lb _f]	2.5	2.2
SM	[% MAC]	25	23
n_z	[-]	1.00	1.00
C_{D0}	[-]	0.0235	0.0234
K	[-]	0.038	0.032
C_{Lmax}	[-]	1.63	1.73
V	[$\frac{ft}{s}$]	14.4	12.3
V_{stall}	[$\frac{ft}{s}$]	11.0	9.4
V_{Pmin}	[$\frac{ft}{s}$]	12.1	10.2
$C_{l,\beta}$	[rad ⁻¹]	-0.088	-0.054
$C_{n,\beta}$	[rad ⁻¹]	0.033	0.033
T_{max}	[lb _f]	6.5	1.9
E	[Wh]	93.2	93.2
$P_p + P_{pl}$	[W]	3.5	2.8
fitness	[h]	130.75	132.54

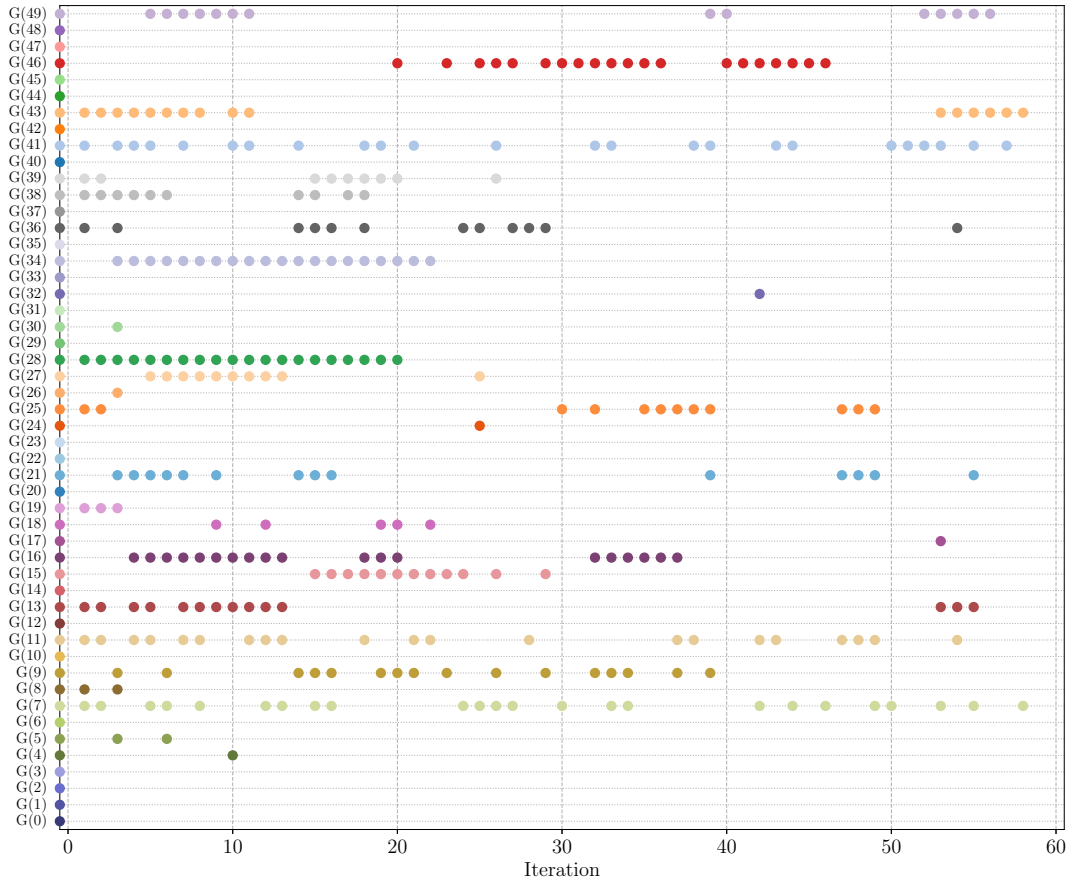


Figure H.13: Scatter of constraints not meeting the tolerance threshold for the relaxed constraints for the nacelle configuration, no payload.

