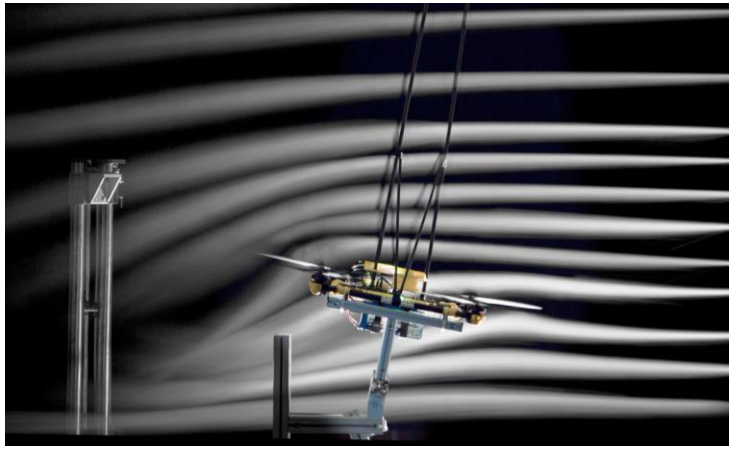
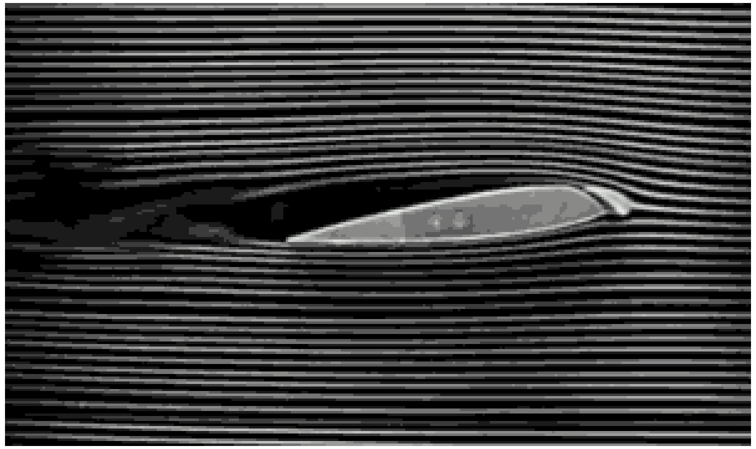
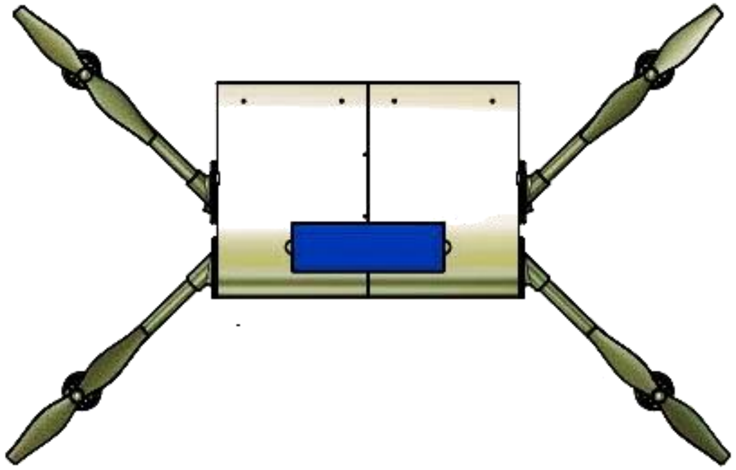
QUADFOIL – NOVEL QUADCOPTER DESIGN

- Low energy density of Li-ion or Li-po battery compared to fossil fuel necessitates the aerodynamic optimization of design.







Quadcopter + **Airfoil** = **Quadfoil**

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Created by – Sarthak Deshmukh
Aerospace Graduate Researcher

Project Overview

Design

1. Define Requirements
2. Airfoil Selection
3. Wing Sizing
4. Prototyping

Flight Testing

1. Mission Plan & Flight Parameter Setup
2. Aircrafts
3. Data Segregation and Analysis

CFD Analysis

1. Geometry & Mesh Generation
2. Mesh Assessment & Refinement
3. Overall model setup and simulation run
4. CFD Analysis & Experimental Data Comparison

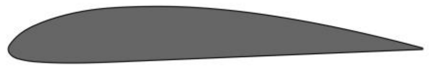
Wind Tunnel Testing

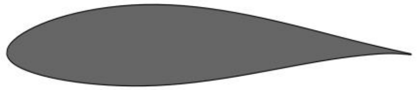
1. Design of Experiment (DOE)
2. Tunnel Setup
3. Test and Objectives
4. Tunnel data analysis and comparison with flight test data.

Design Requirements

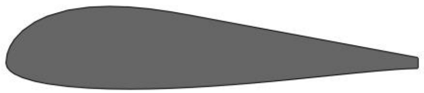
- **Ability to carry inspection/surveillance equipment**
- **Long endurance and range**
- **Point takeoff and landing capability**
- **Quick deployment and turn-around time**
- **Scalable**

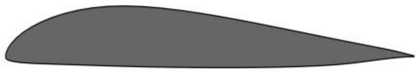
Selecting the Optimum Airfoil for Quadfoil UAV

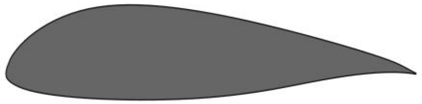

(a) Clark Y airfoil


(b) NACA 633418 airfoil


(c) NASA GA W1 airfoil


(d) ARA-D 20% airfoil


(e) Eppler E334 airfoil


(f) Eppler1233 airfoil

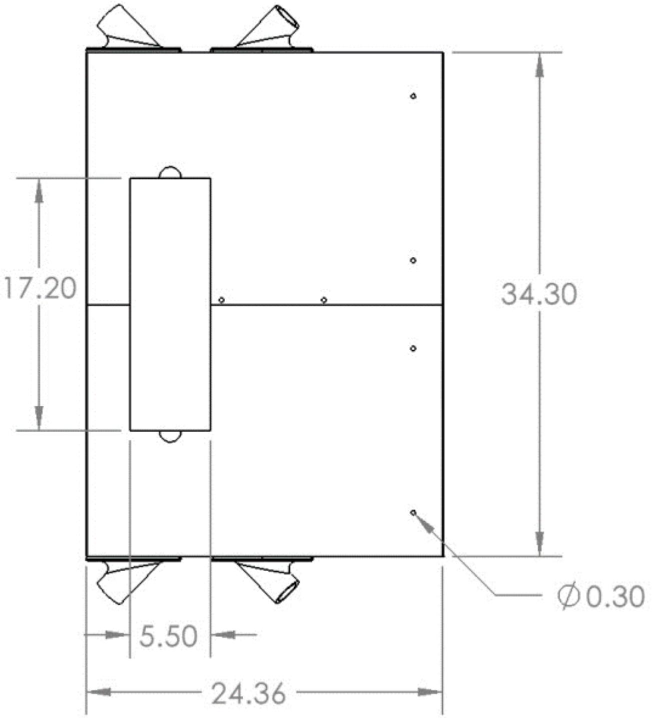
Airfoil Name	**Maximum L/D ratio**	**Airfoil Area (Square units)**	**Maximum t/c ratio**	**ΔCm for 0-15deg AOA**	**Overall Score**
Clark Y	72	0.08	11.7	0.07	12
NACA 663418	69	0.11	18	0.0045	16
NASA GA(W)-1	61	0.12	17	0.04	15
ARA-D 20%	52	0.13	20	0.02	19
Eppler E334	66	0.08	11.9	0.08	8
Eppler 1233	56	0.12	18.9	0.075	14

The ARAD 20 airfoil meets all our design requirements, making it the optimal choice for our Quadfoil UAV.

Electronics Needs Dictates the Wing Sizing of Quadfoil

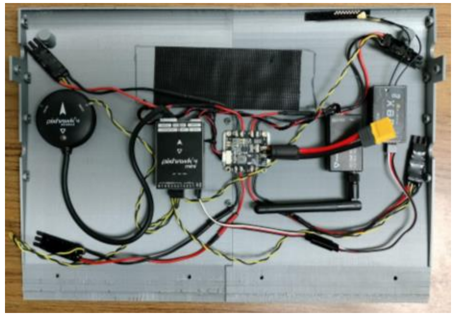
The size of the wing is a critical factor in the overall design of the Quadfoil UAV

- **Onboard Electronics:** The primary factor influencing wing size is the weight and volume of the onboard electronics.
- **Weight Distribution:** Ensuring proper weight distribution for stability and performance.
- **Efficiency:** Balancing wing size to achieve optimal aerodynamic efficiency and flight performance.



Wing – Fuselage (Dimensions in inches)

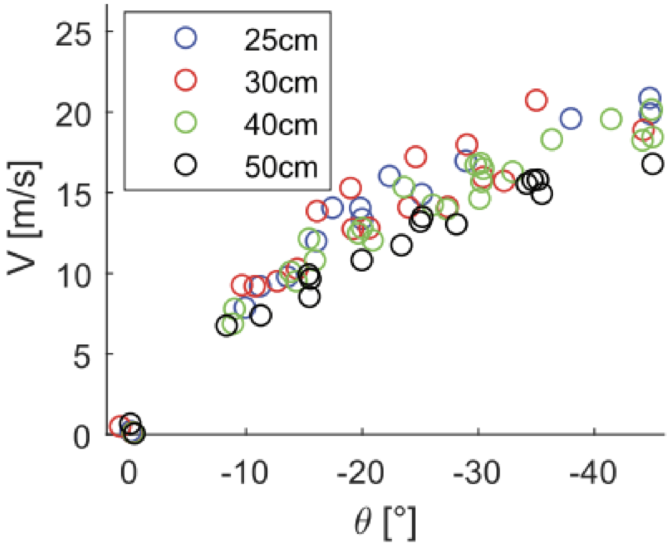
Conventional Quadcopter



Quadfoil Onboard Electronics

By considering the onboard electronics and other factors, we can determine the appropriate wing size to ensure the Quadfoil UAV meets its design goals.

Pitch Angle and Bracket Design Selection



Cruise speed at pitch angles from hover to -45° for four different body diameters and mass 2.1 kg.

