Group: 5

ME3120 TURBO MACHINERY AND AIRCRAFT PROPULSION

Semester 5

PROPELLER DESIGN PROJECT

By

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Date of submission	09/10/2024
Due date of submission	09/10/2024

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1 Problem Description

An aircraft with a total weight of 1500 kg is intended to be powered by two 2-bladed twin propellers. The propellers are connected to an engine that rotates at 6000 RPM through a 1:2 gearbox. The aircraft must take off at a speed of 220 km/h at sea level within a maximum runway distance of 1 km. At this takeoff speed, the aerodynamic drag acting on the aircraft is 1.8 kN.

The propeller blades are designed using the NACA 0009 aerofoil profile, which has been selected for its balance between aerodynamic performance and structural considerations. The propeller design ensures that the speed of the blade tip remains below the transonic limit of 0.74 Mach. The hub-to-tip width ratio of the propeller is set to 3, and the radius at the propeller hub is determined to be 300 mm due to the size of the engine cover. The propeller blades will be discretized into 20 segments to accurately analyse aerodynamic performance along the span.

The task includes conducting specimen hand calculations for the propeller design, developing an Excel/MATLAB routine to perform the necessary calculations with clear user instructions, and creating a 3D model of the propeller using SOLIDWORKS. Additionally, the required engine power for the aircraft to achieve the desired performance must be calculated. The selection of the NACA 0009 profile will be justified based on its aerodynamic efficiency and suitability for the aircraft's takeoff conditions

1.1 Assumptions and design considerations

- The propeller will be made for climbing rather than cruising because climbing data has been provided.
- Just the axial and radial velocity components are taken into consideration.
- The input of induced flow from other parts is negligible.
- The loading at a given radial position of the blade can be determined independently of the other radial positions.
- Constant accelerations and drag during taxiing of the aircraft.
- During the take-off, angle of attack is selected such that to maintain maximum CL.
- Reynolds number is for the whole propeller is assumed constant and calculated at 50% of blade length.
- Fixed pitch propeller is suggested for the application.
- Constant sound speed during the entire process.

2 Background Research

Propeller design is a critical factor in aircraft thrust and mainly important when planning to achieve adequate force required for take-off and climbing capabilities. In this project, the task is to come up with a two bladed fixed pitch propeller for an aircraft with total weight of 1500 kg to be able to lift off at the speed of 220 km/hr, on a distance of 1 Km, to overcome a drag force of 1.8 kN. The background research of this undertaking focuses on analysing the essential aspects of the propeller together with affirmations for assumptions and decisions made during the design process.

The propeller is designed for climbing rather than for cruising to allow efficient creation of thrust during takeoff and during a climb. In climbing conditions, the aircraft flies at a slower speed than during the cruise but at higher angle of attack. Here, high thrust is required to overcome both, aerodynamic and gravitational loads. According to A.G. Smith’s Anatomy of Propeller, different propellers are designed for different objectives and hence for climbing as implied by the title, propellers are inclined to be more powerful at low rpm as this is important in climbing or taking off an aircraft. The fixed pitch propeller is ideal for this flight regime because, although it is simple and less efficient, high-performance climbing is valued more than efficient cruising.

The choice of the NACA 0009 aerofoil is informed by its symmetric nature because it offers reasonable lift at small angles of attack. The airfoil characteristic of the NACA 0009 is nearly symmetrical which would provide stable characteristics of the airfoil in various conditions and hence more desirable for propellers during dynamic conditions such as take-off or climbing. : Concerning the Theory of Wing Sections advanced by Abbott and von Doenhoff, it is said that symmetrical sections such as the NACA 0009 are suitable for propellers as they provide good efficiency and little drag, especially when used in conditions characterized by fluctuations in the angle of attack. [1].

Due to analysis simplification, for this project the Reynolds number is fixed, and it is computed at 50% of the blade chord length. This assumption is in sync with ground realities in propeller design as described in Frank Delp’s Aircraft Propellers and Controls. The relationship between Reynolds number and pressure distribution is found using the mid-point of the blade, to ease the calculation and at the same time give enough approximation of the aerodynamic forces. Also, due to the assumption that the radial load of each of the propeller blade is not influenced by other sections, it became possible to use two-dimensional lift and drag data at any radial position and support the discretization of the blade into 20 segments, [2].

The blade tip of the propeller should not go at transonic velocities because it causes problems resulting from compressibility and shock waves. In this design, the blade tip speed is below 0.74Mach and this ensures subsonic operation, which doesn't have efficiency losses attributed to transonic shock waves. For the aerodynamic aspects, as pointed out by Ahmed and Koh in the book titled Aircraft Propulsion, controlling the tip speed is important to meet the requirement of high Aero-elastic efficiency and in addition, to prevent the imparting of any structural stress from transonic effects, [3].