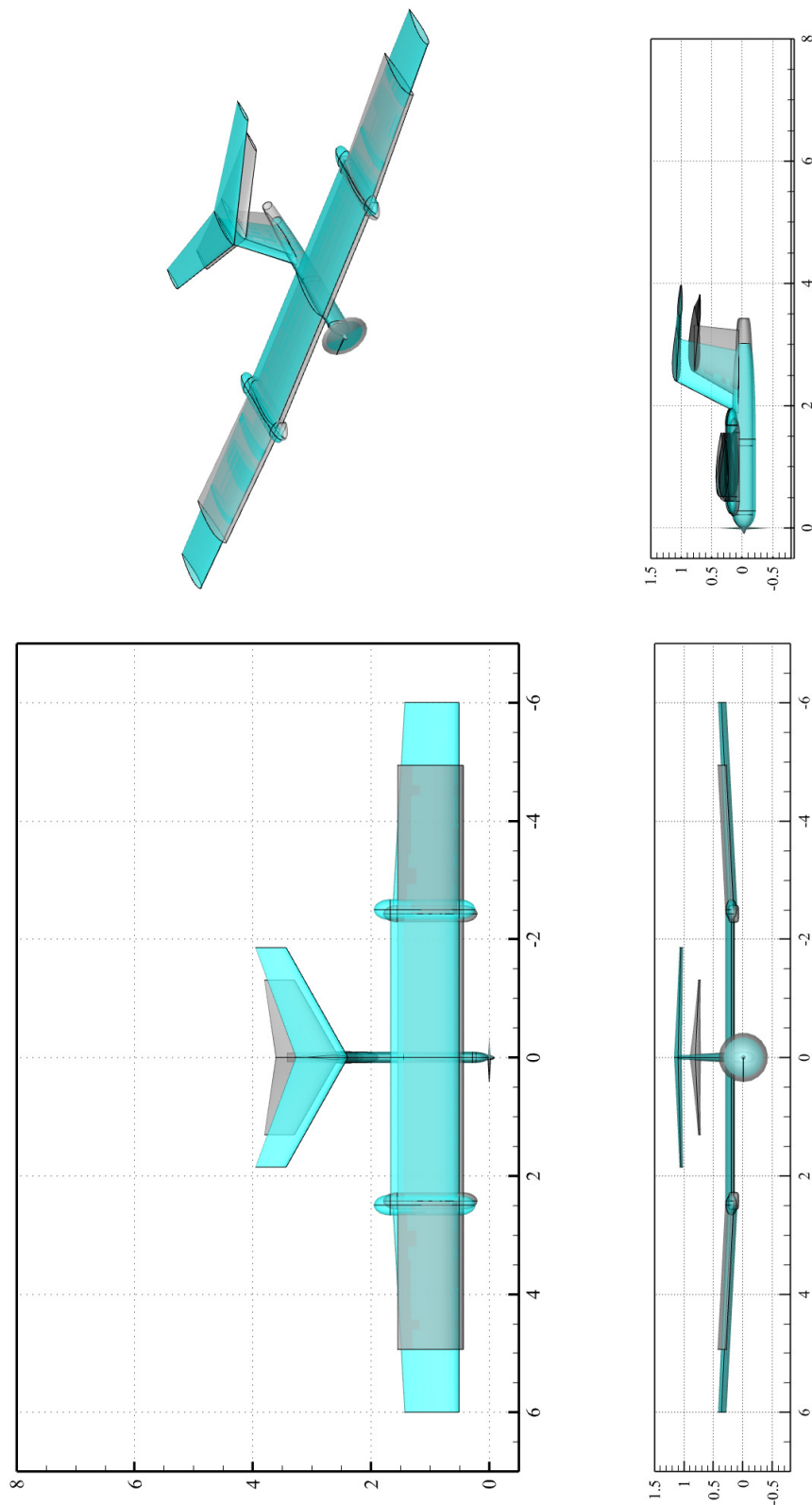


Figure H.14: Solution for the nacelle configuration without payload with relaxed constraints shown in blue, the original vehicle shown in grey. Axes are in feet.