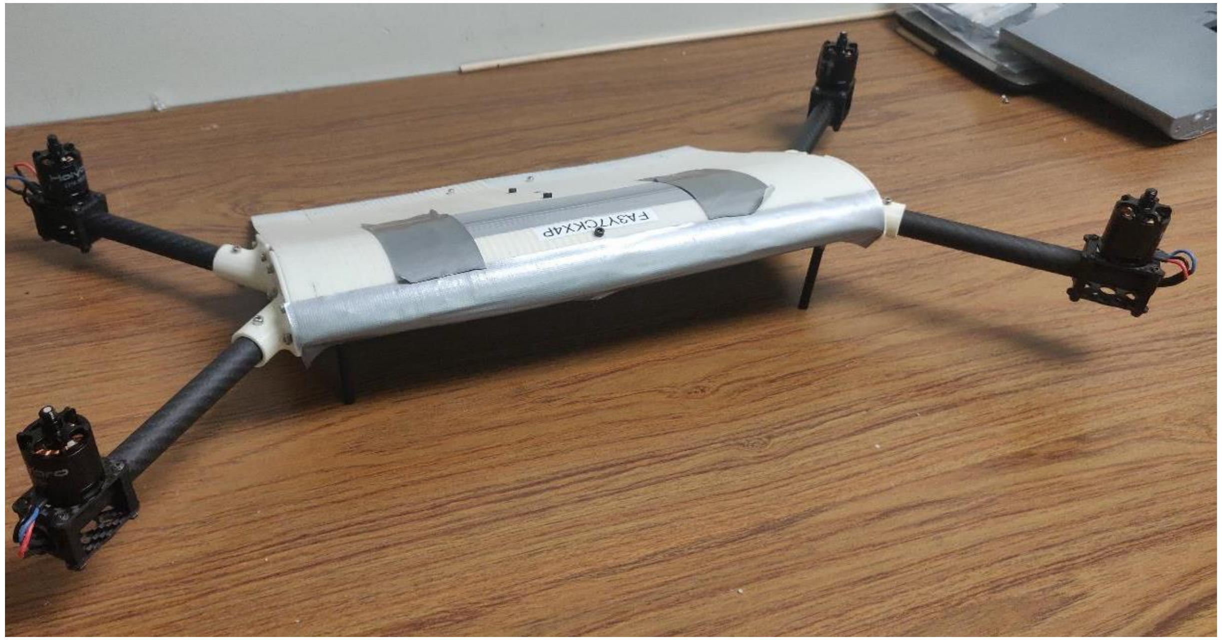
40 cm sphere quadcopter.

[Source:Theys and Deschutter]

Designed brackets for 15^0 AOA

The pitch angle variation with velocity was observed in the study conducted by Theys and Deschutter on the quadcopter shown above. The pitch angle varied from -10° to -25° while traveling at speeds between 5 m/s and 15 m/s. A 15° AOA connector was initially chosen to begin the flight testing.

Prototyping with different manufacturing techniques.



Prototype 1 fabrication using Fused Deposition Modeling (FDM) 3D printing technique.
(Modeled and built with assistance of Katie Moncure)

Prototype 2 fabrication using carbon fiber epoxy layup
(Built with assistance of Soham Dedhia)

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Primary Objectives of Flight Testing

To determine whether the Quadfoil design demonstrates superior efficiency in forward flight compared to conventional quadcopters.

Power Consumption:

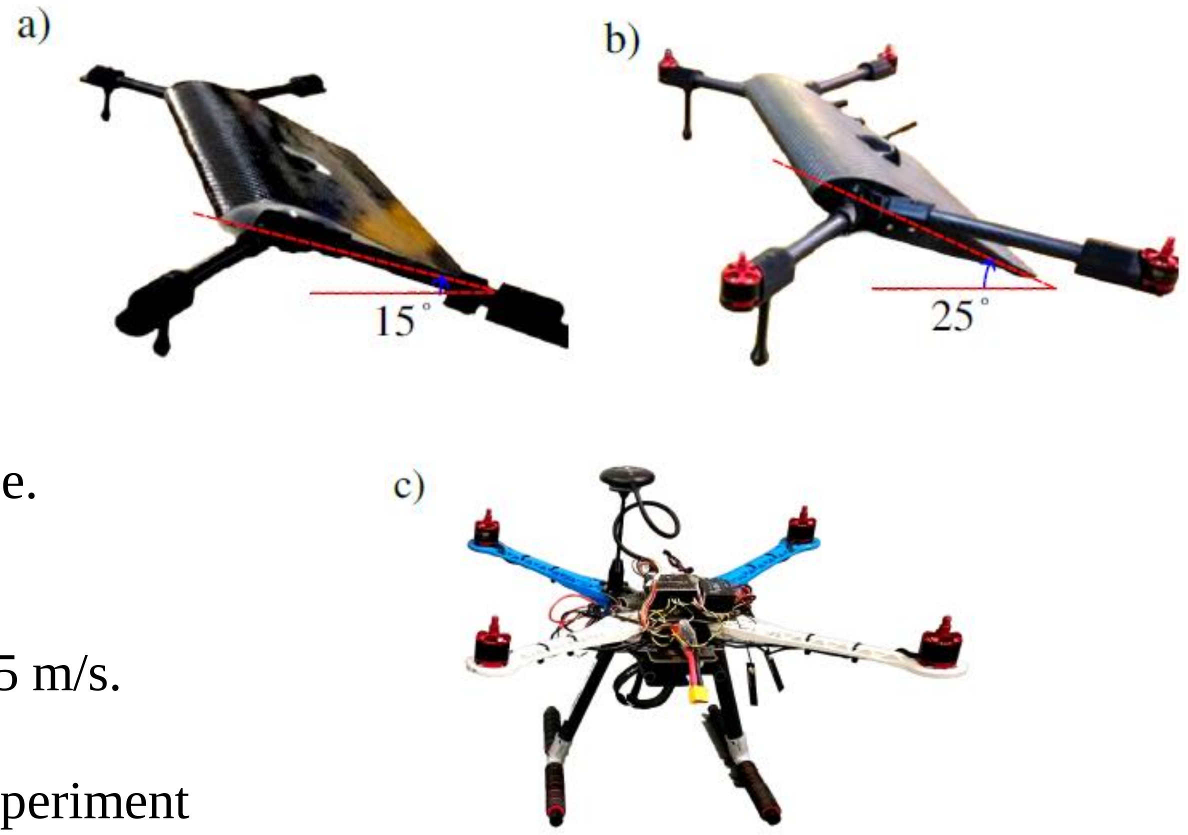
- Measured at speeds from 5 m/s to 15 m/s.
- Insights into power efficiency.

Energy Consumption:

- Total energy used at speeds from 5 m/s to 15 m/s.
- Helps in calculating estimated range and endurance.

Pitch Data:

- Monitored pitch attitude at speeds from 5 m/s to 15 m/s.
- Stability and aerodynamic behavior.
- Important for deciding test matrix or Design of Experiment (DOE) used in wind tunnel study.



(a) 15° connector and (b) 25° connector Quadfoil aircraft
(c) Conventional quadcopter

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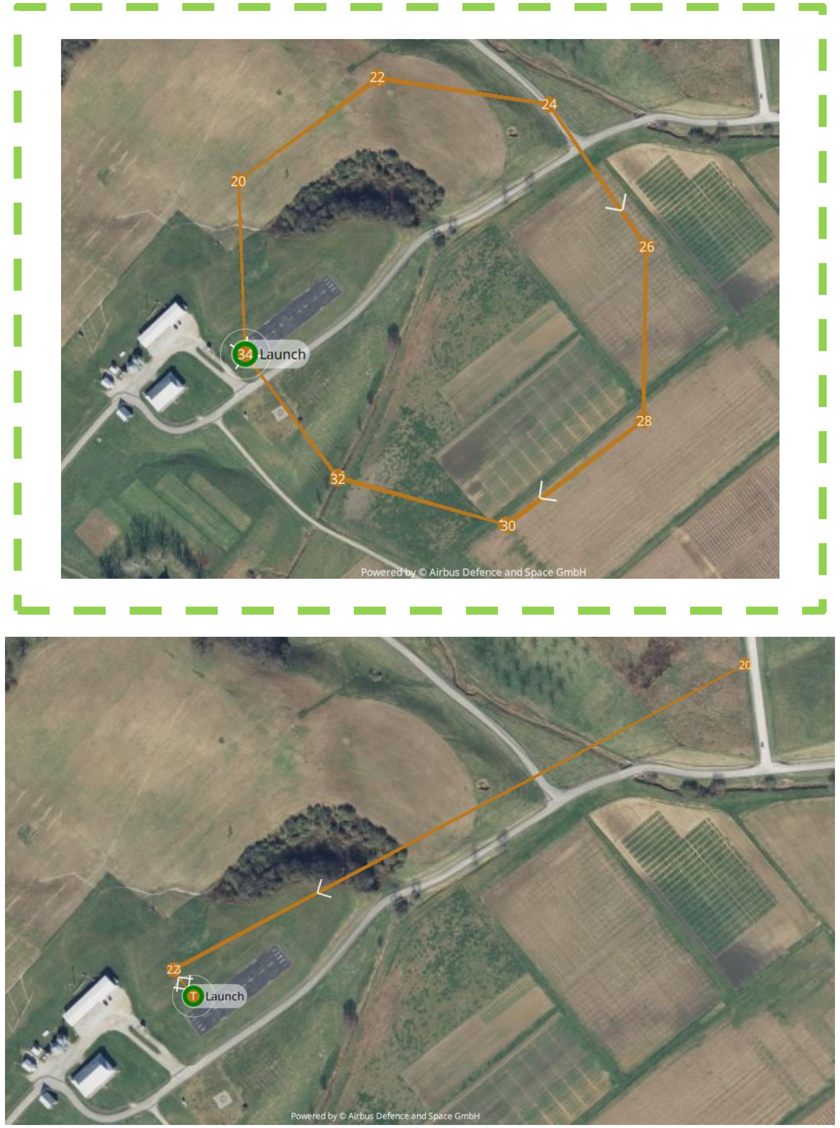
Flight Plan Selection

An octagon-shaped flight plan was selected over out and return flight plan, and the altitude was selected to be 25 m AGL to avoid tree cover. Some of the advantages for octagon flight path are listed below.