3 Calculation

3.1 Design Parameters

Table 1 Design Parameters

Aircraft mass	1500	kg
No of blades B	4	
Aerodynamic drag (at takeoff)	1800	N
Engine Rpm	6000	
Take off speed	61.11	m/s
Runway max distance (m)	800	With 1.25 S.F.
Gear ratio	1: 2	
Propeller rpm	3000	
Angle during takeoff (Assumption)	5	deg
Airfoil	NACA 0009	
Max blade tip velocity	0.74	M
Hub-to-tip width ratio	1: 3	
No of blade segments	20	
No of blades	4	
size of engine cover	300	mm

3.2 Required thrust for take off

- According to assumptions:
  - ✓ Constant accelerations during taxiing.
  - ✓ Ground traction is neglected.
  - ✓ To guarantee safe aeroplane takeoff, a 1.25 safety factor has been set for the airfield. The takeoff distance of 800 meters was chosen since the maximum runway distance is 1 km.
- Required acceleration was calculated using below eqn.,

$$V_{TO}{}^2 = V_0{}^2 + 2 \times a \times s$$
$$61.11^2 = 0^2 + 2 * a * 800$$
$$a = 2.3341 \text{ ms}^{-2}$$

- So, if required thrust is T;

$$F = ma$$
$$2T = D + ma + mgSin(\theta)$$
$$2T = 1800 + 1500 * 2.3341 + 1500 * 9.81 * Sin(5)$$
$$T = 3291.8234N$$

3.3 Final Calculation

- **Maximum blade length**

$$\text{Tip Speed} = 0.74 \times \text{Sound Velocity} = 0.74 \text{ x } 340 = 251.6 \text{ m/s}$$

$$\text{Maximum Diameter} = \frac{2 \times \text{Tip Speed}}{2\pi/60 \times \text{RPM}}$$

$$\text{Maximum Diameter} = \frac{2 \times 251.6}{2\pi/60 \times 3000}$$

$$\text{Maximum Diameter} = 1.6017 \text{ m}$$

So,

$$\text{Blade Length} = \frac{\text{Maximum diameter} - \text{Engine cover diameter}}{2}$$

$$\text{Blade Length} = 0.6509 \text{ m}$$

- **Element size**

$$\text{Element size} = \frac{\text{Blade Length}}{20}$$

$$= 0.0325 \text{ m}$$

- **Reynolds number at 0.5R**

Since the rotational velocity increment from root to tip causes variations in the Reynolds number of each component on a propeller blade, an average value was determined for the Reynolds number, which is equal to the Reynolds number at the blade's 0.5 radius.

$$\text{Reynolds number} = \frac{\mathrm{v} \times \mathrm{l}}{\nu}$$

Where;

V = resultant velocity at 0.5R

L = chord length

ν =kinematic viscosity of air

$$\text{Resultant vel. V } = \sqrt{\mathrm{V_{Rotational\ speed\ at\ 0.5R}}^2 + \mathrm{V_{Forward}}^2}$$

$$= \sqrt{125.8^2 + 61.11^2}$$

$$= 139.8578 \text{ m/s}$$

So,

$$\text{Reynolds number} = \frac{139.8573 \text{ x } 0.15}{0.00001562}$$

$$= 1343060.1248$$

- **Data obtained from Aerofoil Tools according to the calculated Reynolds number for NACA 0009 air foil.**