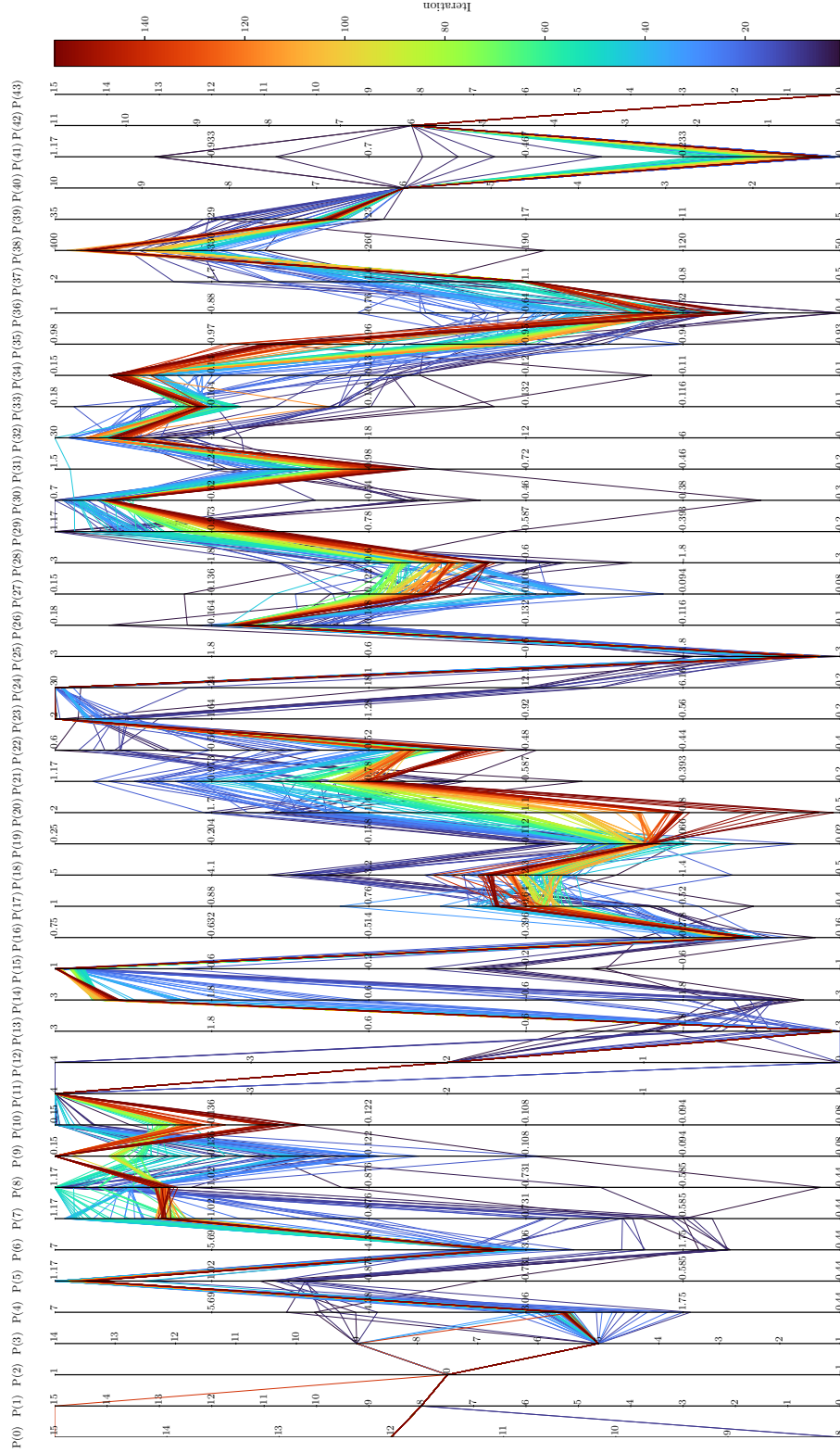


Figure H.15: Design variables for the nacelle configurations without payload, with relaxed constraints.

H.4 Nacelles with Payload

This section outlines the results of the optimization with relaxed constraints for the nacelle configuration with payload. The optimization was manually ended after 144 iterations due to the fitness plateauing and a solution with no violated constraints existing within the solution set. The fitness is shown in Figure H.16. The values of the constraints not meeting the tolerance threshold are shown in Figure H.17 and the scatter plot of these constraints are in Figure H.18.