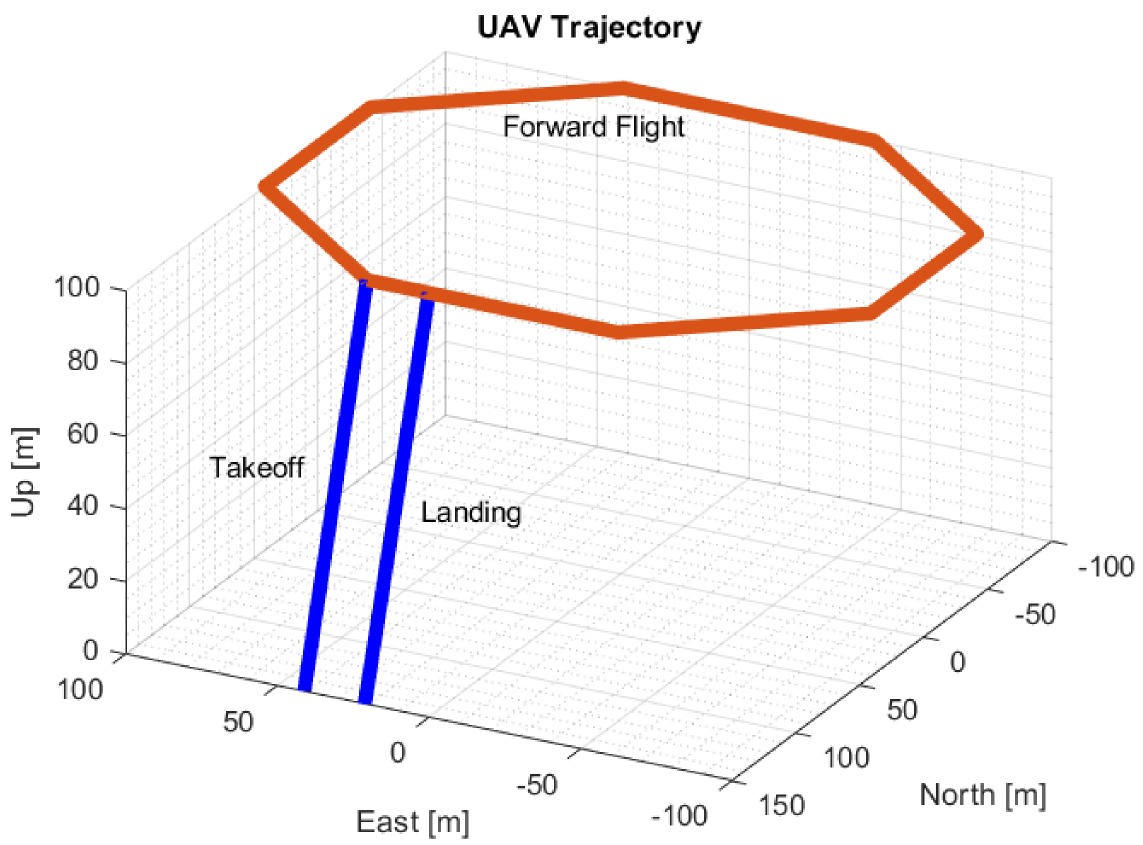
- **Mitigates the risk of Beyond Visual Line of Sight (BVLOS) operation**
- **Avoids Constant Acceleration/Deceleration**
- **Accounts for headwind and tailwind effects.**



Octagon flight path & Out and return flight path.

The octagon-shaped flight plan ensures more consistent and reliable test results.

Mission Planning and Avionics Setup



Octagon mission profile

PARAMETERS	FUNCTION
CUST_ROT_ENABLE	Enables custom rotation for flight Controller (FC) and other sensors
AHRS_ORIENTATION	Overall FC board orientation
COMPASS_ORIENTATION	First external compass orientation
CUST_ROT1_PITCH	Custom pitch for flight controller
CUST_ROT2_PITCH	Custom pitch for GPS + Compass module
BATT1_MONITOR	Enables monitoring the battery voltage and current
WP_NAV_RADIUS	Defines the distance from a waypoint, that when crossed indicates it has been hit.

Important parameters for avionics setup before flight test.

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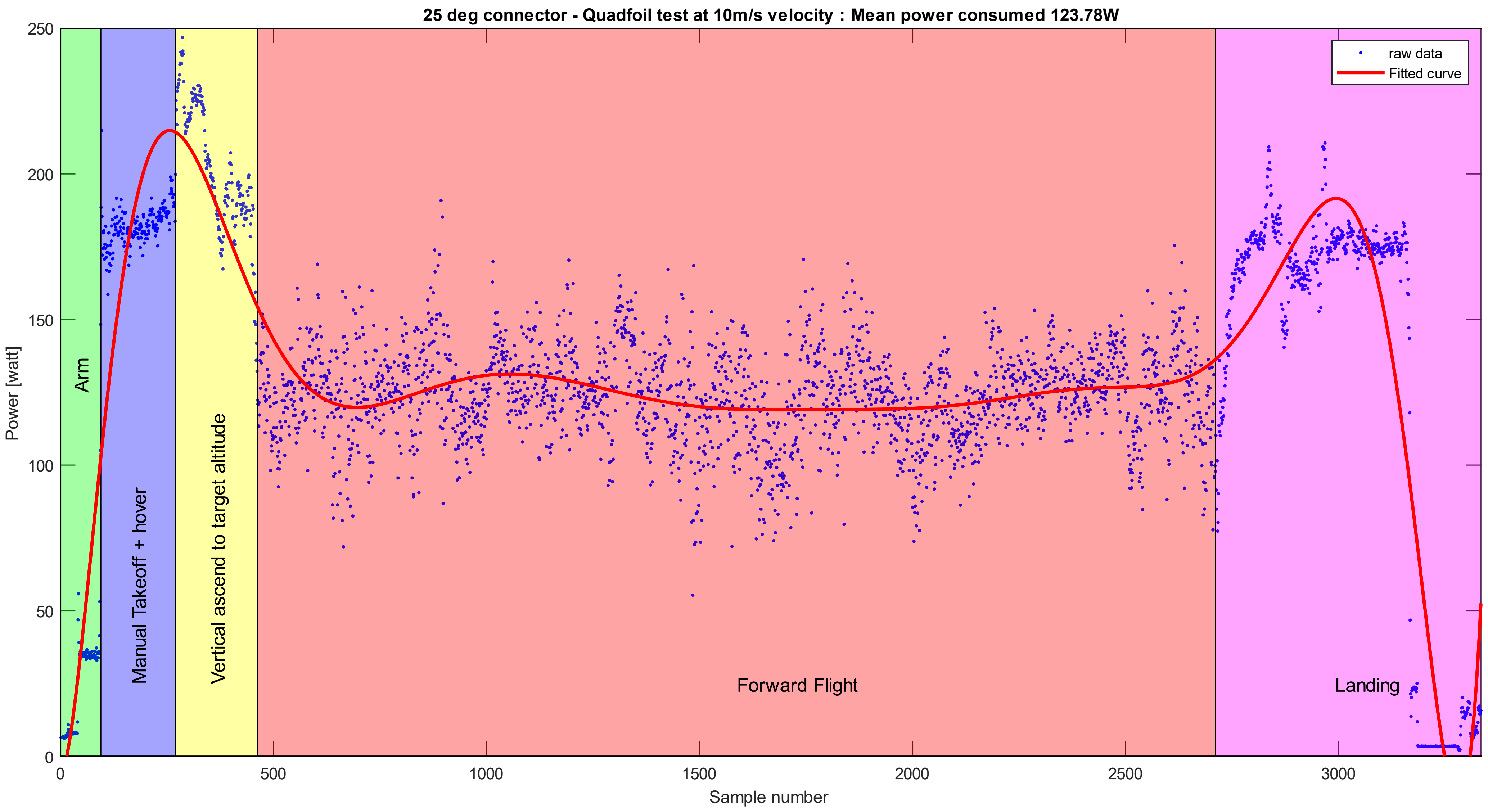
Data Collection and Segregation



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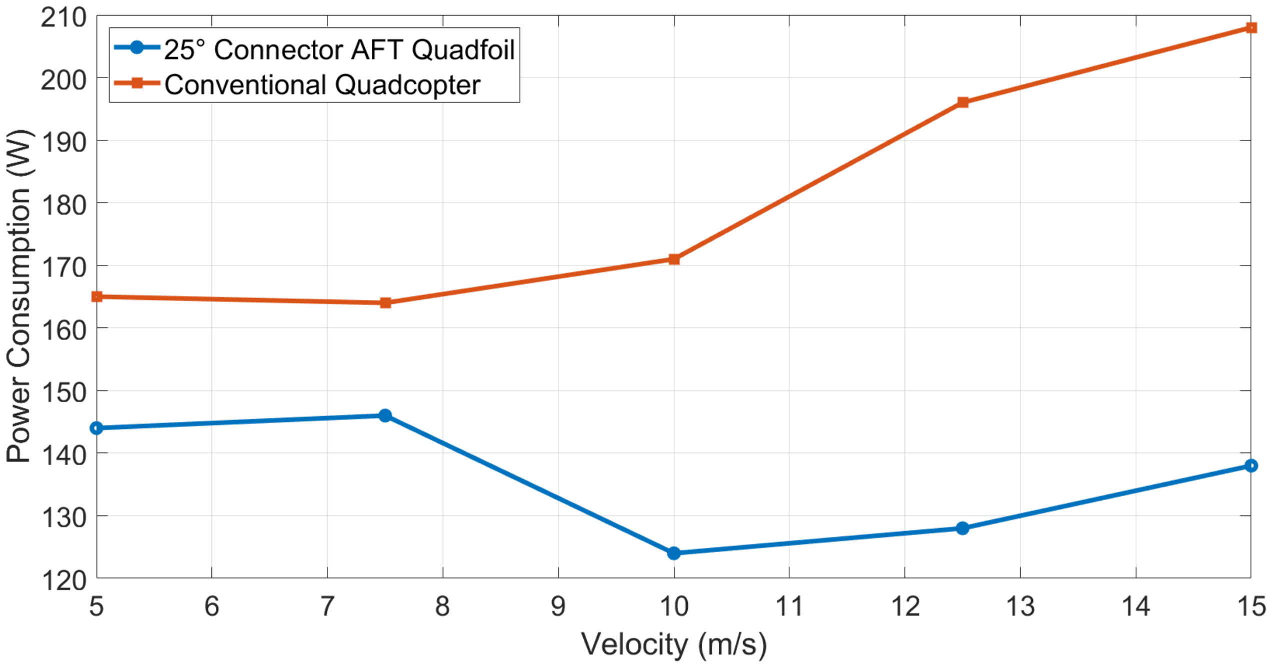
Power Comparison between Quadfoil and Conventional Quadcopter

Velocity (m/s)	Quadfoil Design (QFD) power consumption (W)	Conventional Quadcopter power consumption (W)	Efficiency (%)	QFD Pitch (deg)	QFD AOA (deg)
5	144	165	13	-9.6	15.4
7.5	146	164	11	-11.8	13.2
10	124	171	27	-16.7	8.3
12.5	128	196	35	-18	7
15	138	208	34	-22	3



Quadfoil UAV consumed least power (124W) while flying at 10 m/s.