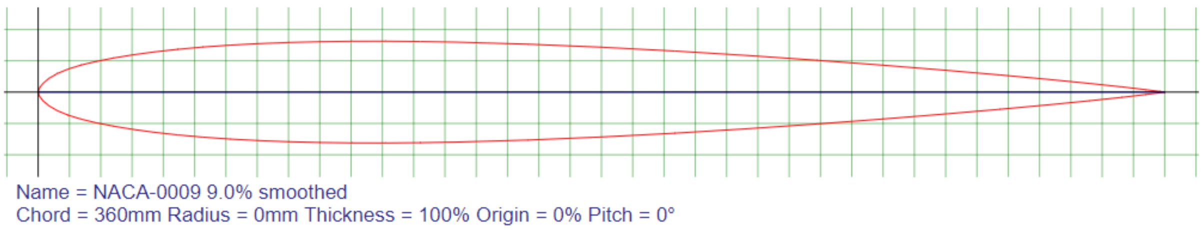


Figure 1 NACA0009 aerofoil

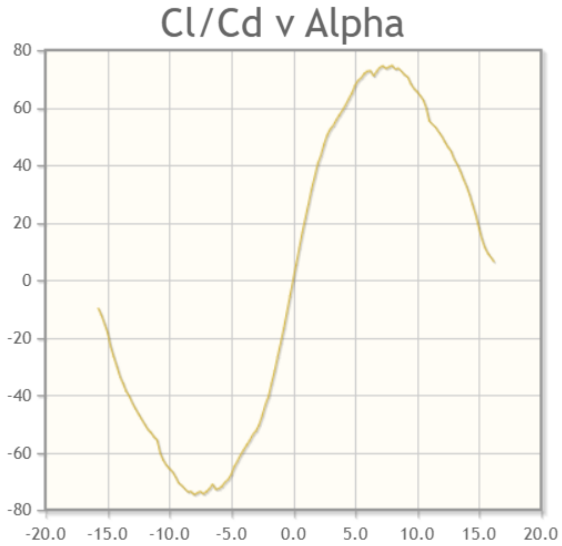


Figure 2 Variation of CL/CD vs AOA

- **Why choosing max C_L/C_D than max C_L when designing propeller**.

A fixed-pitch propeller's design must balance generating enough thrust with maintaining aerodynamic efficiency to be used for climbing. Optimising the lift-to-drag ratio (CL/CD) guarantees the propeller works effectively by reducing drag and generating sufficient lift. The maximum (CL/CD) for the NACA 0009 aerofoil is attained at around 7 degrees angle of attack (AoA), at which point the propeller reaches its maximum efficiency. But at this point, the lift coefficient might not be high enough to produce the thrust needed for take-off and climbing, which means this condition is unacceptable for our design, where thrust is crucial, [4].

It is more reasonable to choose an angle of attack (AoA) that is marginally higher than the point of maximum (CL/CD) but below the stall angle rather than focussing only on maximising (CL). This method keeps the propeller within a safe working range where the flow stays linked to the blade, preventing stall, but nevertheless enables the propeller to generate higher CL, boosting thrust. The design maintains reasonable aerodynamic efficiency while providing enough thrust for ascending by choosing an AoA that is near, but not at, the stall angle, which is approximately 12-14 degrees. For dependable take off performance free from the possibility of stalling or excessive drag, this balance is essential, [5].

Alpha	CL	CD	CDp	CM	Top Xtr	Bot Xtr	CL/CD
-3.5	-0.3864	0.00309	-0.0013	-0.0015	0.9052	0.0774	-125.05
-3	-0.3322	0.00295	-0.0014	-0.0009	0.8624	0.1182	-112.61
-2.5	-0.2775	0.00281	-0.0015	-0.0006	0.8135	0.1718	-98.754
-2	-0.2225	0.00269	-0.0015	-0.0003	0.7578	0.2364	-82.714
-1.5	-0.1672	0.0026	-0.0016	-0.0002	0.6979	0.3034	-64.308
-1	-0.1115	0.00254	-0.0016	-0.0001	0.6367	0.371	-43.898
-0.5	-0.0558	0.0025	-0.0016	0	0.5732	0.4398	-22.32
0.5	0.0558	0.0025	-0.0016	0	0.4407	0.5723	22.32
1	0.1115	0.00253	-0.0016	0.0001	0.3732	0.6371	44.0711
1.5	0.1671	0.0026	-0.0016	0.0002	0.3027	0.6991	64.2692
2	0.2224	0.0027	-0.0015	0.0003	0.2332	0.7571	82.3704
2.5	0.2775	0.00281	-0.0015	0.0006	0.1716	0.8128	98.7544
3	0.3322	0.00295	-0.0014	0.0009	0.1162	0.8631	112.61
3.5	0.3865	0.00309	-0.0013	0.0014	0.077	0.9052	125.081
4	0.4402	0.00323	-0.0011	0.0021	0.0526	0.9387	136.285
4.5	0.4931	0.00338	-0.001	0.0031	0.0369	0.9645	145.888
5	0.5471	0.00355	-0.0009	0.0038	0.0275	0.9828	154.113
5.5	0.6076	0.00376	-0.0007	0.0029	0.0212	0.9917	161.596
6	0.67	0.00398	-0.0005	0.0016	0.0169	0.9964	168.342
6.5	0.7347	0.00424	-0.0003	-0.0003	0.0138	0.9981	173.278
7	0.7985	0.0045	-2E-05	-0.002	0.012	0.9998	177.444
7.5	0.8493	0.0048	0.0003	-0.0008	0.0107	1	176.938
8	0.8986	0.00516	0.00068	0.0008	0.0094	1	174.147
8.5	0.9494	0.00551	0.00104	0.002	0.0086	1	172.305
9	0.9999	0.00599	0.00157	0.0032	0.0077	1	166.928
9.5	1.0517	0.00641	0.00203	0.0041	0.0072	1	164.072
10	1.1026	0.00693	0.0026	0.0051	0.0067	1	159.105
10.5	1.152	0.00763	0.00338	0.0063	0.0062	1	150.983
11	1.2018	0.00826	0.00409	0.0075	0.0059	1	145.496
11.5	1.2504	0.00901	0.00492	0.0087	0.0055	1	138.779
12	1.2959	0.01004	0.00606	0.0103	0.0051	1	129.074
12.5	1.339	0.01126	0.00744	0.0122	0.0049	1	118.917
13	1.3822	0.01237	0.00866	0.0141	0.0047	1	111.738