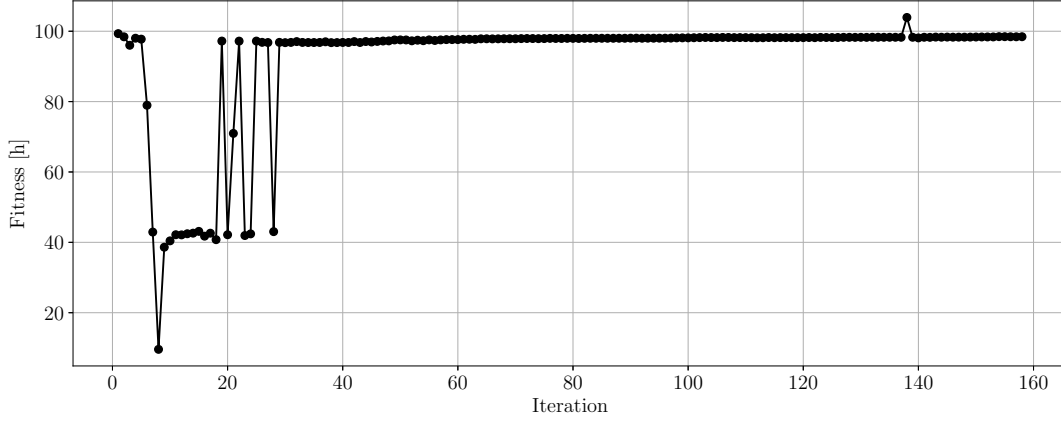


Figure H.16: Fitness convergence with relaxed constraints for the nacelle configuration, with payload.

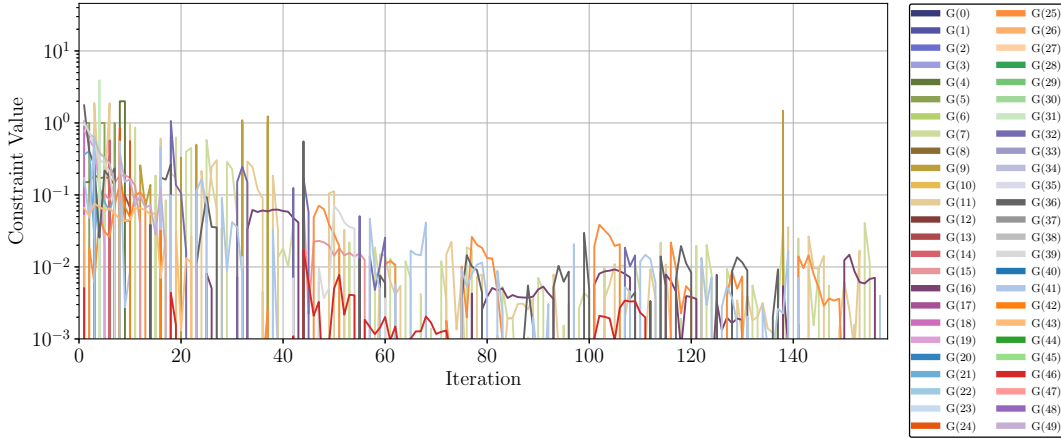


Figure H.17: Constraints violated and their values for the relaxed constraints for the nacelle configuration, with payload.

Iteration 140 does not violate any constraints and has a fitness of 98.11 h. This is considered the solution of this optimization. The evolution of the design variables is shown in Figure H.19. This configuration maximizes the span, continuously brushing against the span constraint (G(7)) and maintaining a rectangular planform. The optimization



Figure H.18: Scatter of constraints not meeting the tolerance threshold for the relaxed constraints for the nacelle configuration, with payload.

algorithm is generally seeking to minimize the cruising airspeed, which continuously violated the lift coefficient constraint (G(41)) by using too low of an airspeed. The geometry is shown in Figure H.20 in blue, with the result from Chapter 5 shown in grey. Similar to the other optimization results, the solar cell area is not maximized on the wings. By changing the solar array from $P(1)=8$ to $P(1)=12$, the fitness increases from 98.11 h to 101.27 h, effectively increasing the energetic margins on the days that LALE solar-powered flight can occur continuously.

Table H.6 summarizes vehicle properties for the solution from Chapter 5 and the solution in this section, referred to as the original and new UAVs, respectively. The new vehicle significantly outperforms the original, largely due to its increased span. The propulsion system and energy capacity are appropriately balanced to accommodate the larger power draw of the payload, with energy capacities matching the clean configurations with payload, irrespective of their spans. Based on the definition of the fitness function, the nacelle configuration greatly benefits from the higher span in cases with and without payload.