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Estimated Range and Endurance Comparison for Flight Tested Aircraft

Parameter	25° Connector QFD	Conventional Quadcopter
Velocity (m/s)	10	
Battery Voltage (V)	14.8	
Total Battery Charge (mAh)	3300	
Available Battery Charge (80%)	2640	
Total Distance Covered (m)	2400	
Total Flight Time (minutes)	5.5	
Cruise Power (W)	124	171
Takeoff/Landing Power (W)	191.8	180.8
mAh consumed per unit distance (mAh/m)	0.238	0.3
mAh consumed for 2 octagon rounds	572.5	789.5
Estimated Range (km)	10	7.2
Estimated Endurance (min)	18	14

$$\text{Efficiency \%} = \frac{(Higher\ \text{Value} - Lower\ \text{Value})}{\left(\frac{Higher\ Value + Lower\ Value}{2}\right)} \text{x } 100$$

Quadfoil design

- Endurance improvement = 31.6 %
- Range improvement = 25 %

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CFD Analysis in FlightStream Software

FlightStream: Surface Vorticity Solver
Based on potential flow theory and vorticity transport equation.

Governing Equations

- **Potential Flow Equation:** Assumes irrotational flow
- **Vorticity Transport Equation:** Describes evolution of vorticity in the flow field.
- **Boundary Conditions:** Ensure flow tangency condition on the surface of the body.
- **Panel Method**
  - Discretize surface into small panels
  - Calculate induced flow by summing contributions from all panels.

Potential Flow Equation

$$\nabla^2 \phi = 0$$

Vorticity Transport Equation:

$$\frac{\partial \omega}{\partial t} + (\mathbf{u} \cdot \nabla)\omega = (\omega \cdot \nabla)\mathbf{u} + \nu \nabla^2 \omega$$

Boundary Conditions:

$$\mathbf{u} \cdot \mathbf{n} = 0$$

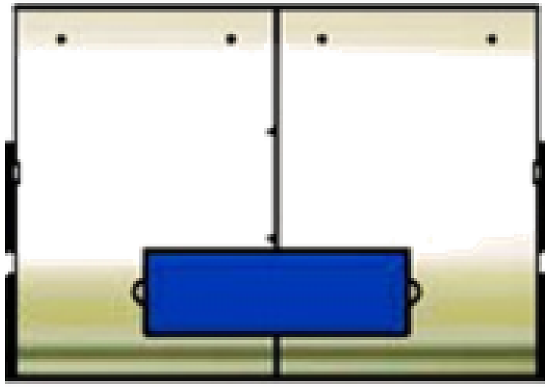
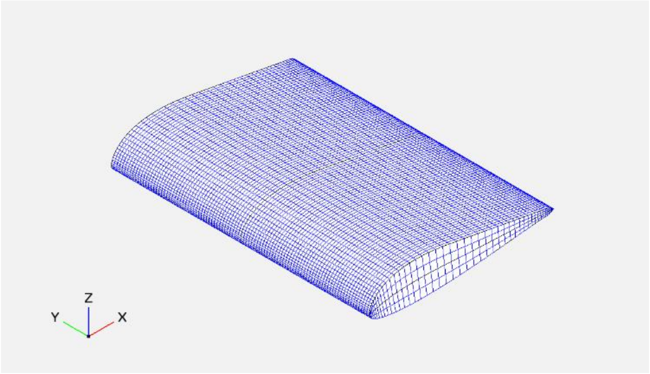
Panel Method:

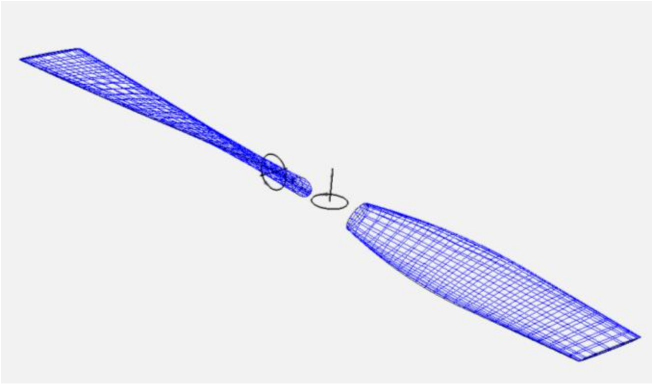
$$\mathbf{u}(\mathbf{r}) = \frac{1}{4\pi} \int_S \frac{\gamma \times (\mathbf{r} - \mathbf{r_t})}{|\mathbf{r} - \mathbf{r_t}|^3} \, dS_t$$

Geometry and Mesh Generation for Wing and Propeller

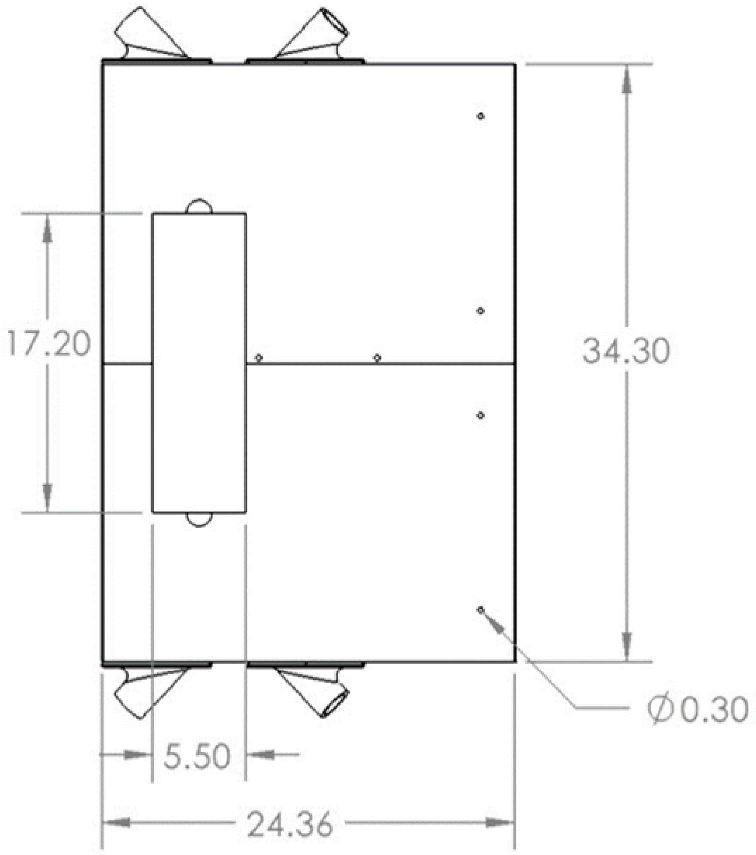
Isolated geometry of wing and propeller was modelled in OpenVSP software for the low fidelity analysis in Flight Stream CFD software.

- Primary mesh elements: triangles and quadrilaterals.
- Rectangle mesh elements used for wing and propeller geometry.
- Tessellation values in OpenVSP:
  - U: divisions along the longitudinal direction (chordwise division).
  - W: divisions along the lateral direction (spanwise division).

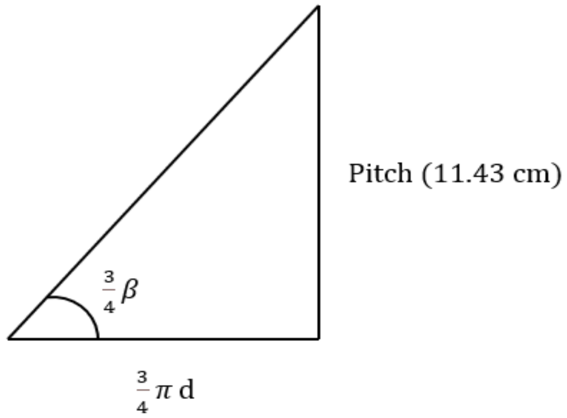



Sizing of Wing and Propeller for Modelling it in OpenVSP Software.



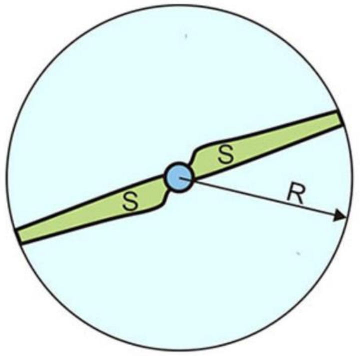
Pitch:
Forward distance traveled by a propeller in one complete revolution



$$\frac{3}{4}\beta = tan^{-1}\left(\frac{11.43}{\frac{3}{4}\pi d}\right)$$

$$\beta = 12^{\circ}$$

Solidity Ratio:
Proportion of total disk area occupied by the blades



2 blade wind rotor

$$\sigma = \frac{Nc}{\pi R} = 0.1257$$

β – Pitch,
σ – Solidity Ratio,
R- Radius,
N- No of Rotor Blades,
c – Mean Aerodynamic Chord