Figure 3 xfoil data

According to aerofoil tool data and xfoil data CL**= 1.2018 occurs at AOA= 11 degrees**

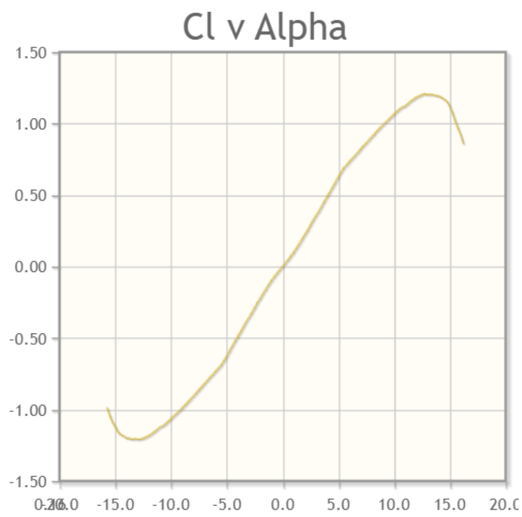


Figure 4 Variation of CL vs AOA

- **Calculating element radius and for each blade element**

Element radius is the distance from hub to the centre of each blade element.

Sample calculation;

Element radius for blade element 1 = 0.15+(0.0325/2)

= 0.1663 m

Element radius for blade element 2 = 0.1663 + 0.0325

= 0.1988 m

3.3.1 Iterative solution procedure for blade element theory

- **Initial guesses for inflow factors Ua and U_θ**

  It is assumed that Ua and U_θ are 1/10th of V_f and V_t respectively.

- **Pitch Angles**

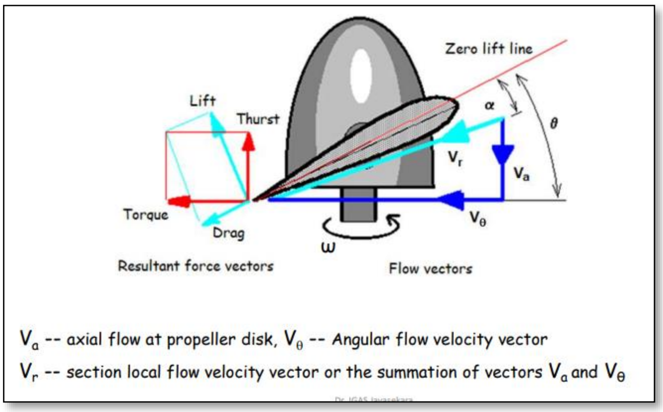


Figure 1 Velocity profile of a blade

Here we have given the opportunity to have the optimum Angle of Attack of 11 degrees for each blade element when calculating pitch angles.

Sample calculation for blade element 01;