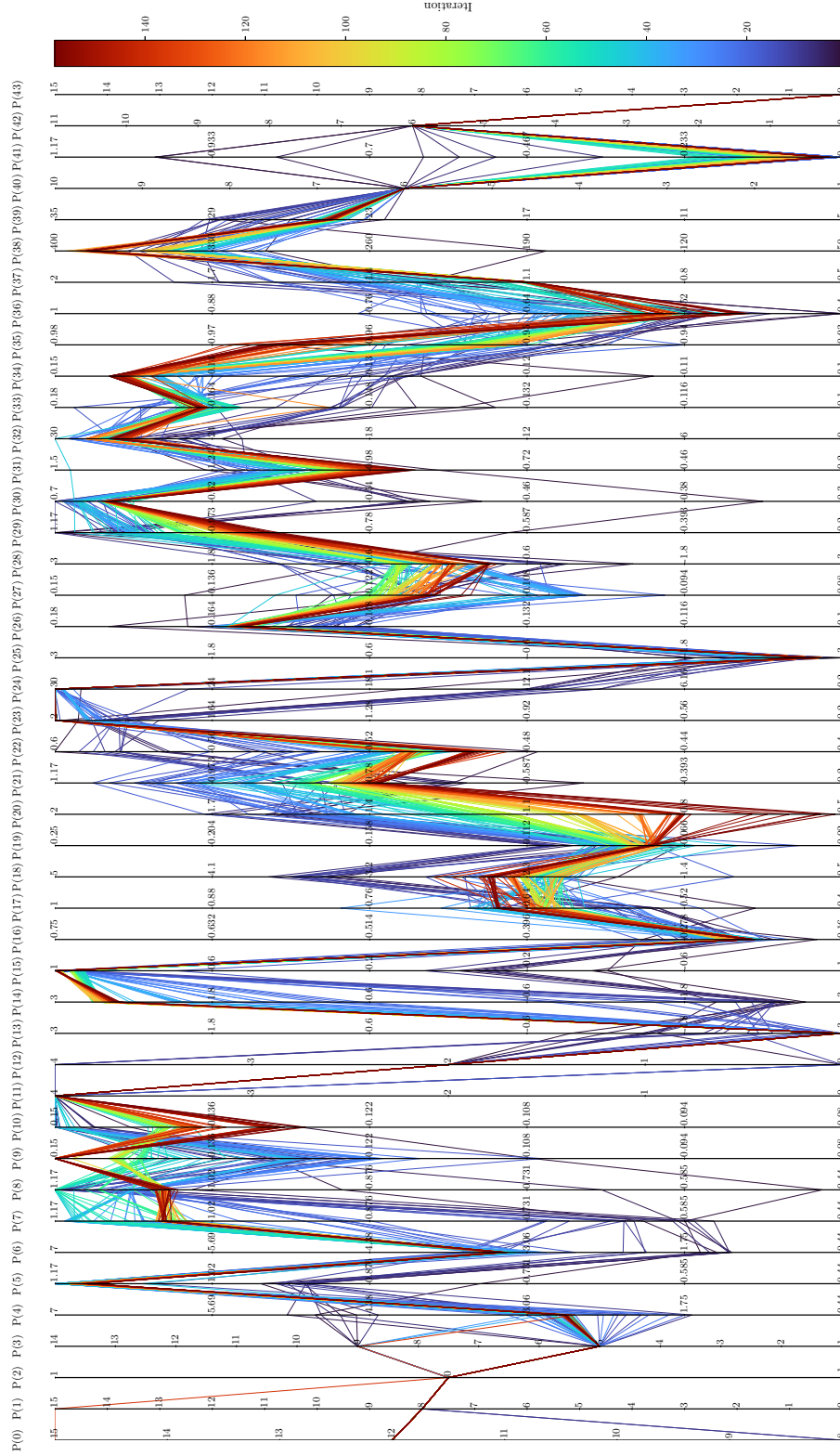


Figure H.19: Design variables for the nacelle configurations with payload, with relaxed constraints.