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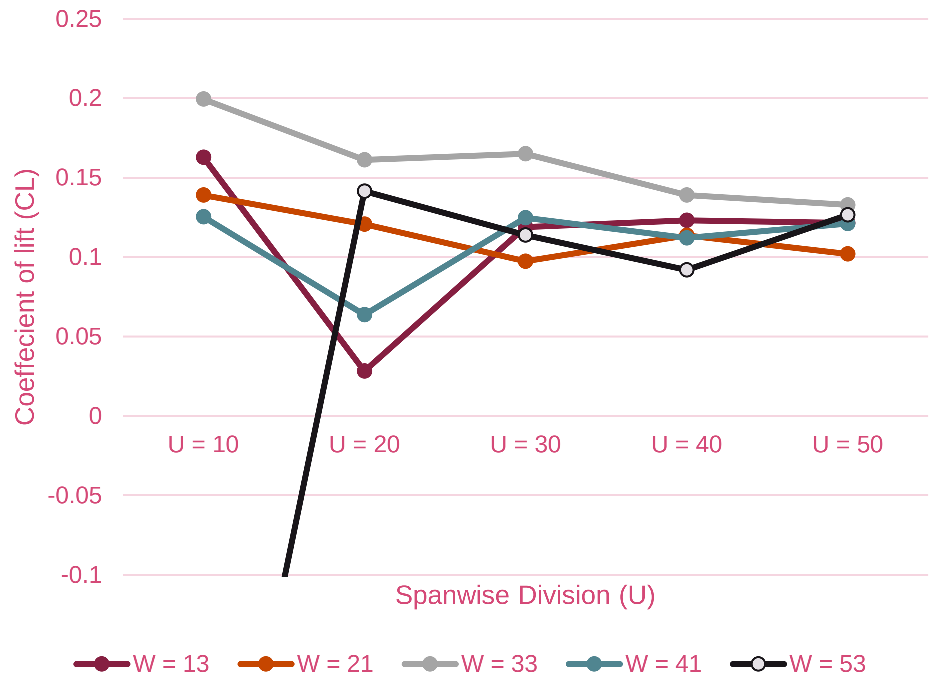
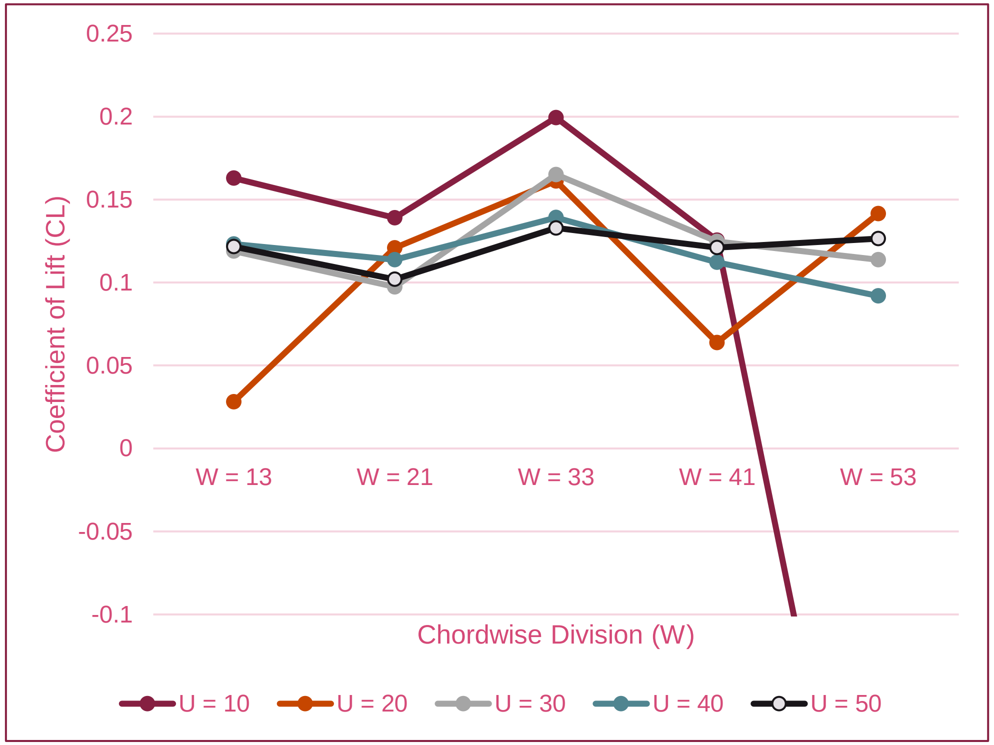
Mesh Quality Assessment and Refinement for Wing Section.



Variation of chordwise division (U =10 to 50) with constant lines of spanwise division (W) .



Variation of spanwise division (W =13 to 53) with constant lines of chordwise division (U) .

CL values for 2-degree AOA, V = 10 m/s for wing section at different mesh resolutions.

The CL value for W = 41 has the least change between U = 30 to U = 50 as seen in the figure on left. Similarly, the CL value for U = 50 has the least change from W = 33 to W = 53 as seen in the figure on right.

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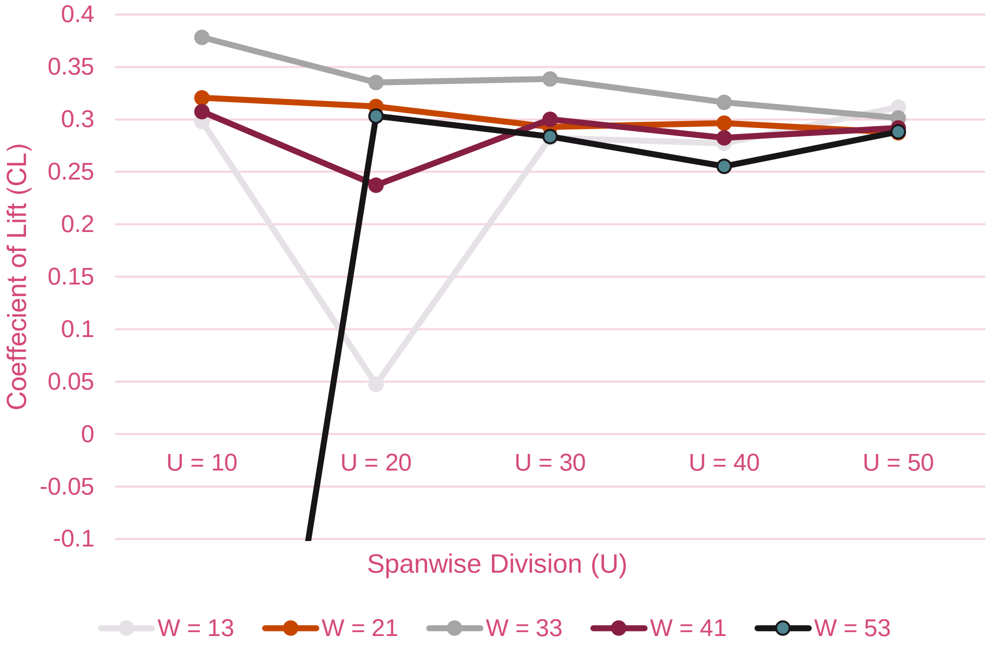

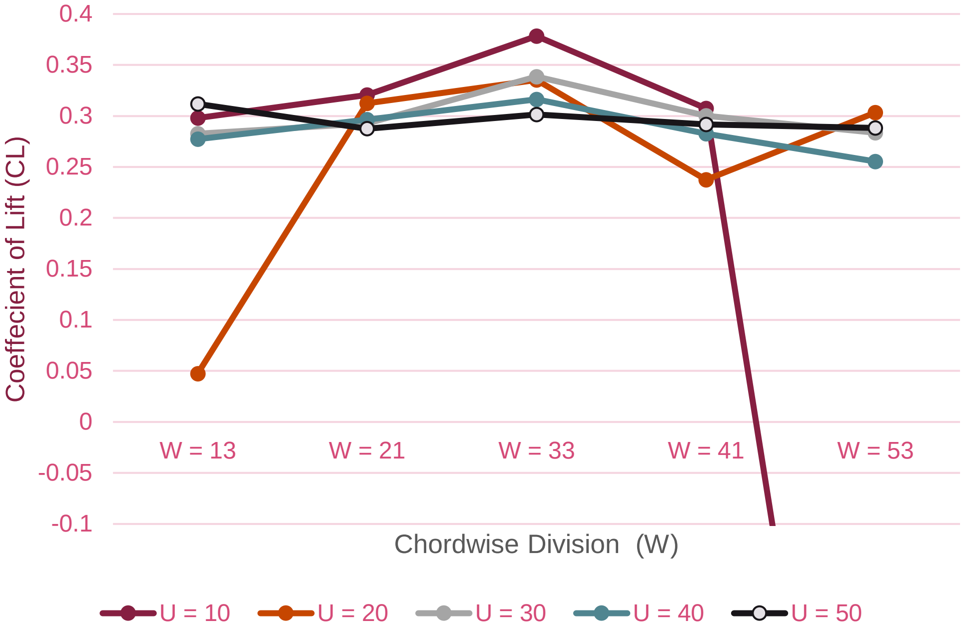

Variation of chordwise division (U =10 to 50) with constant lines of spanwise division (W) .

Variation of spanwise division (W =13 to 53) with constant lines of chordwise division (U) .

CL values for 8-degree AOA, V = 10 m/s for wing section at different mesh resolutions.

The CL value for W = 41 has the least change between U = 30 to U = 50 as seen in the figure on left.
Similarly, the CL value for U = 50 has the least change from W = 33 to W = 53 as seen in the figure on right.

Therefore, the selected chordwise and spanwise division for further study are U = 50 and W = 41.

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Aerodynamic Forces and Coefficient Data for Selected Tessellation (Wing Section).