$$V_a = V_f + u_a$$
$$V_a = 61.11 + 6.11$$
$$V_a = 67.22 \text{ m/s}$$

$$V_\theta = V_t - u_\theta$$

$$V_\theta = 52.23 - 5.223$$

$$V_\theta = 47.01 \text{ m/s}$$

$$\varphi = \tan^{-1}\left({}^{V_a}/_{V_\theta}\right)$$

$$\varphi = \tan^{-1}({}^{67.22}/_{47.01})$$

$$\varphi = 55.03 \text{ deg}$$

Element	Radius	Vt	Approximate Pitch angle	Initial Uθ	Vθ	Vf	Va	chord	Element Edges	Mean chord
1	0.16625	52.22897787	66.03629135	5.222897787	47.00608008	61.1111	67.2222	0.36	0	0.354
2	0.19875	62.43915399	61.10565061	6.243915399	56.19523859	61.1111	67.2222	0.348	1	0.342
3	0.23125	72.64933011	56.79402433	7.264933011	65.3843971	61.1111	67.2222	0.336	2	0.33
4	0.26375	82.85950624	53.03218519	8.285950624	74.57355561	61.1111	67.2222	0.324	3	0.318
5	0.29625	93.06968236	49.74814306	9.306968236	83.76271413	61.1111	67.2222	0.312	4	0.306
6	0.32875	103.2798585	46.8742224	10.32798585	92.95187264	61.1111	67.2222	0.3	5	0.294
7	0.36125	113.4900346	44.35017093	11.34900346	102.1410311	61.1111	67.2222	0.288	6	0.282
8	0.39375	123.7002107	42.12397673	12.37002107	111.3301897	61.1111	67.2222	0.276	7	0.27
9	0.42625	133.9103869	40.15151985	13.39103869	120.5193482	61.1111	67.2222	0.264	8	0.258
10	0.45875	144.120563	38.39572369	14.4120563	129.7085067	61.1111	67.2222	0.252	9	0.246
11	0.49125	154.3307391	36.82556246	15.43307391	138.8976652	61.1111	67.2222	0.24	10	0.234
12	0.52375	164.5409152	35.41509868	16.45409152	148.0868237	61.1111	67.2222	0.228	11	0.222
13	0.55625	174.7510914	34.14262345	17.47510914	157.2759822	61.1111	67.2222	0.216	12	0.21
14	0.58875	184.9612675	32.98992116	18.49612675	166.4651407	61.1111	67.2222	0.204	13	0.198
15	0.62125	195.1714436	31.94165554	19.51714436	175.6542992	61.1111	67.2222	0.192	14	0.186
16	0.65375	205.3816197	30.98486393	20.53816197	184.8434578	61.1111	67.2222	0.18	15	0.174
17	0.68625	215.5917959	30.10854377	21.55917959	194.0326163	61.1111	67.2222	0.168	16	0.162
18	0.71875	225.801972	29.30331547	22.5801972	203.2217748	61.1111	67.2222	0.156	17	0.15
19	0.75125	236.0121481	28.56114796	23.60121481	212.4109333	61.1111	67.2222	0.144	18	0.138
20	0.78375	246.2223242	27.87513479	24.62223242	221.6000918	61.1111	67.2222	0.132	19	0.126
								0.12	20	

Figure 6 Calculated Pitch Angles & Mean chords

- **Estimating thrust and torque values using Blade Element Theory.**

If the number of propeller blades is B, then:

$$\Delta T = \tfrac{1}{2}\rho V_r^2(\ C_L\ \cos(\phi)\ -\ C_D\ \sin(\phi))Bcdr....(1)$$

$$\Delta\tau = \tfrac{1}{2}\rho V_r^2(C_D\ \cos(\phi)\ +C_L\ \sin(\phi))Brcdr....(2)$$

Figure 7 Eqns. for torque and thrust using blade element theory

Sample Calculation for blade element 1;

$$\mathrm{Vr} = \sqrt{\mathrm{Va}^2 + \mathrm{V\theta}^2}$$

$$\mathrm{Vr} = \sqrt{67.22^2 + 47.01^2}$$

$$\mathrm{Vr} = 82.03 \text{ m/s}$$

$$\varphi = \tan^{-1}\left({}^{\mathrm{V_a}}/_{\mathrm{V_\theta}}\right)$$

$$\varphi = \tan^{-1}({}^{67.22}/_{47.01})$$

$$\varphi = 0.9605 \text{ rad}$$

At AoA= 11 degrees;

$C_L = 1.2018$

$C_D = 0.00826$

So,

$$\Delta T = \frac{1}{2} \times \rho \times {V_r}^2 \times (C_L \cos(\varphi) - C_D \sin(\varphi)) \times B \times c \times dr$$

$$\Delta T = \frac{1}{2} \times 1.184 \times 82.03 \times (1.2018 \cos(0.9605) - 0.00826 \sin(0.9605)) \times 4 \times 0.177 \times 0.0325$$

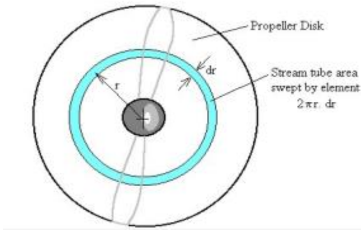
$$\Delta T = 129.332 \text{ N}$$

$$\Delta \tau = \frac{1}{2} \times \rho \times {V_r}^2 \times (C_D \cos(\varphi) + C_L \sin(\varphi)) \times B \times c \times r \times dr$$

$$\Delta \tau = \frac{1}{2} \times 1.184 \times 82.03^2 \times (0.00826 \cos(0.9605) + 1.2018 \sin(0.9605)) \times 4 \times 0.177 \times 0.16625 \times 0.0325$$

$$\Delta\tau = 5.0054 \text{ Nm}$$

Then by using these approximate values of thrust and torque, improved estimates of inflow factors U_a and U_θ were calculated from momentum theory.



$$\Delta T = \rho 4\pi r(V_f + u_a).u_a.dr\ldots\ldots\ldots$$
$$\Delta\tau = 4\pi\rho r^2(V_f + u_a)(\omega r - u_\theta).dr.$$

Figure 8 Eqns. for thrust and torque from momentum theory

From the thrust eqn. a quadratic eqn. for U_a can be obtained. And then by solving that, upgraded Ua can be found. From torque eqn. only unkown variable U_θ can be found.