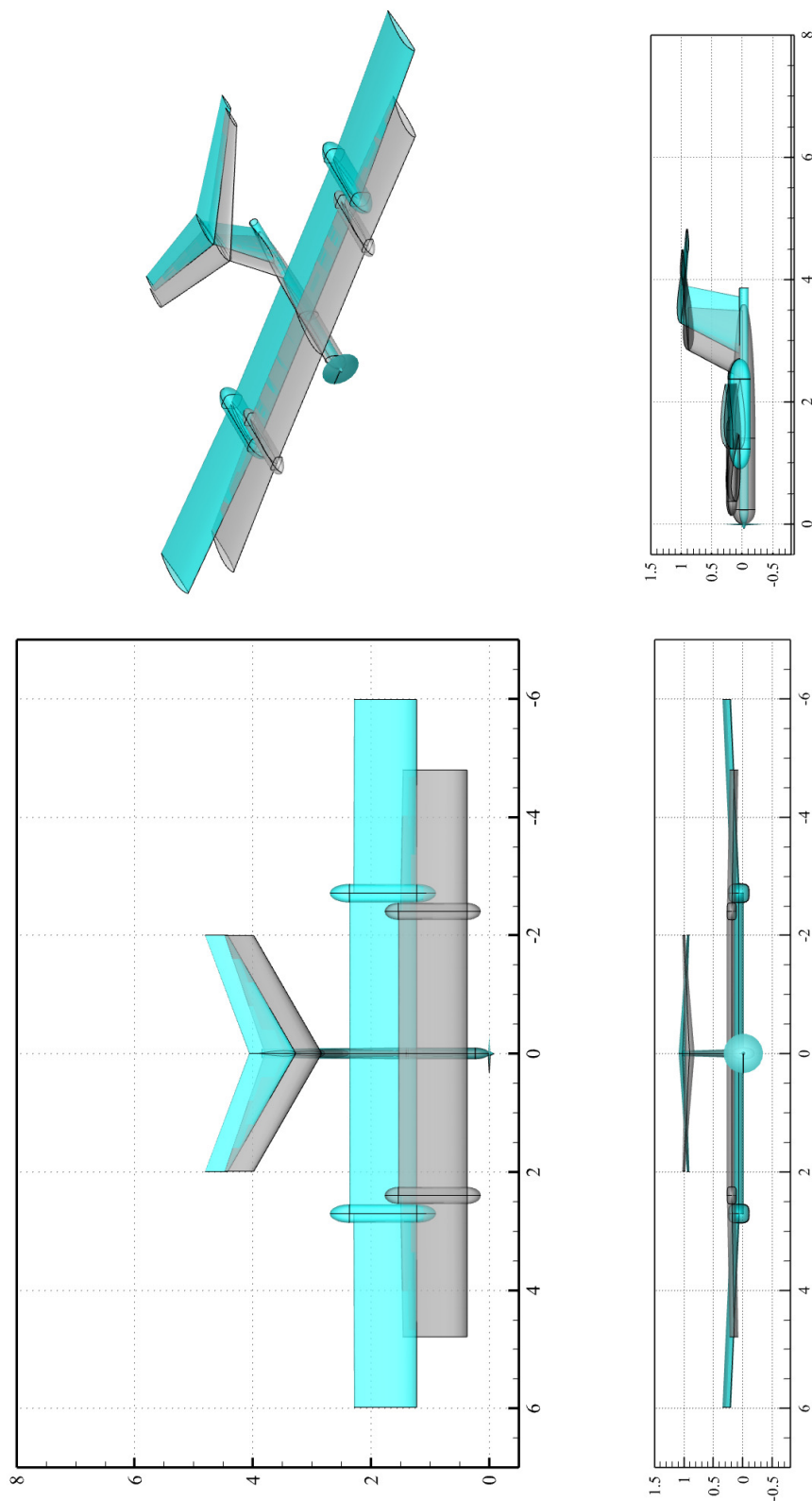


Figure H.20: Solution for the nacelle configuration with payload with relaxed constraints shown in blue, the original vehicle shown in grey. Axes are in feet.

Table H.6: Nacelle configuration with payload vehicle properties for different span constraints.