Angle of Attack (AOA)	Force coefficient along x-axis (Cx)	Force coefficient along y-axis (Cy)	Force coefficient along z-axis (Cz)	Coefficient of lift (CL)	Coefficient of induced drag (CDi)	Zero lift drag coefficient (CDo)	Overall drag coefficient (CD)	CL/CD
2	0.11	0	0.18	0.17	0.01	0.1	0.11	1.54
4	0.11	0	0.24	0.23	0.02	0.11	0.13	1.77
6	0.1	0	0.29	0.28	0.03	0.11	0.14	2
8	0.11	0	0.36	0.35	0.04	0.12	0.16	2.19
10	0.1	0	0.41	0.39	0.05	0.12	0.17	2.29
12	0.1	0	0.45	0.42	0.06	0.13	0.19	2.21
14	0.1	0	0.50	0.47	0.08	0.14	0.22	2.14
16	0.09	0	0.58	0.53	0.1	0.15	0.25	2.12

Aerodynamic coefficients for a wing section having U = 50, W = 41 mesh division

Validation of CFD Data with Static Thrust Stand Test for Propeller Section

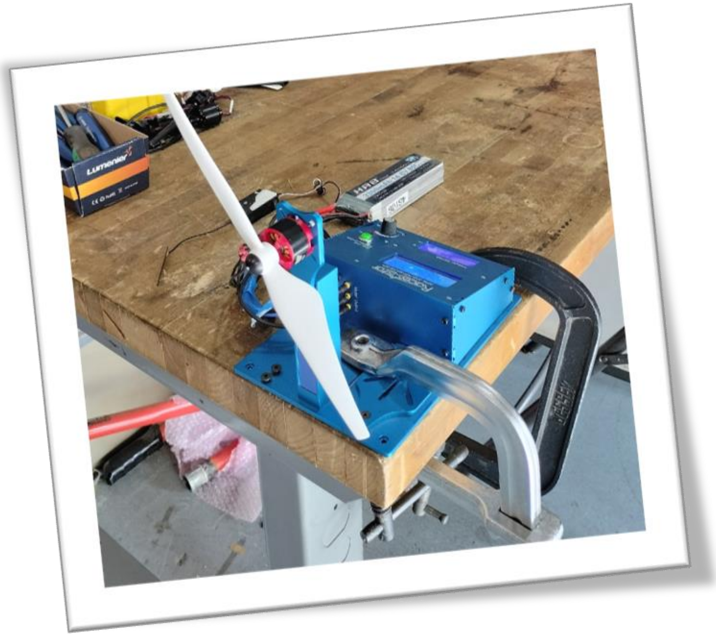
Validation Process

- Conducted static propeller thrust test in the lab.
- Used the same propulsion system as in-flight testing.

Data Utilization

- RPM and power data obtained from the static thrust test was used for rotor (propeller) simulations in Flight Stream CFD software.

RPM	Thrust generated (gf)	Single motor power (W)	Overall Power (W)
5000	175	24	96
5400	209	29	116
5800	**240**	**36**	**144**
6200	280	42.3	169.2
6600	318	50.7	202.8
7000	347	54.1	216.4
7400	382	63.2	252.8
7800	433	75.3	301.2
8200	492	87	348
8600	515	94	376
9000	574	109.3	437.2



Static thrust test stand set up

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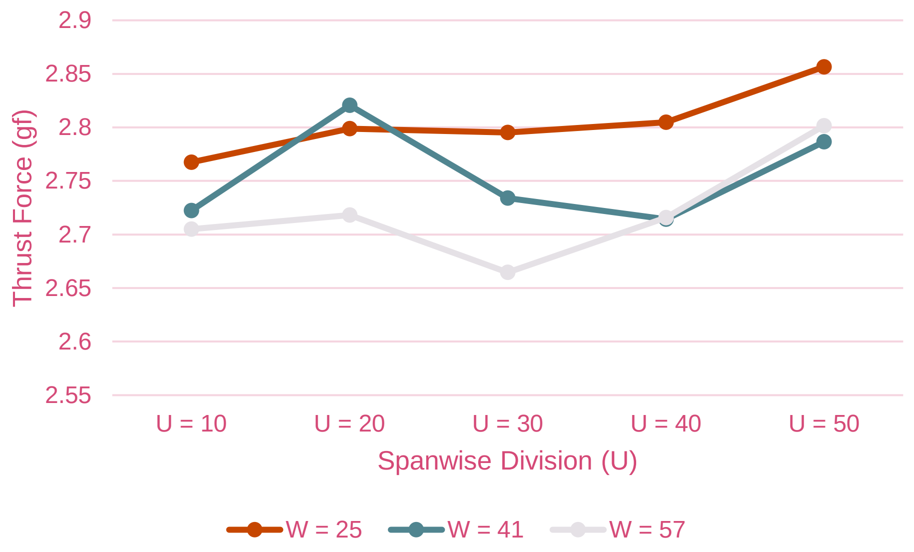
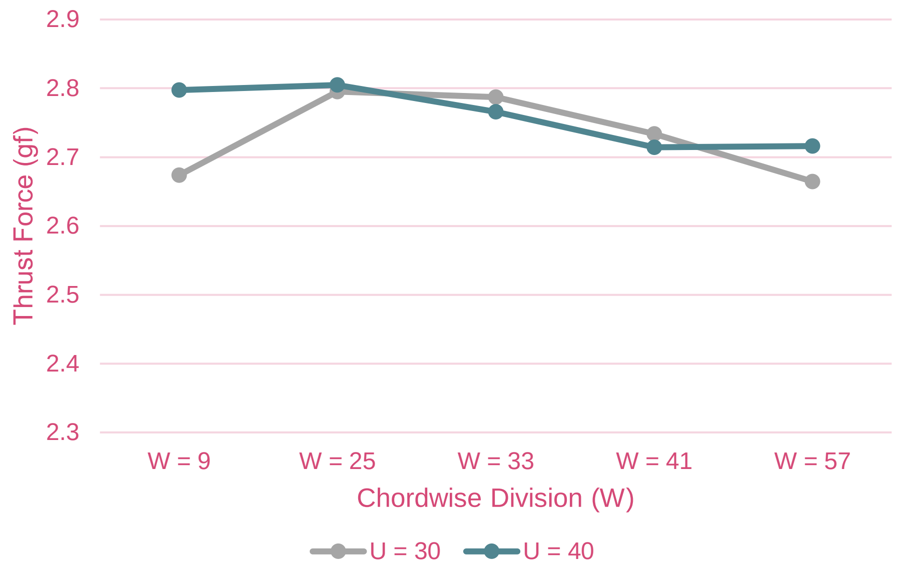
Mesh Quality Assessment and Refinement for Single Propeller.



Variation of chordwise division (U =10 to 50) with constant lines of spanwise division (W) .



Variation of spanwise division (W =9 to 57) with constant lines of chordwise division (U) .

Thrust produced by propeller placed in axial position for V = 10 m/s at different mesh resolutions.

The thrust force value for W = 25 has a least change between U = 20 to U = 40 as seen in the figure on left.
Similarly, the thrust value for U = 40 has the least change from W = 9 to W = 25 as seen in the figure on right.

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Therefore, the selected chordwise and spanwise division for further study are U = 40 and W = 25.

Comparison of tested static thrust values to computed CFD values

Time increment value was decided to provide 10 deg change in blade motion. Time stepping iteration value was decided to provide 2 full rotations of propeller.

Parameter	Value
Time Increment (sec)	0.0003 (Varying with RPM)
Time Stepping Iteration	72
Angular Velocity (rad/s)	586.43 (Varying with RPM)
Angle of attack (deg)	0
Freestream Velocity (m/s)	0.1
Number of Iterations	500
Convergence Threshold	1.00e-05
Reference Velocity (m/s)	0.1
Minimum coefficient of pressure	-100

RPM	Thrust from Test Stand (gf)	Angular Velocity (rad/sec)	CFD Thrust at $\rho = 1.225$ kg/m^3 (gf)	Local Density Adjusted CFD Thrust (gf)	% Difference between Test Stand & CFD Analysis
5600	**220**	586.5066	260.1529	**231.65**	**5.3**
6000	**250**	628.4	294.2915	**262.1**	**4.82**
6400	**290**	670.2933	338.6748	**301.57**	**4**
6800	**327**	712.1866	386.1060	**343.81**	**5.14**
7200	**377**	754.08	434.8827	**387.25**	**2.71**
7600	**408**	795.9733	484.1692	**431.14**	**5.67**
8000	**449**	837.8666	540.6727	**481.45**	**7.22**

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The percentage thrust difference between static thrust data and CFD thrust data comes out to be 5 %, which is reasonable for a low fidelity CFD analysis.

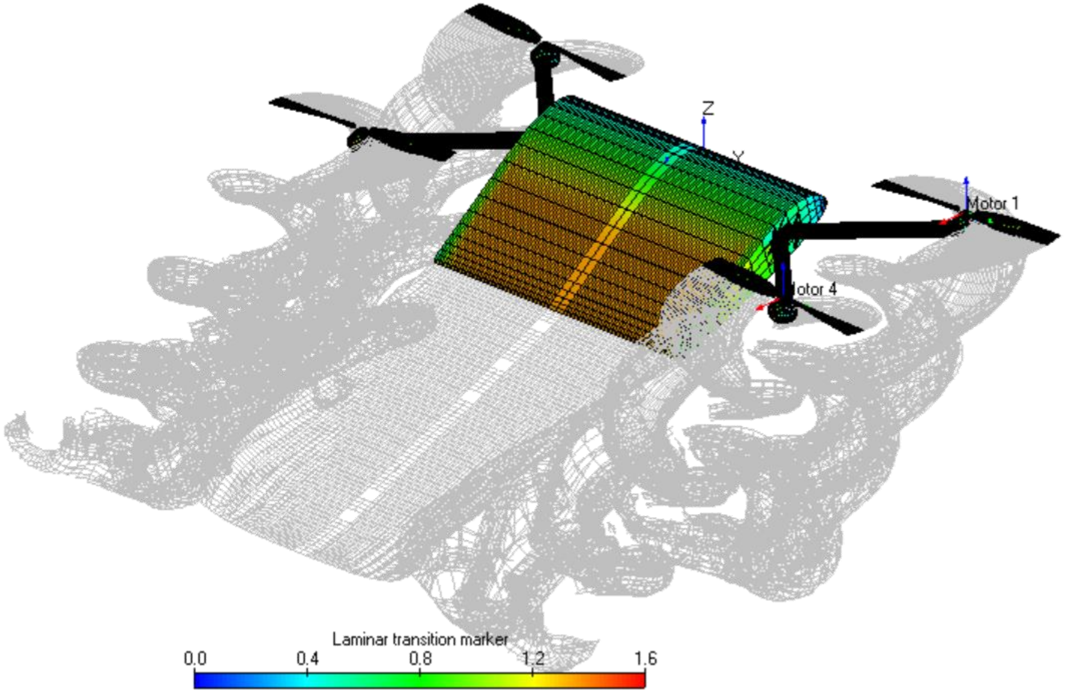
Full Body Simulation: Essential for Power Comparison between CFD and Flight Tests

Simulation Purpose

- To compare power between CFD and flight tests.

Assumption

- Angular velocity for motors obtained from static thrust test data.
- Used same RPM readings for front and back rotors. Should be higher for back rotors in forward flight.
- Ideally should be obtained from experimental flight test logs (RPM sensors).