New Ua = 19.330 m/s

New U_θ = 13.438 m/s

This process was repeated until values for (Ua) and (U_θ) have converged to a 10^{-4} tolerance.

Initial iterations.

Element 01										
U_a (m/s)	U_θ (m/s)	V_a (m/s)	V_θ (m/s)	V_r (m/s)	ϕ (degrees)	θ (degrees)	ΔT (N)	Δτ (Nm)	U'_a (m/s)	U'_θ (m/s)
6.111	5.223	67.222	47.006	82.027	55.036	66.036	129.332	31.203	19.330	13.438
19.330	13.438	80.441	38.791	89.306	64.255	75.255	115.682	40.592	17.657	2.298
17.657	2.298	78.768	49.931	93.261	57.630	68.630	156.033	41.551	22.450	11.782
22.450	11.782	83.561	40.447	92.836	64.171	75.171	125.393	43.834	18.853	2.511
18.853	2.511	79.964	49.718	94.160	58.129	69.129	156.834	42.586	22.541	11.204
22.541	11.204	83.652	41.025	93.170	63.876	74.876	127.666	44.041	19.129	2.950
19.129	2.950	80.240	49.279	94.164	58.444	69.444	155.432	42.732	22.382	10.766
22.382	10.766	83.493	41.463	93.222	63.591	74.591	129.126	43.984	19.305	3.366
19.305	3.366	80.417	48.863	94.098	58.716	69.716	153.993	42.794	22.218	10.378
22.218	10.378	83.329	41.851	93.248	63.333	74.333	130.390	43.912	19.457	3.741
19.457	3.741	80.569	48.488	94.034	58.960	69.960	152.691	42.844	22.070	10.031
22.070	10.031	83.181	42.198	93.272	63.101	74.101	131.523	43.846	19.594	4.077
19.594	4.077	80.705	48.152	93.978	59.178	70.178	151.527	42.889	21.937	9.719
21.937	9.719	83.048	42.510	93.295	62.893	73.893	132.544	43.788	19.716	4.379
19.716	4.379	80.827	47.850	93.929	59.374	70.374	150.487	42.930	21.817	9.439
21.817	9.439	82.929	42.790	93.317	62.707	73.707	133.463	43.737	19.826	4.649
19.826	4.649	80.937	47.580	93.886	59.550	70.550	149.557	42.968	21.711	9.188
21.711	9.188	82.822	43.041	93.338	62.540	73.540	134.291	43.692	19.924	4.891
19.924	4.891	81.035	47.338	93.849	59.708	70.708	148.726	43.002	21.615	8.962
21.615	8.962	82.726	43.267	93.358	62.390	73.390	135.035	43.651	20.013	5.108
20.013	5.108	81.124	47.121	93.816	59.850	70.850	147.983	43.033	21.529	8.759

Finally converged values obtained for required tolerance;

20.788	6.989	81.899	45.240	93.563	61.084	72.084	141.604	43.319	20.788	6.989
20.788	6.989	81.899	45.240	93.563	61.084	72.084	141.604	43.319	20.788	6.989
20.788	6.989	81.899	45.240	93.563	61.084	72.084	141.604	43.319	20.788	6.989
20.788	6.989	81.899	45.240	93.563	61.084	72.084	141.604	43.319	20.788	6.989
20.788	6.989	81.899	45.240	93.563	61.084	72.084	141.604	43.319	20.788	6.989
20.788	6.989	81.899	45.240	93.563	61.084	72.084	141.604	43.319	20.788	6.989
20.788	6.989	81.899	45.240	93.563	61.084	72.084	141.604	43.319	20.788	6.989
20.788	6.989	81.899	45.240	93.563	61.084	72.084	141.604	43.319	20.788	6.989
20.788	6.989	81.899	45.240	93.563	61.084	72.084	141.604	43.319	20.788	6.989
20.788	6.989	81.899	45.240	93.563	61.084	72.084	141.604	43.319	20.788	6.989
20.788	6.989	81.899	45.240	93.563	61.084	72.084	141.604	43.319	20.788	6.989
20.788	6.989	81.899	45.240	93.563	61.084	72.084	141.604	43.319	20.788	6.989
20.788	6.989	81.899	45.240	93.563	61.084	72.084	141.604	43.319	20.788	6.989
20.788	6.989	81.899	45.240	93.563	61.084	72.084	141.604	43.319	20.788	6.989
20.788	6.989	81.899	45.240	93.563	61.084	72.084	141.604	43.319	20.788	6.989
20.788	6.989	81.899	45.240	93.563	61.084	72.084	141.604	43.319	20.788	6.989
20.788	6.989	81.899	45.240	93.563	61.084	72.084	141.604	43.319	20.788	6.989
20.788	6.989	81.899	45.240	93.563	61.084	72.084	141.604	43.319	20.788	6.989
20.788	6.989	81.899	45.240	93.563	61.084	72.084	141.604	43.319	20.788	6.989
20.788	6.989	81.899	45.240	93.563	61.084	72.084	141.604	43.319	20.788	6.989
20.788	6.989	81.899	45.240	93.563	61.084	72.084	141.604	43.319	20.788	6.989
20.788	6.989	81.899	45.240	93.563	61.084	72.084	141.604	43.319	20.788	6.989
20.788	6.989	81.899	45.240	93.563	61.084	72.084	141.604	43.319	20.788	6.989