Parameter	Unit	$G(7) \leq 10$	$G(7) \leq 12$
n_{PV}	[-]	36	40
S	[ft ²]	10.92	13.17
$\bar{c}$	[ft]	1.14	1.10
b	[ft]	9.57	11.95
λ	[-]	0.92	0.93
$\mathcal{R}$	[-]	8.39	10.85
w_{nac}	[ft]	0.30	0.31
h_{nac}	[ft]	0.16	0.34
w_{fuse}	[ft]	0.21	0.19
h_{fuse}	[ft]	0.39	0.15
$\mathcal{R}_{HT}$	[-]	5.46	6.80
S_{HT}	[ft ²]	2.91	2.35
$\mathcal{R}_{VT}$	[-]	1.95	2.45
S_{VT}	[ft ²]	1.62	1.49
x_{CG}	[% MAC]	44	30
W_{batt}	[lb _f]	1.2	7.2
W_{struc}	[lb _f]	0.2	0.7
W_{prop}	[lb _f]	0.5	0.2
W_{pl}	[lb _f]	0.8	0.9
OEW	[lb _f]	0.7	1.0
W_0	[lb _f]	2.7	9.0
SM	[% MAC]	18	18
n_z	[-]	1.00	1.00
C_{D0}	[-]	0.0246	0.0217
K	[-]	0.054	0.030
C_{Lmax}	[-]	1.67	1.73
V	[$\frac{ft}{s}$]	15.0	24.8
V_{stall}	[$\frac{ft}{s}$]	11.4	18.8
V_{Pmin}	[$\frac{ft}{s}$]	13.7	20.4
$C_{l,\beta}$	[rad ⁻¹]	-0.048	-0.072
$C_{n,\beta}$	[rad ⁻¹]	0.062	0.033
T_{max}	[lb _f]	2.9	1.6
E	[Wh]	93.2	559.4
$P_p + P_{pl}$	[W]	12.0	26.5
fitness	[h]	5.07	98.11

H.5 Relaxed Constraint Optimization Summary

When considering the vehicles without additional payload, the increased span does not have significant advantages over the smaller wingspans. The differences in fitness indicate small variations in the energetic margins that each vehicle possess, but do not indicate the clean or the nacelle configurations at either span to be more optimal than the other. The longer wingspan increases potential aeroelastic complications, makes

ground handling more difficult, and decreases the feasibility for hand launching the vehicle for takeoff.

When considering the configurations with payload, the wingspan has a considerable effect. The larger wingspan vehicles are able to recharge the batteries in months 6, 3, and 9, with the nacelle configuration having the most notable increase in fitness. The longer wingspan is better suited for the applications with higher power draws. The results of the parametric analysis indicate that this configuration would have the better performance at higher latitudes, which is not considered in the optimization to reduce the number of design variables and total analyses. Considering the direct solutions from the optimization, the nacelle configuration has the higher fitness for applications with payload, where none of the solutions maximize the number of solar panels on the wing. If considering the fitness when the number of solar cells is maximized on the wing planform, the clean configuration out-performs the nacelle configuration due to its larger solar array across the same span and its ability to charge the batteries faster, increasing the charge margin. The clean configuration is all around the better configuration, which is consistent with the results of this appendix and Chapter 5. The clean configuration as the most optimal for LALE solar-powered flight is also consistent with the various LALE solar UAVs that have been designed, tested, and flown continuously for 24+ h.

I NEAT as Path Planning Validation

The orbit-raising optimization problem is presented as a validation of using the NEAT algorithm for generating neural networks to use in trajectory optimization problems. The formulation of the problem is taken from [116]. The problem is formulated as a constant-thrust rocket, T , operating for a given duration, t_f , with the objective of finding the direction of the thrust vector, β , to transfer the rocket from an initial orbit, r_o , to a maximum final orbit, $r(t_f)$, shown in Figure I.1.