Velocity (m/s)	Experimental Power Consumption (W)	Overall Torque of Motors from CFD Analysis (Nm)	Angular Velocity (rad/sec)	Power from CFD Analysis (W)
7.5	146	0.6724	607.37	408.39
10	124	0.2156	586.43	**126.43**
15	138	0.633	596.9	377.8

Wind Tunnel Testing of Quadfoil in Low-Speed Tunnel Section at ODU

Objectives :-

1. To find forces acting on the drone about 3 conventional axis with varying velocity.
2. To measure power required to rotate the motors at varying velocity.
3. To obtain trim condition parameters at varying velocity.
4. To evaluate the additional efficiency provided by incorporation of airfoil in the center body.
5. To assess the maneuvering characteristics of the design at 10 m/s velocity.

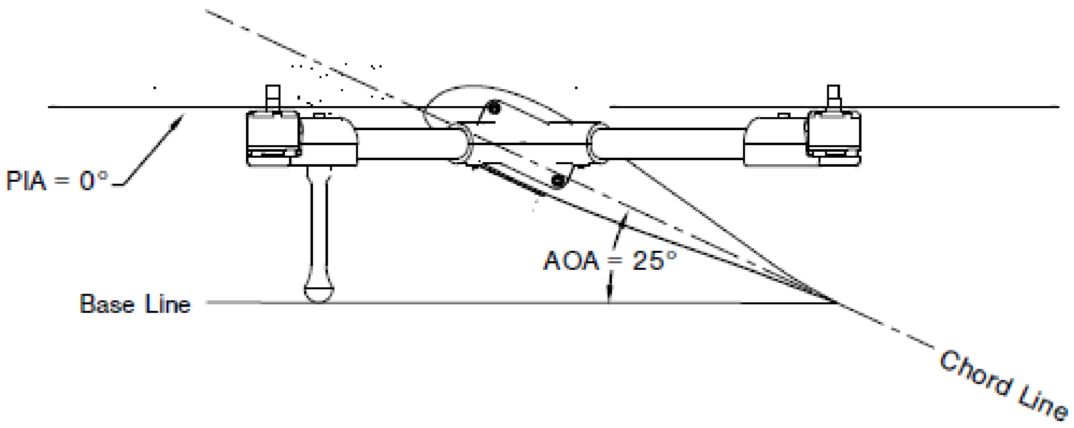
Note : - Tests were conducted after my graduation.

Created by – Sarthak Deshmukh
Aerospace Graduate Researcher

Equipment's Used and Test Carried Out

Equipment Used:-

1. Aircraft – Quadfoil
2. Connector – 25 deg fixed AOA arm connector
3. Propellers – a. 9.4 x 5 propeller , b. 10 x 4.5 propeller

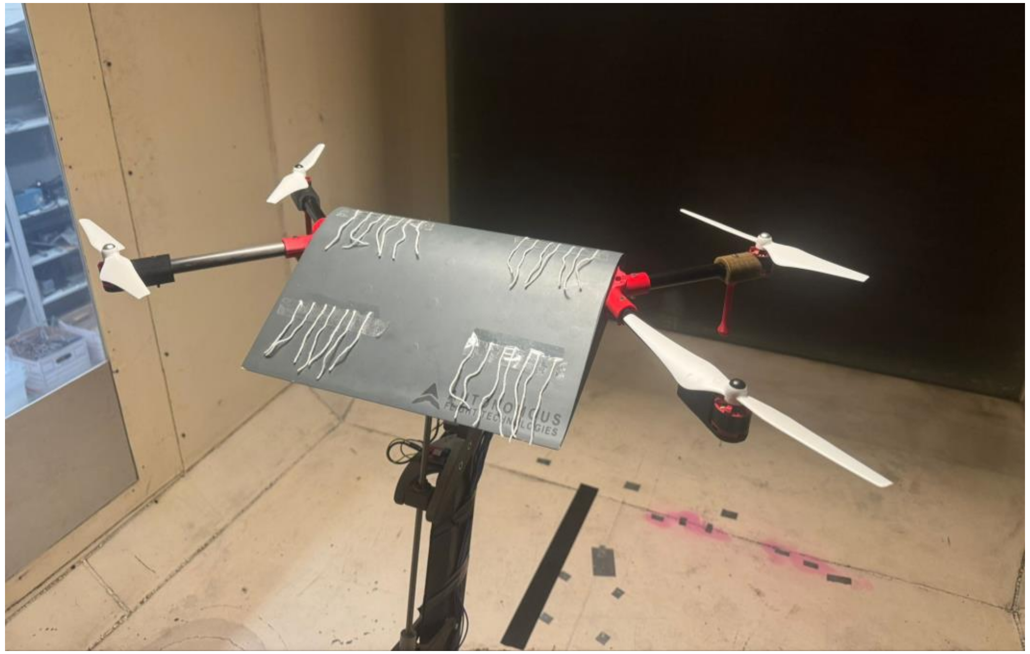


Reference figure for PIA

Tests Carried Out :-

1. Dynamic rotor test with 9.4 x 4.5 propeller at velocity 0, 2.5, 5, 7.5, 10, 12 m/s.
2. Dynamic rotor test with 10 x 4.5 propeller at velocity 0, 2.5, 5, 7.5, 10, 12 m/s.
3. Static rotor test with props off the body at velocity 0, 2.5, 5, 7.5, 10, 12 m/s.
4. Tuft flow visualization test.
5. Dynamic rotor test with pitch sweep (herein Propeller Incidence Angle (PIA) was varied from -15 deg to +10 deg)
6. Roughness test to force the transition in the flow (laminar to turbulent). It may help in delaying flow separation.

Images for Clarity



Tuft flow test

Roughness test

Note : - The tufts were cut in half to prevent them from getting tangled (for the actual test).

Design of Experiment (DOE) to Find Trim Condition Values

- Wind tunnel test matrix was designed to bracket the trim conditions measured in flight testing.
- α_p, n_f and n_r were varied over a regular grid of points known as the face-centered central composite design (FCD).

The central composite design is arguably the most popular second order design from response surface methodology (RSM).

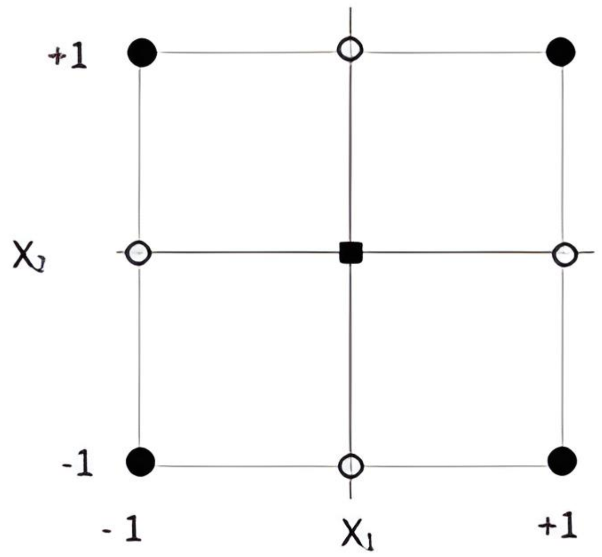
The regression model equation can be written as

$y = \beta 0 + \beta 1 x 1 + \beta 2 x 2 + \beta 12 x 1 x 2 + \beta 11 x_1{}^2 + \beta 22 x_2{}^2 + \epsilon$

Where, y - Response (dependent variable)
x1 and x2 - Independent variables.
β's - Regression coefficients
ϵ - Random error.

- The trim conditions for the level flight are

1. L = W = 2.86 lb (or 1.3 kg)
2. D = 0
3. $M_{CG} = 0$



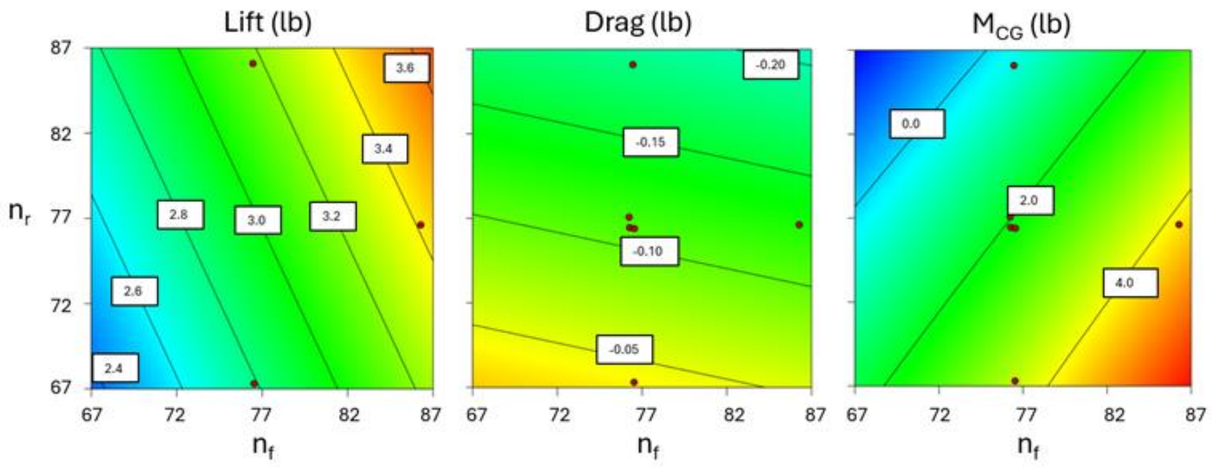
Face centered central composite design in two factors

Created by – Sarthak Deshmukh
Aerospace Graduate Researcher

Tunnel Test Matrix Used for V = 10 m/s & its Corresponding Response Contour

α_p	n_f	n_r	L (lbs)	D (lbs)	M (in-lb)	Power (W)
-15	76.1	76.3	3.11	-0.04	2.26	143
-15	66.7	86.4	2.94	-0.09	-1.39	147
-15	86.2	66.2	3.35	0.05	5.91	143
-15	66.5	66.5	2.49	0.05	1.77	97
-15	86.3	86.3	3.77	-0.08	2.70	192
-15	76.5	86.1	3.96	-0.19	1.81	163
-15	86.1	86.1	3.30	0.19	1.63	144
-17	76.5	76.4	2.98	-0.1	2.18	139
-17	76.5	76.5	2.78	-0.04	3.52	166
-17	67.3	67.3	3.01	0.19	1.98	147
-17	86.3	86.3	3.02	0.19	2.11	139
-17	76.6	76.6	2.96	-0.1	2.09	139
-17	76.6	76.6	2.55	-0.07	2.18	120
-17	86.3	86.3	2.94	-0.09	1.65	97
-17	86.0	86.0	3.00	-0.04	1.77	144
-19	66.5	66.5	2.22	-0.09	1.65	97
-19	86.7	86.7	3.47	-0.32	5.79	192
-19	66.7	76.2	2.59	-0.27	-1.62	147
-19	66.5	66.5	2.22	-0.09	1.65	97
-19	86.7	86.7	3.47	-0.32	5.79	192
-19	86.0	86.0	3.47	-0.32	5.79	192
-19	66.7	66.7	2.59	-0.27	-1.62	147
-19	86.6	86.6	2.75	-0.2	2.03	139
-19	76.2	76.2	2.75	-0.2	2.03	139

The pitch/PIA (α_p) data from the flight test and RPS (n_f and n_r) data from the static thrust test was used to design the 17 runs seen in the table on the left. The α_p varied from -15 to -19 deg, n_f and n_r varied from 67-87 RPS.