- Same procedure will be followed for all 20 blade segments until Ua and U_θ is converged for 4 decimal places and obtained the thrust and torques produced by each component to calculate the total torque and thrust generated by propeller.

$$\text{Total thrust} = \sum \Delta T$$

$$\text{Total torque} = \sum \Delta\tau$$

We compare the calculated total thrust value with the required thrust, which is determined by the drag force necessary to counteract the aircraft mass and acceleration during takeoff. If the total thrust which produced by the propeller is less than the needed thrust to force, we change the design parameters; the changes begin with the chord length. In the context of the present work, the proposed concepts are associated with the increase or optimisation of the chord length in order to improve the lift and thrust to be generated by the blades of a propeller. Moreover, if the thrust is not maximized, we consider the number of blades, as the distribution of load between a larger number of blades will allow for greater total thrust. We are then able to arrive at the desired propeller through a series of these modifications to meet the performance demands for take-off in the aircraft in question.

3.4 Thrust and torque generates by the proposed propeller

Element	Radius	Thrust	Torque	V_a (m/s)	V_θ (m/s)	V_r (m/s)	θ (degrees)
1	0.16625	141.60427	43.3187	81.8989	45.2401	93.5633	72.084189
2	0.19875	136.5599	46.8108	78.5869	46.2899	91.2067	70.500714
3	0.23125	62.770876	42.4833	68.9769	47.8351	83.9405	66.258919
4	0.26375	143.89154	55.8919	75.5457	52.0588	91.7457	66.429129
5	0.29625	150.9837	60.7834	74.7408	55.7957	93.2703	64.257815
6	0.32875	159.02644	65.7078	74.1506	59.8342	95.2809	62.098946
7	0.36125	79.298687	61.621	67.6015	63.7257	92.9028	57.69047
8	0.39375	176.15478	75.293	73.3091	68.4707	100.312	57.954515
9	0.42625	184.64079	79.7987	72.9758	72.9701	103.199	56.002226
10	0.45875	192.76879	84.0151	72.6691	77.5506	106.277	54.138786
11	0.49125	200.34365	87.8696	72.3743	82.1933	109.516	52.365113
12	0.52375	207.18419	91.2898	72.0806	86.8847	112.892	50.679511
13	0.55625	213.11758	94.2033	71.7799	91.6145	116.386	49.078678
14	0.58875	217.97611	96.5383	71.4661	96.3754	119.982	47.558369
15	0.62125	221.5953	98.2239	71.1339	101.161	123.667	46.113845
16	0.65375	223.81272	99.1909	70.7792	105.968	127.432	44.740163
17	0.68625	224.46727	99.372	70.3982	110.791	131.266	43.43236
18	0.71875	223.39875	98.7023	69.9878	115.629	135.161	42.185579
19	0.75125	220.44758	97.1205	69.545	120.479	139.11	40.995134
20	0.78375	215.45463	94.5689	69.0668	125.339	143.109	39.856551
	Final	3595.4976	1572.8				

So, the propeller provides the required thrust which requires at take-off.

3.5 Required engine power

- Advance Ratio

$$J = \frac{V}{N \times D}$$

$$J = \frac{61.11}{(3000/60) \times 1.6017}$$

$$J = 0.7631$$

- Coefficient of thrust

$$C_T = \frac{T}{\rho \times N^2 \times D^4}$$

$$C_T = \frac{3595.4976}{1.225 \times (3000/60)^2 \times 1.6017^4}$$

$$C_T = 0.1784$$

- Torque Coefficient

$$C_\tau = \frac{\tau}{\rho \times N^2 \times D^5}$$

$$C_\tau = \frac{1572.8}{1.225 \times (3000/60)^2 \times 1.6017^5}$$
$$C_\tau = 0.078$$