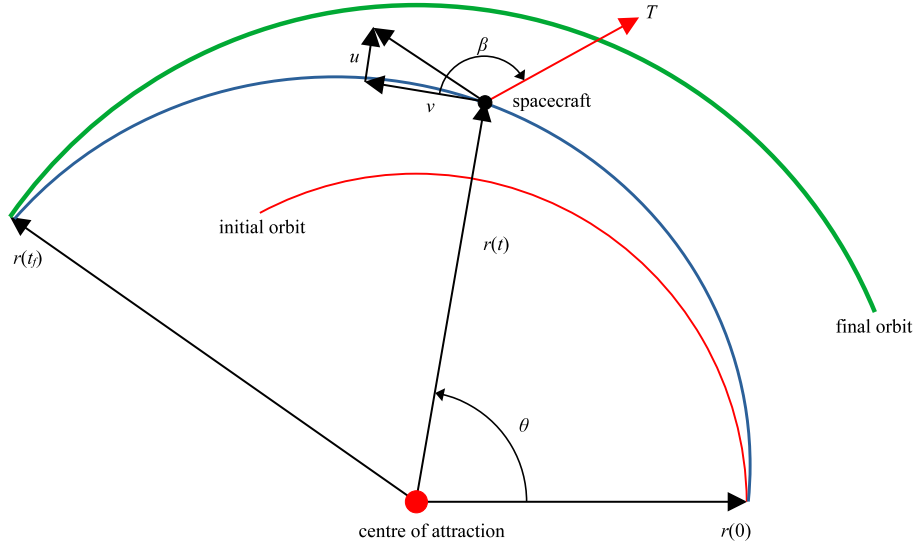


Figure I.1: Visualization of the orbit-raising problem [117].

The objective is to maximize the function

$$J = r(t_f)$$

subject to the following dynamic constraints:

$$\begin{aligned}\dot{r} &= u \\ \dot{u} &= \frac{v^2}{r} - \frac{\mu}{r^2} + \frac{T \sin \beta}{m_o - |\dot{m}|t} \\ \dot{v} &= -\frac{uv}{r} + \frac{T \cos \beta}{m_o - |\dot{m}|t}\end{aligned}$$

with the following boundary conditions:

$$\begin{aligned}r(0) &= r_0 \\ u(0) &= 0 \\ v(0) &= \sqrt{\frac{\mu}{r_o}}\psi_1 = u(t_f) = 0 \\ \psi_2 &= v(t_f) - \sqrt{\frac{\mu}{r(t_f)}} = 0\end{aligned}$$

The constants for each equation is given in Table I.1.

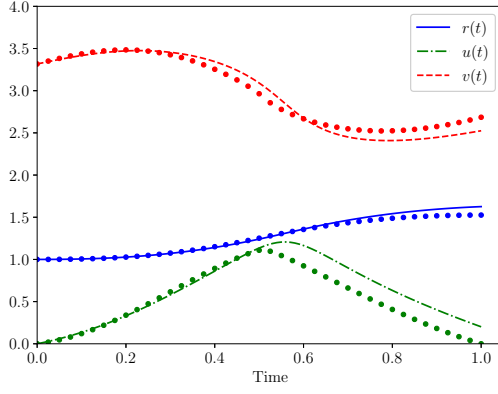
Table I.1: Constants used in equation formulation.

Variable	Value
t_f	1
r_o	1
μ	11
T	1.55
m_o	1
$\dot{m}$	-0.25

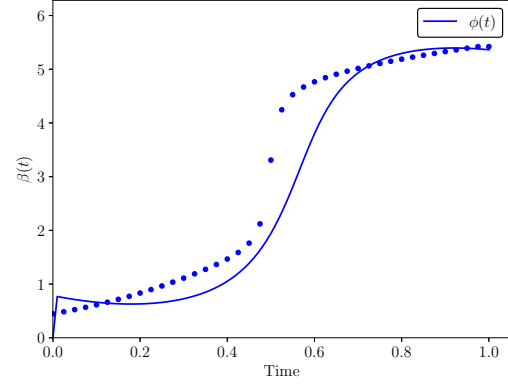
In NEAT, the thrust vector angle β is the output and r , u , and v are the input states. The fitness function is defined as

$$J = 3r(t_f) - 5\psi_1^2 - 5\psi_2^2$$

The boundary conditions at t_f are enforced in the fitness function where the network is penalized for not satisfying each bound. The results of the control problem are shown in Figure I.2. The solution from classical optimization is shown as individual data points and the NEAT solution is shown as the continuous curves. The angle β is in radians. The NEAT solution is achieved after 100 generations. The definition of the objective function for the NEAT algorithm has a large influence on the resulting network, where the optimization constraints are enforced by the ψ_1 and ψ_2 terms. The values of each term in the fitness function and the total fitness is summarized in Table I.2 for both optimal control techniques. Although the solutions are not numerically identical, there is considerable agreement in the final states, showing the effectiveness and applicability of using NEAT as an adaptive trajectory optimization tool.



(a) States.



(b) Control.

Figure I.2: Orbit-raising optimization results using trajectory optimization (data points) and NEAT (continuous curves).

Table I.2: Fitness values for both optimal control techniques.

Variable	NEAT	Classical
$r(t_f)$	1.626	1.526
ψ_1	0.041	0.000
ψ_2	0.006	0.000
J	4.577	4.577