Response contours for v= 10 m/s using 9.4 x 5-inch propeller

VIRGINIA TECH.

Created by – Sarthak Deshmukh
Aerospace Graduate Researcher

Results from V = 10 m/s Test Run

Regression coefficients at v = 10 m/s using 9.4 x 5-inch propeller

V = 10 (m/s)	L (lbs)	D (lbs)	M (in-lb)	Power (W)
Intercept	-0.61	0.32	2	118.84
αp	0.08	-0.02	0.56	21.65
n_f	0.04	-0.01	0.14	-1.88
n_r	0.02	-0.01	-0.05	2.46
$\alpha^2{}_p$	-	-	0.02	0.63
$n^2{}_f$	-	-	-	0.03

Summary of fit statistics for v = 10 m/s using 9.45-inch propeller

V = 10 (m/s)	L (lbs)	D (lbs)	M_{CG} (in-lb)	Power (W)
Standard Deviation	0.03	0.01	0.03	1.04
Mean	2.98	0.11	2.12	143.1
R^2	1	0.98	1	1
Adjusted R^2	1	0.98	1	1
Predicted R^2	1	0.97	1	1

Optimizer constraints and solution for v = 10 m/s using 9.45-inch propeller

Constraints				Solution
Name	Goal	Lower Limit	Upper Limit	Trim Condition
αp	in range	-19	-14	-14.01
n_f	in range	67	87	67.05
n_r	in range	67	87	79.43
Lift (lb)	target = 2.86			2.86
Drag (lb)	target = 0			0
M_{CG} (in-lb)	target = 0			0
Power (W)	none			133.75

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Aerospace Graduate Researcher

Trim Condition Data for Quadfoil UAV

Trim condition power data for 9.4 x 4.5 propeller from the wind tunnel study.

- **Table B1 data** is derived from nonlinear regression model executed in MATLAB.
- **Table B2 data** is derived from regression model created using Ordinary Least Square (OLS) method, executed in Design Expert software by Old Dominion University researchers. The power value was calculated from regression model equations for each of the velocities.

Velocity (m/s)	Trim PIA	Front rps	Rear rps	Lift (lb)	Drag (lb)	My (in-lb)	Trim Power (W)
0	-0.4	88.97	86.54	2.86	0	0	204.07
2.5	-5	84.4	87.4	2.86	0	0	186.79
5	-10	81.04	86.01	2.86	0	0	169.72
7.5	-12	77.46	85.21	2.86	0	0	161.91
10	-14	67.29	79.47	2.86	0	0	133.16
12	-17	66.9	79.12	2.86	0	0	125.7

Table B1

Velocity (m/s)	Trim PIA	Front rps	Rear rps	Lift (lb)	Drag (lb)	My (in-lb)	Trim Power (W)
0	-0.5	88.86	86.32	2.86	0	0	201.15
2.5	-5	83.97	86.95	2.86	0	0	184.01
5	-8.4	81.17	86.01	2.86	0	0	170.9
7.5	-12	77.42	85.16	2.86	0	0	160.94
10	-14	67.05	79.43	2.86	0	0	133.78
12	-17	66.71	78.7	2.86	0	0	124.21

Table B2

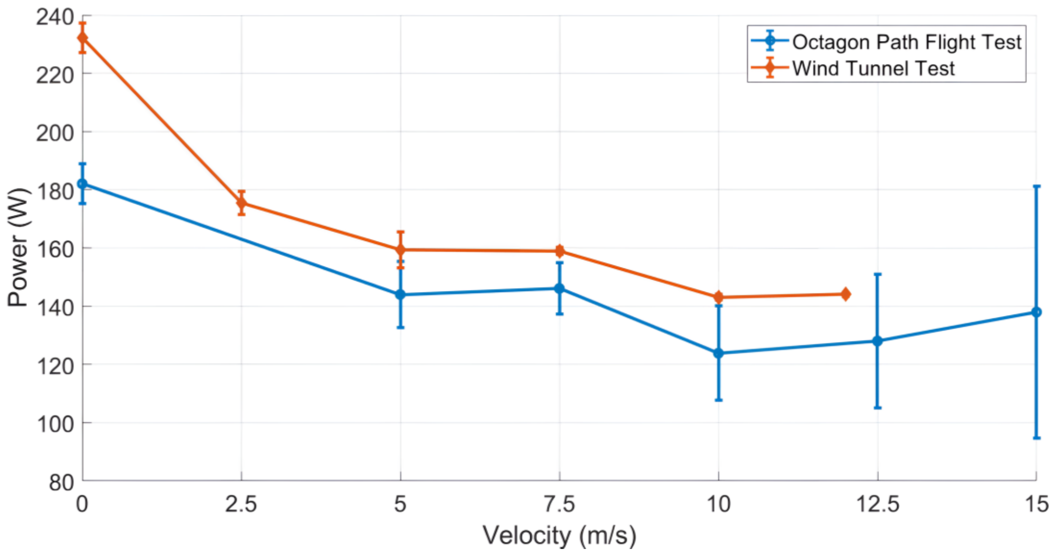
- Both models provide similar values. ODU’s model (Table B2 data) can be considered since there is **enough literature** for it. This data is crucial for **CFD simulation model** and for **designing flight control laws** for this new configuration.

Power Comparison between Flight Test and Wind Tunnel Test

Power comparison between octagon path flight test data and tunnel test data at various velocities.

- Table B3 includes the power data along with **standard deviation** to assess the **uncertainty** at each velocity.

Velocity (m/s)	Octagon Path Flight Test		Wind Tunnel Test	
	Power (W)	Standard Deviation	Power (W)	Standard Deviation
0	182.04	6.77	232.12	5.1
2.5			175.42	3.95
5	144	11.3	159.38	6.18
7.5	146.07	8.75	158.98	1.28
10	123.83	16.2	143.1	1.04
12			122.05	0.78
12.5	128.02	23		
15	137.96	43.3		



Power comparison with uncertainty

The trend for power from both the test is similar. However, there is a discrepancy at v = 0m/s. The local density while conducting flight test was reported to be 1.107 kg/ m^3 , which is lower than the local density while conducting tunnel test (i.e. 1.21 kg/m^3). As per the power consumed during induced hover formula , the flight test power must be higher than the tunnel power. Probable cause for this discrepancy is recirculation effect.

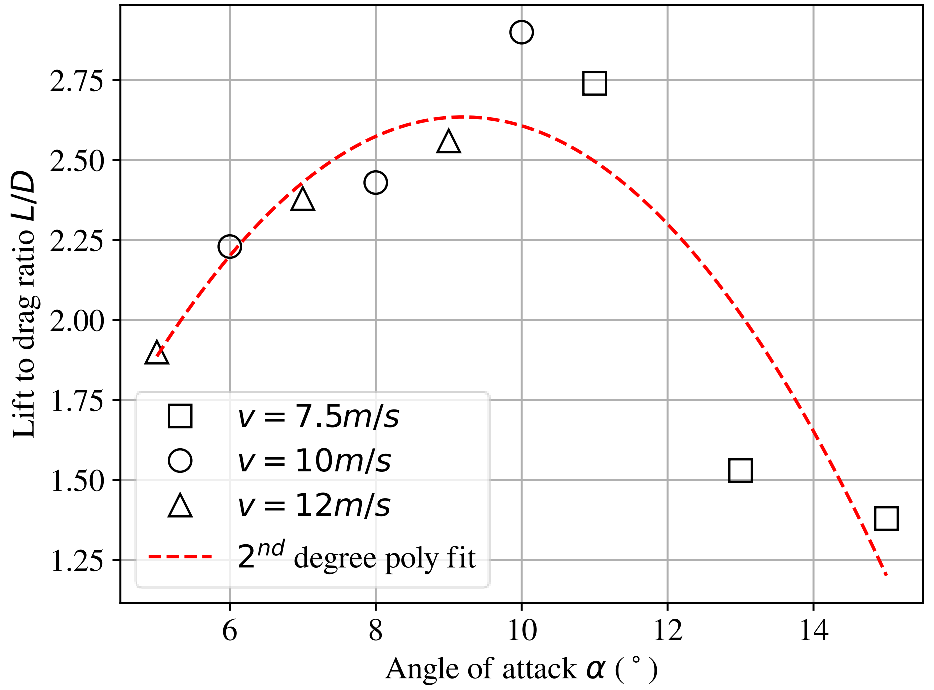
$$P = \frac{T^{3/2}}{\sqrt{2\rho A}}$$

Where,
T – Thrust in N or lbf, ρ - Density in kg/m or slugs/ ft^3
P – Power in W or hp, A – Rotor disc area m^2 or $inch^2$

Created by – Sarthak Deshmukh
Aerospace Graduate Researcher

Prop Off Study to Understand the Aerodynamic Advantage of Wing

Velocity (m/s)	AOA (deg)	Lift (lbf)	Drag (lbf)	L/D ratio
7.5	15	0.21	0.16	1.31
7.5	13	0.24	0.15	1.6
7.5	11	0.33	0.12	2.75
10	10	0.55	0.19	2.89
10	8	0.46	0.19	2.42
10	6	0.38	0.17	2.24
12	5	0.44	0.23	1.91
12	7	0.61	0.26	2.35
12	9	0.71	0.28	2.54



This data can be compared to the CFD simulation results, the mesh resolution as well as boundary conditions of the simulations could be improved if there is larger percentage error.