- Propeller efficiency

$$\eta_{propeller} = \frac{J \times C_T}{2\pi \times C_\tau}$$
$$\eta_{propeller} = \frac{0.7631 \times 0.1784}{2\pi \times 0.078}$$
$$\eta_{propeller} = 0.2778$$

- Propeller Power

$$\begin{aligned}\text{Propeller power} &= T \times V \\ &= 3595.4976 \times 61.11 \\ &= 219.723\ \text{kW}\end{aligned}$$

- **Minimum Engine Power required**

$$\text{Engine power required} = \frac{219.723}{0.2778} = 790.94\ \text{kW}$$

4 3D Model

Figure 9: Isometric view

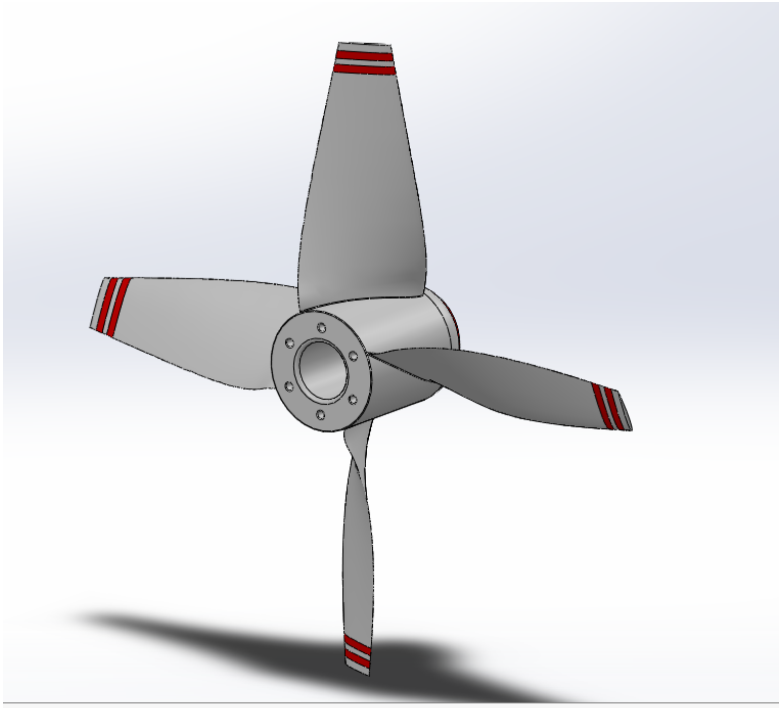


Figure 10: Isometric back view

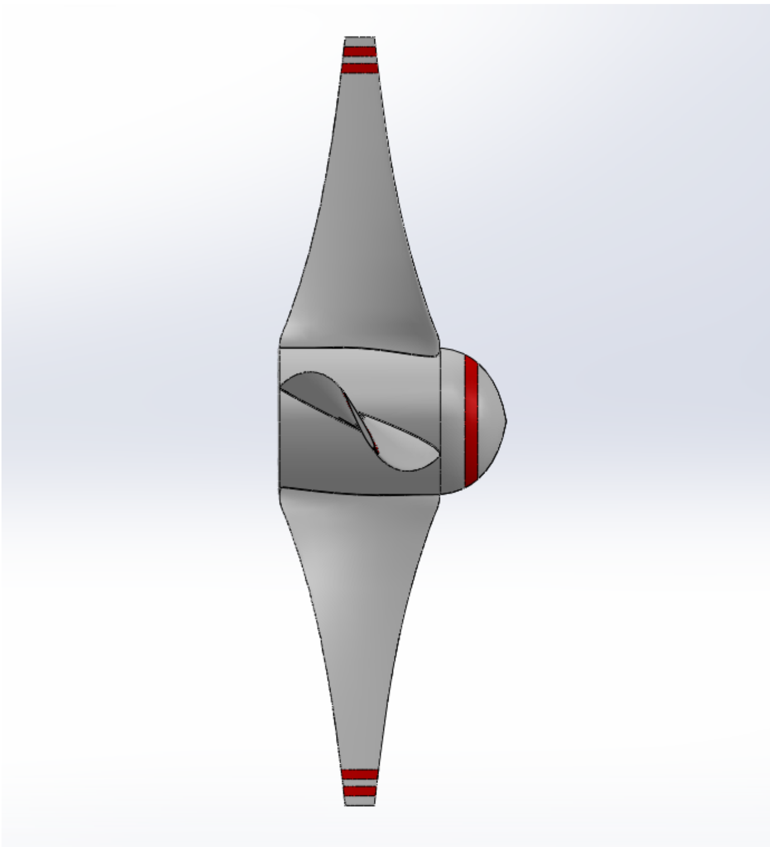


Figure11: side view

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